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Inspring Designs in Aeronautics, Astronautics and Wind Energy

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Delft Aerospace Design Projects 2016

Delft Aerospace Design Projects 2016

Inspiring Designs in
Aeronautics, Astronautics and Wind Energy

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PREFACE

The Design Synthesis Exercise forms the closing piece of the third year of the Bachelor degree course in aerospace engineering at TU Delft. Before the students move on to the first year of their Master degree course, in which they join one of the Faculty's disciplinary groups in preparation for their final year MSc thesis project, they learn to apply their acquired knowledge from all aerospace disciplines in the design synthesis exercise.

The objective of this exercise is to improve the students' design skills while working in teams with nine to ten of their fellow students for a continuous period of approximately ten weeks with a course load of 400 hours. They apply knowledge acquired in the first years of the course; improve communication skills and work methodically according to a plan.

Despite the fact that the final designs result from a design process executed by small groups of students with limited experience, it may be concluded that the designs are of good quality. Not only the members of the scientific staff of the Faculty of Aerospace Engineering have expressed their appreciation of the results, but also the external experts and industry, which have supported the design projects

This book presents an overview of the results of the Fall Design Synthesis Exercise 2015 and the Spring Design Synthesis Exercise of 2016, based on summaries of each of the projects. The Design Synthesis Exercise Coordination Committee, responsible for the organisation and execution of the exercise, has made this book with the aim to present an overview of the diverse nature of the various design topics, and of the aerospace engineering course itself. In addition, the book is intended as an incentive for further improvements to the exercise.

Finally the coordinating committee would like to thank the student-assistants, the academic counsellors, the educational office and all who have contributed to the success of this year's exercise.

The Design Synthesis Exercise Coordination Committee 2015-2016:
ir. V.P. Brügemann, dr.ir. C. Simao Ferreira, ir. J.A. Melkert, dr.ir. E. Mooij, ir. J. Sinke, dr.ir. W.J.C. Verhagen

1. THE DESIGN SYNTHESIS EXERCISE

1.1 Introduction

The design synthesis exercise forms a major part of the curriculum at the Faculty of Aerospace Engineering, Delft University of Engineering. The main purpose of the exercise is the synthesis of the curriculum themes presented in the first two years of the educational program at the faculty.

Since this design exercise is organized approximately half-way through the complete five-year program (three year Bachelor of Science in Aerospace Engineering + two year Master of Science in Aerospace Engineering), the design results are not expected to be of a professional quality. Nevertheless the students and their tutors strive to create the best design they can. This is accomplished in an iterative way. Such an iterative process is a typical element of building up design experience.

The way in which a project is carried out and reviewed is only partly focused on the design result. The design process itself is of greater importance. It is especially important for the students to work as a team, since this best reflects a design process in 'real life'. In this way, the students can take full advantage of their personal qualities.

1.2 Objective

The design synthesis exercise helps to meet the faculty's requirement to enlarge the design content of the aerospace engineering course. The goal of the exercise itself is to improve the design skills of the students, in particular project management, communication, team-work and the application of the knowledge gathered in the first three years of the course.

The student has the opportunity to increase his experience in designing. The whole process of designing is dealt with, from the list of requirements up to the presentation of the design. Typical aspects of this process, such as: decision making, optimization and conflicting requirements, will be encountered. Acquiring experience often means going through iterative processes, so design decisions can be altered to make sure that the design requirements are met. The arguments supporting the decisions are reviewed, as well as the way the project is managed. Aspects of design methodology and design management are also taken into account.

During the project the student is expected to work in a team. This means that a student learns to cooperate, to schedule and meet targets, manage the workload, solve conflicts, et cetera. In this field, effective communication is of major importance. Apart from these capabilities the student is expected to be able to communicate ideas and concepts regarding the project subject with specialists and non-specialists. By means of integrated short courses in written reporting and oral presentation, the communicative skills of a student will be developed and assessed.

1.3 Characteristics of the exercise

The characteristics of the design synthesis exercise are:

- For all students, the design component of the study is reinforced by the design synthesis exercise.
- The design synthesis exercise consists of a design project integrated with workshops and courses on oral presentation, sustainable development, systems engineering and project management.
- The exercise has a fixed end date. This means that the third year ends with the design exercise.
- All discipline groups of the faculty provide the support needed during the exercise. This enhances the multi-disciplinary nature of the exercise in general and the design projects in particular.

- The design process is supplemented by lectures on design methodology and project management, as applied to the exercise.
- Aspects of sustainable development, such as noise emission, the use of raw materials, energy consumption and environmental impact are addressed explicitly during the exercise.
- Integrating short courses on oral presentations develops the communicative skills.

1.4 Organization and structure of the exercise

Students indicate their preferences after presentations by the staff introducing all project subjects. Students are divided into groups of approximately ten persons, as much as possible according to their preferences. The exercise takes place during a continuous period of eleven weeks, the last educational term of the third year of the Bachelor course. Technical aspects of the project take up 60 percent of the time; the remaining 40 percent is spent on general topics supporting the project work. General topics are spread over the full period of the exercise. The general topics are sustainable development, design methodology and project management and oral presentations.

1.5 Facilities

To complete the exercise design within the given period of time, the groups of students can make use of several facilities. Each group has its own room, with various facilities (tables, chairs, computers, flip-over charts et cetera). Commonly used software like CATIA, Matlab, MS Office, MS Project, Python, MSC Nastran and more project specific software are available. A special library is available, containing literature on typical project subjects. Finally each group has a budget for printing and copying.

1.6 Course load

The course load is measured in credit points according to the European Credit Transfer System, ECTS: 1 credit point equals 28 hours of work. The total course load is 15 ECTS credits.

1.7 Support and assistance

An essential part of designing is making choices and design decisions. During a technical design process, the choices made in the first stages are often based on qualitative considerations. When details of a design take shape, quantitative analysis becomes increasingly important.

The considerations accompanying these design choices need mentoring and tutoring, since students lack experience in this field. The execution of the project demands a fair amount of independent work of the design team. This means that the team itself is capable of executing the design process. The task of the team of mentors is mainly to observe and give feedback on the progress. The team of mentors consists of a principal project tutor and two additional coaches. Each has a different area of expertise. The method of working, the organization, the communication of the team and the collaboration within the team itself are also judged. Where necessary, the mentors will correct the work and work methods of the team. Warnings of pitfalls and modeling suggestions for certain problems during design will be given when needed, to ensure a satisfactory development of the design.

1.8 Design projects 2015-2016

The Design Synthesis Exercise 2015 is divided into 28 different design assignments. In the table below an overview is given of these subjects. In the following chapters the results of the design teams are covered in detail, as well as the important design characteristics. These are: problem introduction, design specification or list of requirements, conceptual designs, the trade-off to find the “best” design, a detailed design and finally conclusions and recommendations.

Fall DSE

Nr.	Project Title	Principal Tutor
F1	Modular Multipurpose Research UAV (M2RU)	Roeland de Breuker
F2	Airlifting containers	Joris Melkert
F3	Affordable, reusable and sustainable university launch system	Marc Naeije
F4	Aerobatic Light Sport Aircraft Family (LSA)	Roelof Vos

Spring DSE

Nr.	Project Title	Principal Tutor
S1	WiFly: Emergency Wireless Flying Network for Disaster Areas	Ferry Schrijer / Sander van Zuijlen
S2	Delivery System for Stratospheric Aerosol Geoengineering	Steve Hulshoff
S3	Capture a Small Asteroid and Change its Orbit	Ernst Schrama
S4	Piggyback to the Outer Planets	Wouter van der Wal
S5	Wings for Aid	Joris Melkert
S6	Design of a UAV for NavAid Calibration and Testing	Paul Roling
S7	Low-cost Narrow-body Aircraft	Wim Verhagen
S8	AntiDrone-Drone	Erik-Jan van Kampen
S9	SCULPTUR Surveillance Civil UAV Led by Propellor-based Gas Turbine	Matteo Pini
S10	A Purpose Designed Air Safety Investigation UAV	Calvin Rans
S11	Ultra-long Range Business Jet	Jos Sinke
S12	Orrery	Roger Groves
S13	TinySats (Femto-Satellites)	Prem Sundaramoorthy
S14	Finding Venusian Volcanoes	Daphne Stam
S15	Suborbital Satellite to Space	Chris Verhoeven
S16	Lynx Spacecraft: a Deployable UHF Transponder Payload	Trevor Watts
S17	Floating Wind Turbines – the Future of Wind Energy	Axelle Viré
S18	Rediscovering the Magnus Effect in Aircraft	Santiago Garcia Espallargas

1.9 The design exercise symposium

The one-day design exercise symposium forms the conclusion to the design project during which all student teams present their designs. The presentations cover the design process as well as the design result. The symposium is primarily intended for participating students, mentors and tutors. Other staff and students and external experts are invited as well.

A group of experts from within the faculty as well as from industry form the jury and assess the presentations in style and technical content. Three criteria determine the score of the group:

1. technical content (35%)
2. presentation (20%)
3. design content (35%)
4. sustainable development (10%)

The jury of experts this year consisted of:

Fall DSE

Hester Bijl	TU Delft
Egbert Torenbeek	emeritus professor Aircraft Design
Boudewijn Ambrosius	emeritus professor Astrodynamics
Kees Sudmeijer	Dutch Space
Jenny van der Pols	Aircraft Design and Integration ADSE
Marcus Basien	Aircraft Design & Certification Ltd.

Spring DSE

Hester Bijl	TU Delft
Gianfranco Chiocchia	Politecnico di Torino
Pascal Bauer	ENSMA Poitiers
Robert Jan de Boer	HvA
Paolo Astori	Politecnico di Milano
Rob Hamann	SEC-kwadraat
Lukas Roffel	Thales
Jenny van der Pols	ADSE
Gustavo Alonso	UPM Madrid
Daniel García Vallejo	US Sevilla
Jean-Luc Boiffier	ONERA + Supaero
Emmanuel Benard	ISAE Toulouse
Daniel Hanus	CVUT Prague
Vassili Toropov	Queen Mary University London
Luís Braga Campos	IST Lisbon
Robert Hewson	Imperial College
Martin Lemmen	Light Product Development
Ton Maree	TNO

Mark Oort	Airbus Defence and Space
Richard Cooper	Queen's University Belfast
Lex Meijer	Airbus Defence and Space
Ron van Manen	Cleansky

2. MODULAR RESEARCH UAV

Students: J.M. Bakker, S. Potkamp, R.L. Bokdam, S.A. Doesburg, J.M. de Wilde, M. van Rijsingen, J.J.P. van den Berg, M. Baars, P. Nijman, M.J. Perdeck

Project tutor: dr.ir. R. De Breuker

Coaches: Ir. V.S.V. Dhanisetty, Ir. T. Visser

2.1 Introduction

The faculty of aerospace engineering of the TU Delft is the largest and most sophisticated faculty with respect to aeronautical and space engineering in Northern-Europe. This is mainly due to the extensive research facilities available to researchers and students. The high speed and low speed wind tunnels provide the researchers the opportunity to accurately analyse the flow around wings and even complete small aircraft. The research simulator SIMONA is able to test new flight control schemes and can simulate in-flight conditions. Furthermore the research aircraft of the faculty, a Cessna Citation II which combines the capabilities of the wind tunnel and the simulator can be used.

All have one thing in common, the high costs to make use of these facilities. Especially the use of the Cessna is very capital intensive and subject to stringent regulations. A promising solution for solving these issues is the use of a Modular Unmanned Aerial Vehicle (MUAV). The investments and operational costs are much lower, and the regulations are more lenient. It is therefore that the TU Delft has assigned a team of ten students to design a MUAV that can be used as a platform to test new and innovative aeronautical concepts.

Scope

The project objective statement is:

"Develop a low-cost modular platform for in-flight testing of aeronautic concepts with 10 students in 10 weeks."

The main focus during the design phase will be the modularity of the flying platform, as this is the innovative part of the project and of primary interest to the client. Because of the limited available time of 10 weeks and the strong focus on modularity, other parameters of the vehicle will be designed in less detail. Off-the-shelf solutions will be chosen for some subsystems of the platform such as the control systems, software and the engine. Two example modules will be designed alongside the MUAV to demonstrate the capabilities of the aircraft. These two modules are a box wing and a set of seamless high lift devices.

Design requirements and constraints

In this section, the most important requirements and constraints will be discussed. They are divided in requirements concerning modularity, weight and size, flight conditions and propulsion.

With the modular platform it should be possible to test different wings, propellers, High Lift Devices (HLD) and flight control schemes. Therefore the MUAV should have a fixed fuselage and empennage, capable of sufficiently controlling the aircraft around its three axes. The standard wings should be able to accommodate leading edge, trailing edge and tip devices and should be detachable from the fuselage.

The standard wings have a wing span of 6 m, zero leading edge sweep and an aspect ratio of 10. However, the platform should be able to fly with a variety of planforms. An Independent Design Space (IDS) was chosen in which it should be possible to have a stable and controllable aircraft with every possible combination of parameters. In order to achieve this, the platform should have adjustable stability margins. After consulting multiple experts the IDS was defined as:

- Leading edge sweep: between -10 and 15 deg;
- Taper ratio: between 0 and 1;
- Aspect ratio: between 5 and 15;
- Surface area: between 1.5 and 4 m^2.

The Operational Empty Weight (OEW) should not exceed 60 kg and the payload must be at least 20 kg. The OEW contains the basic aircraft with the standard wings and fairings for the leading edge, trailing edge and wing tip. Modules to be tested as well as related data acquisition hardware, sensors and telemetry are counted as payload.

To achieve realistic flight conditions the cruise speed should be at least 55 m/s. The flight altitude is limited by regulations to be below 100 m. Endurance will be no less than 35 minutes including 5 minutes reserve fuel.

Initially a zero emission propulsion system was required. Therefore the possibilities of electric propulsion were investigated. This revealed that these technologies are not yet capable of storing the energy required for the target velocity and endurance within a realistic weight budget. Therefore combustion engines have to be used. To keep sustainability in mind, the choice was made to use bio-fuel.

2.2 Concept generation

After the scope and the main requirements of the project had been determined the conceptual design phase could be started. First a functional analysis was performed, after which design option trees were generated to discover all possibilities for the different functions that should be fulfilled. After a feasibility study and initial trade-offs the most realistic options remained. A trend that could be seen is that fixed interfaces are the best in almost every way, except that they cannot be changed. Making more elements modular adds weight and complexity, but also increases testing possibilities. It seems that this is only worth the trouble when a lot of extra testing possibilities are being created. With the remaining options three concepts have been generated:

- The fixed wing concept;
- The movable wing concept;
- The modular fuselage blocks concepts.

The first concept, called the 'fixed wing concept' is the most conventional one. The longitudinal wing position is fixed. The stability margins can be adjusted by moving payload or adding dead weight. This concept has been used as a benchmark for the other concepts. The planform for this concept can be found in figure 2.1.

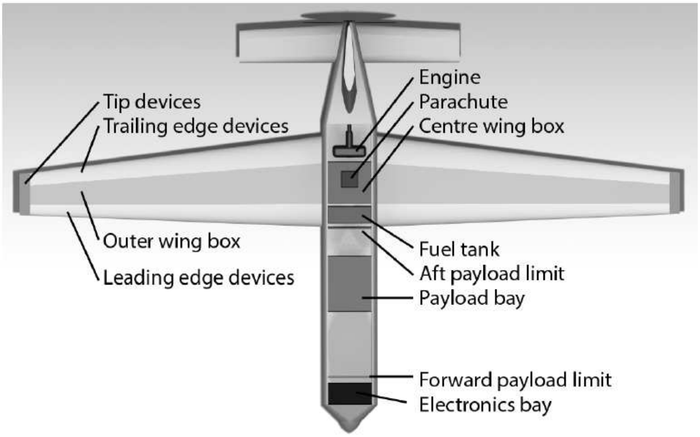

Figure 2.1: "Fixed wing" concept planform

The second concept occupies the middle ground in terms of modularity. The fuselage is fixed, but has multiple mounting points for the wings. Therefore the wing can be moved both forward and backward by half a cord length from its default position. This concept is called the 'movable wing concept'. The planform for this concept looks very similar to the 'fixed wing' concept, except for the longer wing box and shorter fuselage.

The third concept goes all the way in terms of modularity. Its fuselage is composed of 7 blocks which can be arranged in various orders. Therefore this concept is called the 'modular fuselage blocks' concept. An artist impression of this concept can be seen in figure 2.2.

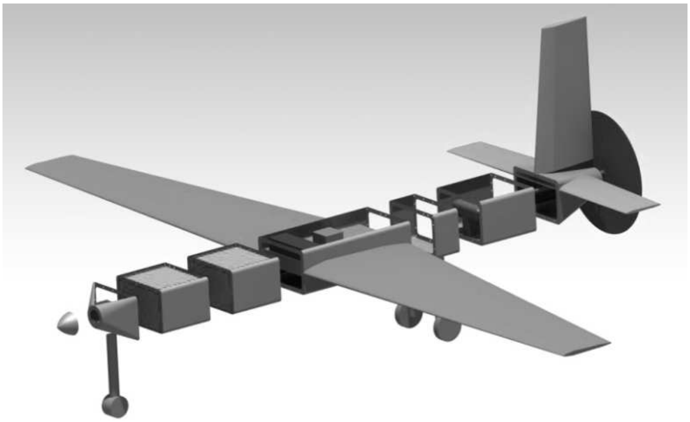

Figure 2.2: Artist impression of the 'modular-fuselage-blocks' concept

2.3 Trade-off

After analysing the three concepts generated above a final trade-off can be made. The criteria used for the trade-off are:

- Independent Design Space coverage;
- Performance;
- Additional configurations;
- Manufacturing cost;
- Operational Empty Weight;
- Technical risk;
- Project risk;
- Development cost;
- Saleability;
- Deployment time.

After close examination of the results, a final decision could be made. Concept 3: Modular fuselage blocks has been eliminated due to severe weight and complexity penalties. The fixed wing concept and the movable wing concept are close and a winner is not obvious. The main differences are a small OEW penalty for the movable wings concept in return for a small increase in testing options and saleability. The question is now, whether the extra modularity and saleability of the movable wing concept weigh up against its weight and cost penalty. This has been put up to a vote within the project team and the decision was unanimous: the movable wing concept is chosen. In this case the tie breaker was that the movable wing concept is more innovative and more interesting to design.

2.4 Final design

After choosing the movable wing concept, this concept was designed in further detail. A preliminary design was made for all subsystems, with an emphasis on the subsystems related to modularity. The final design consists of a modular platform, a default set of wings and two example modules. The default configuration has an OEW of 73.1 kg and can be manufactured for around €43,000.

Modular platform

The fuselage has a length of 4 meters and a rectangular cross-section of 52 cm wide and 31 cm high. The planform is rather conventional, with a tricycle landing gear and conventional tail. The propeller is placed on the back, to have clean airflow over the wings. A side view of modular platform can be seen in figure 2.3.

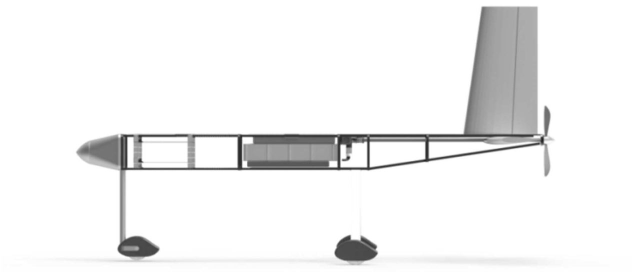

Figure 2.3: Side view of the final design, showing internal layout

In the front of the fuselage the payload bay is mounted. The design of the payload bay is based on a 19 inch rack, a standard often used for electrical equipment. A maximum of 15 kg of payload can be mounted here. Electricity is available at 5, 7.4 and 24 V with a maximum power of 1 kW. To change the stability margins of the MUAV, the position of the payload rack can be moved over a distance of 50 cm. The payload can be accessed by removing the nose cone or opening the hatch on top of the fuselage as can be seen in figure 2.4.

Behind the payload bay is the centre wing box. The centre wing box has a length of 93 cm with 5 attachment points. Therefore, a standard wing can be mounted in 4 different positions or bigger wing boxes can be mounted. All mounting points are designed to carry the loads for all possible wings in the IDS.

The empennage is capable of controlling the MUAV around 3 axes with any of the wings in the IDS attached. An oversized vertical surface and fully movable horizontal surfaces were designed to make this possible.

The MUAV is powered by a 14.5 kW, two-stroke, combustion engine using bio-fuel. With this engine a maximum cruise speed of 59 m/s can be reached. An alternator capable of producing 1.2 kW of electrical energy is attached to the shaft. The power is delivered to a Power Distribution Unit, which transforms it to 5, 7.4 and 24 V. A back-up battery is present to cover power peaks and engine failure.

The fuel system can store 5 litres of fuel, giving a maximum endurance of 2 hours.

The fully programmable control system can both fly using an autopilot and manual controls. The data collected by the standard sensors is saved on an SD-card and sent to the ground station. Extra sensors and actuators can be added to test additional modules.

In case of an in-flight failure, a parachute is on-board. When the parachute is used, the MUAV will land on its landing gear in every configuration, unless one of the wings broke off.

Default wings

The MUAV comes with a default set of wings with a wingspan of 6 m. Modules of up to 10 kg can be attached to the leading edge, trailing edge and wing tips. To make sure producing these modules is as straightforward as possible, the wing box is not tapered, resulting in straight interfaces. When no modules are attached the interfaces are covered by fairings. Figure 2.4 shows an exploded view of the standard wings and the MUAV.

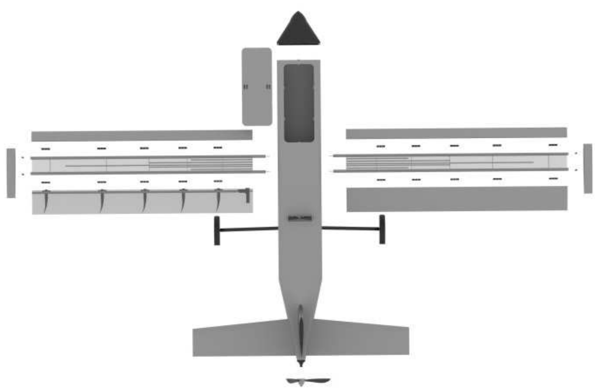

Figure 2.4: Exploded top view of the final design

Example modules

To show potential customers some promising possibilities of the MUAV, two example modules were designed: a box wing and a set of Seamless High Lift Devices (SHLD). The main purpose of a box wing design is the minimization of induced drag. This is because the boxed wing can be seen as an infinite wing, maximizing the effect of a winglet. A render of the modular platform with a boxed wing can be found in figure 2.5.


Figure 2.5: MUAV with the boxed wing attached

The concept of SHLD design revolves around creating a wing-HLD combination without edges or crevices, improving aerodynamic characteristics and reducing profile drag. SHLD also produce less noise than conventional HLD which makes them attractive for airports near residential areas. As can be seen in figure 2.6, the example module uses ‘horn-shaped’ eccentric beams, which can be rotated to deflect the 0.3 mm aluminium skin.

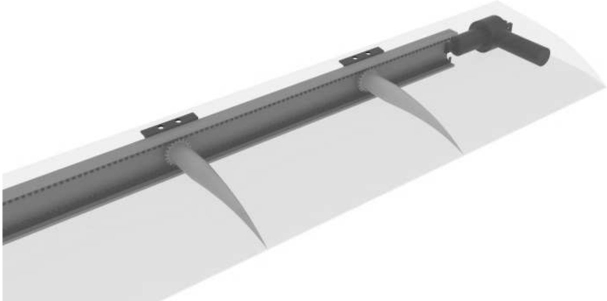
Figure 6: Seamless high lift device example module

2.5 Sustainability

Sustainability has been a driving factor in the design of the MUAV, both for selecting the type of propulsion and the materials. On top of that, the modularity has advantages related to sustainability as well.

As mentioned before, electric propulsion is not yet viable within the specified requirements. Therefore bio-fuel is used. This does not result in a zero emission vehicle, but since the greenhouse gasses emitted were absorbed during the production of the fuel, the vehicle has a zero net emission. The availability of bio-fuel varies widely for different parts of the world. Fortunately, combustion engines can be adapted to run on multiple types of fuel. Therefore the MUAV will be able to run on the fuel most available at the site of the costumer.

In the design of the MUAV, the recyclability is incorporated in the design through the proper selection of highly recyclable materials and simplicity of the modules. This was one of the driving parameters in the choice for aluminium as aluminium is much better recyclable than carbon structures. It is also chosen to use only one kind of aluminium alloy, to allow for easier recycling and a lower loss in material grade.

One of the most important assets of the aircraft considering sustainability is the modular design in itself. Although it requires a little more material to achieve this amount of modularity, it creates many opportunities. One of these is the fact that all components can be upgraded and changed one at a time. This means that a specific End of Life solution can be designed for every separate component of the aircraft. On top of that, only the components that need to be changed will be replaced. Last is that the access and modularity make repair much more easy, this saves energy and material.

On top of that, this research platform can and will be used for the testing of concepts that can make the aircraft industry more efficient, and therefore more sustainable. Research on concepts that in theory can greatly reduce the aerodynamic drag that is encountered during flight, such as the boxed wing concept and SHLD, will likely get a boost. This research platform thus greatly facilitates the testing of sustainable innovations.

2.6 Business case

As the MUAV is a research tool, the potential customers are: universities, research institutes and aircraft (part) manufacturers. Since the MUAV is designed to complement wind tunnel testing, the size of the market was estimated using the amount of wind tunnels as a measure for the size of the aeronautical research market. This resulted in an estimated market size of 3100. To get a return of investment in 5 years, 19 MUAVs should be sold for a price of €90,000, which is a market penetration of only 3%.

The manufacturing costs of new modules for the MUAV are around €3,000, depending on their complexity. Combined with the sales price, this makes the MUAV priced very competitive. For example a test day at a wind tunnel costs around €4,500 and the Cessna Citation has an operational cost of €1,500 an hour, which is even excluding the production of the module to be tested. On top of that the initial investments for the other two test possibilities require millions. Therefore the MUAV can be considered to have plenty of market potential.

2.7 Conclusions and recommendations

The goal of this project was to develop a low-cost modular platform for in-flight testing of aeronautic concepts. After ten weeks, this has resulted in a preliminary design of a Modular Unmanned Aerial Vehicle (MUAV) capable of flying with all possible wing configurations within the IDS:

- Leading edge sweep: between -10 and 15 deg;
- Taper ratio: between 0 and 1;
- Aspect ratio: between 5 and 15;
- Surface area: between 1.5 and 4 m^2.

Besides that the MUAV has:

- A cruise speed of 59 m/s;
- A maximum payload mass of 20 kg;
- An endurance of up to 2 hours;
- A parachute safety system;
- An operational empty weight of 73 kg;
- Manufacturing costs of around €43,000;
- Zero net-emission;
- Equipped with a default wing, optimized for HLD testing.

To illustrate the testing possibilities two example modules have been designed. A box wing and a set of seamless high lift devices have been designed as example modules.

This project shows that the concept of a modular testing platform has great potential. However, there is always room for improvement. An iteration in the preliminary design phase would allow the knowledge and experience gained during the design process to be used to further improve and optimize the design. Only after this the detailed design should be started. The structural calculations should be performed in more detail to find local stress concentrations and to account for dynamic response. A conservative fatigue stress limit has been used throughout the design, stringent quality control on the material consistency would allow for the use of higher fatigue limits, reducing the weight of the aircraft.

Although the design space for the MUAV is large, it could be enlarged through the use of a variable tail length or a modular tail. Computational fluid dynamic studies should be performed to evaluate whether or not a front mounted propeller provides acceptably clean airflow over the wings. Mounting the propeller and engine in the front of the aircraft would reduce the required fuselage length and thus reduce the aircraft weight

Finally it is advised to continue development of the modular UAV, as it is a concept with a lot of potential and of real benefit in the cost reduction of test flights.

3. HELLCAT: LOW COST MILITARY CONTAINER TRANSPORT UAV

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3.1 Introduction

A military base in a foreign country needs many different kinds of supplies. These supplies are transported in standard 20 foot containers. The transportation of these containers is a major undertaking. Especially since military bases are often situated far inland, in hostile territory and are hard to reach by road. Currently the containers are shipped from the Netherlands to the nearest transport hub. From there they are transported by road, using a military cargo convoy. This road transport is not without danger for the convoy and its personnel, as there is a significant chance of getting hit by enemy gunfire or Improvised Explosive Devices (IEDs). Thus the convoy is heavily protected and the road is thoroughly checked. This slows the progress of the convoy down and it can take multiple days to reach its destination. Sometimes these containers are transported by a helicopter. This however is very expensive and the aerial resources from the military can be better used for more urgent missions. Therefore, the Project Objective Statement is:

"Develop an Unmanned Aerial Vehicle to transport a 20-foot container within the requirements and budget set by the Royal Netherlands Air Force (RNLAF)."

3.2 Objectives

To solve this problem, the RNLAF has given this team, through the Aerospace Cluster, the assignment to come up with a solution. Therefore, the Mission Need Statement is:

"Air transport a 20-foot container autonomously and safely over a distance of 250 km (and return without payload), cheaper and faster than by road."

The requirements for the vehicle are:

- Unmanned
- Autonomous, where the ground segment is being taken care of
- VTOL
- Format to be transported: standard 20' container
- Payload weight: 5000 kg (bruto)
- Distance: 250 km one-way with cargo, 250 km back, without cargo
- Unit price of €500,000
- Operational costs of €0.25/kg/100 km
- Development costs must be within the development budget of the RNLAF
- Certifiable according to military certification standards
- Sustainability must be shown in the design (choice of materials), during operations (energy, noise) and in decommissioning (end of life solution)

3.3 Engine selection

It was established early in the project that the engine selection is the most difficult part. Engines certified for aviation are either too expensive, or not powerful enough. Therefore, four concepts are established, based on different engine configurations. After a thorough search, two different engines are selected which are able to suffice to our needs. The first engine is the Honeywell T55. This is a certified turboshaft engine, which can provide a power of 3500 kW. Two of these engines provide the power for the Chinook. This engine has proven itself over the years, is already certified and can deliver sufficient power to complete the mission. The downside of this engine is the price. It costs more than 1 million euro’s which is already twice the budget set by the RNLAF. Though the RNLAF admitted that they thought such an UAV could not be produced for less than 5 million euro’s.

The other option is the General Motors LT4. This is the piston engine that is used for example in the Chevrolet Corvette Z06. It can deliver a power of 485 kW, therefore multiple of these engines are needed to provide sufficient power. The downside of this engine is the weight, the dry weight of this engine is 300 kg per engine, which is the same for the sole turboshaft engine alone. The other downside is that this engine has never been used for aviation before. Therefore, the certification of this engine will be an expensive and time consuming process. The upside is that this engine is relatively cheap, it only costs around €20,000 per engine. Despite at least eight engines are needed, this fits well within the budget set by the RNLAF.

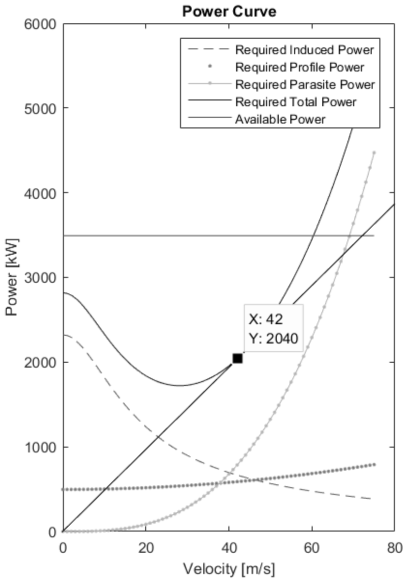

Figure 3.1: Power curve

During this phase in the design process, also electrical engines were considered an option. A helicopter only uses its maximum power during take-off and landing. This can be seen in figure 3.1, when a helicopter gains forward velocity, less power is required. Therefore, during take-off and landing, a lot of excess power would be needed from the piston engines. As a solution to this problem the Tesla model S Ludicrous engine are considered as an Assisted Take Off and Landing system. Electrical engines in general can be boosted to up to 250% of its power for a short amount of time. The advantage is that these engines can deliver a relatively high power, for a relatively low price. The Tesla engines only cost €15,000 (excluding batteries) and can deliver a power of 568 kW each. So the Tesla engines could provide additional power during take-off (with energy from the batteries) and during cruise they can kick-in in emergency situations or provide extra controllability.

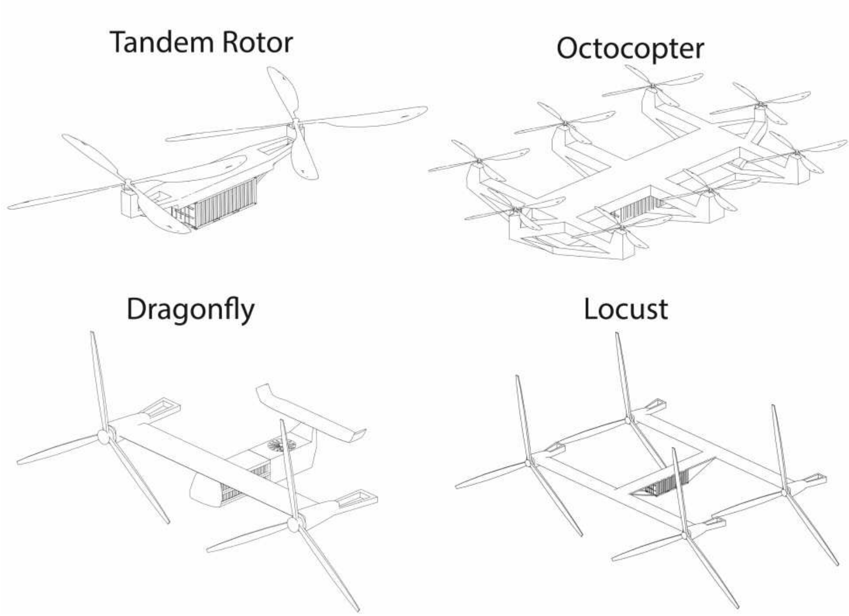

Figure 3.2: Concept sketches

3.4 Conceptual design

After considering all the design options, two main categories remained: The copter concepts and the wing concepts. The copter concepts are based on (multirotor) helicopters and the wing concepts consist of aircraft with tilt-rotors. Each category has a design with the Honeywell T55 and with the GM-LT4. This is done to make a proper trade-off between reliability and unit price. Sketches for the different concepts can be seen in figure 3.2.

The first concept is the ‘Tandem Rotor’, this is a helicopter with two rotors which have a diameter of 17 meters. This design is propelled by the T55 turboshaft engine and is based on the Chinook. The advantages of this concept are the proven reliability and known maintenance schedules, the simplicity of the design and the relatively small size of the (light weight) structure. The downside of this concept is the unit price, which at this point is estimated to be €1,700,000.

The second concept is the 'Octocopter', the octocopter has eight GM-LT4's and two additional Tesla S Ludicrous engines. The eight piston engines are driving eight rotors which each have a diameter of 8 meters. Two of these rotors are 'backed-up' by the electrical engines. The advantage of this concept is the low estimated cost of €570,000, due to the relatively cheap engines. The disadvantage is that these engines are not designed for aviation. Therefore, it is expected that these engines will have a relatively lower reliability.

The third concept is the Dragonfly. The Dragonfly has one large main wing and a smaller tail wing. Two large rotors are driven by a single T55 engine and a smaller rotor inside the structure is driven by a Tesla S Ludicrous engine, in order to provide stability during hover. The main advantage of this concept is the high cruise velocity (twice that of the copter concepts) and the low fuel consumption. The disadvantage is the complex tiltrotor system and the expensive engine. The total costs are, at this point in the design, estimated to be €2,000,000.

The fourth and last concept is the Locust. This concept has two large wings with four large tilt-rotors, which are driven by eight GM-LT4 engines. Two Tesla S Ludicrous engines will support two rotors to improve the controllability and provide extra power when necessary. This concept combines the cheap engines with the high cruise velocity. The total costs are, at this point in the design, estimated to be €1,000,000.

To determine which is the best concept, a trade-off is set up. The main criteria of this trade-off are the technical performance, the RAMS, sustainability, costs and the feasibility. RAMS stands for Reliability, Availability, Maintainability and Safety. The main parameters are given a certain weight, depending on their importance. Each of these main parameters consists of multiple sub criteria which are also given a weight relative to its main parameter. Each concept is given a grade from 0-3 for each sub criterion, where 3 is the highest score. From this, a final grade is computed for each concept. The concept with the highest score will be further designed in the next phase of this project after consultation with the RNLAF. The trade-off can be seen in table 3.1.

Table 3.1: Trade-off matrix

Trade-off criterion	%	Octocopter	Locust	Tandem Rotor	Dragonfly
Technical Performance	18	1: High Rate of Climb	2: High cruise speed	1: Poor ferry range	3: Cruise speed and ferry range
RAMS	22	1: Redundancy	1: Hybrid engines and tilt-rotors	2: Proven concept	3: Proven engine
Sustainability	7	1: Batteries	2: Low fuel consumption	2: Low fuel consumption	2: Low fuel consumption
Cost	39	3: Almost within budget	2: Over budget	0: Far above budget	0: Far above budget
Feasibility	14	2: Large amount of parts	0: Complex system	3: Proven design	1: Complex system
Result	%	29	26	23	22

From table 3.1, it can be seen that the Octocopter concept was the best design according to the trade-off matrix. The main reasons for this are the low price and the fact that the low reliability of the engines is compensated with redundancy. When one engine fails, the Octocopter will still be able to complete the mission.

3.5 Detailed design

The final design is called the Heavy Lifting Low Cost Autonomous Transporter, HELLCAT. From the beginning of the detailed design, the electrical engines are discarded. This is because this system is relatively heavy (especially the batteries) and it only causes 'dead weight' during cruise. And because of the lighter structure, the extra power is not required, not even when one engine fails. For the detailed design the team is divided over four subgroups, the aerodynamics, propulsion, control and structure group. Though there are a lot of interactions and iterations between these groups during this design phase, the results of the design are explained per subgroup.

Structure

The purpose of designing a structure is to make it as light-weight as possible. For the HELLCAT, this is achieved by the effective 'tic-tac-toe' layout. This configuration created the possibility to save manufacturing cost and overall weight. The material, Aluminium 7075 T6510/1, is chosen specifically to fit this design, in order to reduce weight and improve the structural integrity. Using cables and locks to attach the container implies an attachment system that is safe, easy to operate and light-weight. The landing gear of 4.6 meters long supports this by providing enough height such that the container does not touch the ground while landing. Also this makes easy and safe ground operations possible.

Aerodynamics

The structural simplicity of the design implies an aerodynamic challenge. By using lightweight fairing on the drag sensitive beams and engines, the drag is reduced significantly. Using a cylindrical landing gear with zip-zap strips, which trips the boundary layer, reduces the drag even further. A design with two rotor blades, that consists of two aerofoils each, is chosen. The first aerofoil is the Boeing Vertol VR-12 for the root half and the second is the VR-14 for the tip half. This results in an optimal lift production, while keeping the structure as simple as possible. Only a collective pitch system is used for this design. This collective pitch of the rotor can be adjusted by a simple swash plate on the rotor hub. This causes the rotor hub to have a simple and therefore cheap design.

Propulsion

The propulsion group is responsible for calculating the overall performance of the HELLCAT. All the performance characteristics are shown in table 3.2.

Table 3.2: Technical Specifications

Performance	Value	Unit
Cruise speed	150	km/h
Mission range	500	km
Ferry range	700	km
OEW	7,500	kg
Fuel weight	2,000	kg
Payload weight	5,000	kg
MTOW	14,500	kg
Outer dimensions	27.8 x 27.8 x 6.5	m

The propulsion group also designed the engine in more detail. All the necessary components are added, such as fuel systems, gearboxes and Engine Control Units. For safety reasons a performance kit is added to the engines. This performance kit can, due to electronic optimisations, temporarily boost the power output of the engines to 545 kW. This is only used in emergency situations.

Stability, control and operations

After the initial checks and when the flight plan is uploaded, a container is connected to the HELLCAT. From this point on, no further personnel is needed until the delivery of the payload. This autonomy is gained using a custom designed autopilot with active collision avoidance. Through this autopilot the HELLCAT is controlled such that it is able to withstand engine failure and gusts exceeding 20 m/s. This controllability is gained using differential rotor pitch which creates the roll, pitch and yaw moments, as well as the vertical thrust. Lateral control is achieved by adjusting the attitude of the vehicle. To simulate the stability of the HELLCAT, Simulink is used. This Simulink controller results in a stable flight system which can rotate around all axes, perform stable step and ramp responses (without overly exciting the engines) and perform all manoeuvres that could be asked by the navigation and position controller. The autopilot combined with the controllability and advanced positioning systems autonomously achieve an accurate landing. After the rotors stop spinning, the container can be detached and the HELLCAT can prepare itself for the return flight.

3.6 Sustainability

During the design constant attention is paid to minimise the engine power by reducing the drag. The HELLCAT produces 43% less CO_2 than the Chinook for the same mission. The exhaust system with a catalyst further reduces the NO_x concentrations of the exhaust gases by 70%, compared to the Chinook. When an engine is no longer safe enough to be used in the HELLCAT, it still has value in the car engine aftermarket. The main structure is made primarily of aluminium which is easily recyclable and the fairings are made out of biodegradable foam.

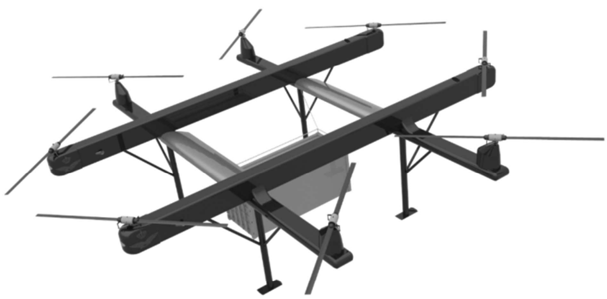

Figure 3.3: The HELLCAT

3.7 Conclusion

With the HELLCAT, future military logistic operations are a problem of the past. This new product can transport large amounts of containers through the air, cheaper and faster than by road. But more important: it saves lives! It can be produced for 1.3 million euro's. Although the unit price exceeds the budget set by the RNLAF, they estimated that such an UAV could not be produced for less than 5 million euro's. To keep the costs down the design is made as simple as possible, with a lot of commercially available off-the-shelf components. The operational costs are €0.20 per kg per 100 km. This fits well within the budget of the RNLAF. When the unit price is evaluated per kg payload times km range, the HELLCAT is at least a factor 4 cheaper than its closest competitors.

3.8 Recommendations

The next step should be testing the GM-LT4 engines under circumstances that simulate a typical mission. There is no reliable data available to determine the reliability of these engines under the difficult (environmental) conditions that the HELLCAT will face. Also the certification procedure should be investigated more thoroughly, as this will be the most critical part for this design. Especially since the HELLCAT is the first of its kind. The stability and control should also be researched in more detail. An autopilot should be developed, since during this project only a simulation of the stability and control is made. These detailed studies and developments should ultimately lead to a safe and reliable design.

4. Quantum Launch System: Affordable and Sustainable Access to Space

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Project tutor: Ir. M.C. Naeije

Coaches: Ir. J.F.G. Schneider, Ir. J. Carvajal Godinez

4.1 Introduction

There is trend in the miniaturisation of satellites. Clusters of small satellites in Low Earth Orbit (LEO) are used more and more often to conduct scientific research or demonstrate novel technologies. These satellites are mostly nanosatellites, defined as having a mass between 1 and 10 kg. However, there is no dedicated launcher available for these missions. The only option for these satellites is to piggy-back on another mission. However, this offers very limited flexibility and can cost as much as $50,000 per kg of payload. The Delft University of Technology has asked for an affordable and sustainable way to launch these nanosatellites that provides more flexibility than piggy-backing. The most important top-level constraints are a payload mass of 60 kg, a first launch in 2021 and a competitive price-per-kg compared to piggy-backing.

Quantum Launch Systems rises to this need by aiming for the market gap with an orbit at 350 km altitude and with flexible launching schedules. There are several challenges that need to be addressed to reach this goal: the cost of developing a launcher, the recurring production cost and operation cost, as well as the lower technological readiness of sustainable solutions in space industry. The mission statement reads:

"The Quantum Launch System will put combined payloads of up to 60 kg into a Low Earth Orbit of 350 km for less than $50,000 per kg from 2021 onwards."

4.2 Concepts and trade-off

In a first stage of the project, different concepts were generated that could potentially perform the mission. A preliminary trade-off was then done on these concept designs. Three designs came out of this trade-off: a jet fighter air-launched rocket, a partially reusable rocket and a railgun. These concepts are shown in figure 4.1. These concepts were designed in some more detail, to find the advantages and disadvantages of each concept. To find out which concept meets best the requirements, a trade-off was done. From this trade-off it was found that a railgun was not feasible due to the requirement of first launch in 2021. The expected loads of 750 g during launch were a major design problem, which could not be met. The air launch was not chosen since the rocket would be too big to fit under a jet fighter aircraft. Next to that, the safety of the pilot of the aircraft was a big issue. So the chosen concept is a partially reusable rocket, the Quantum Launch System.

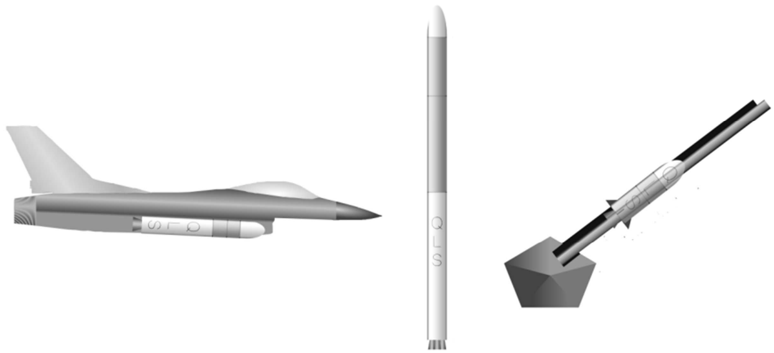

Figure 4.1 The three different concepts present in the final trade-off

4.3 Detailed design

After the trade-off, the chosen design could be developed into more detail. This was done in an iterative process, of which the culmination was the Trajectory Simulation Tool (TST) developed by the team. This tool allowed for an input of all design parameters of the iteration so far, and would conclude with not only a simple ‘yes’ or ‘no’, but also with detailed information and graphs of the flight profile, including losses and margins. This section contains a short description of the different subsystems that resulted from these iterations, and an explanation of the most important design choices that were made in the process.

Layout

The rocket consists of three stages, of which the first will be recovered. Because of this, both the first and third stage will contain a flight computer, flight termination system, communication array, power management system and control system. The general mass and size layout of the launcher is shown in table 4.1.

Table 4.1 General launcher size parameters

Lift-off mass	9191	[kg]
Stages	3	[-]
Total propellant mass	7499	[kg]
Total length	18	[m]
Diameter	1	[m]

Propulsion

The propulsion system uses liquid oxygen and liquid methane (LOX/LCH_4) as propellants. This combination has gained a lot of interest in recent years, but is not used yet on existing launchers. Its performance is comparable to other commonly used propellants, but has one huge advantage in terms of reusability: its combustion process is very clean, which means it leaves no residues in the engine. This is beneficial for reusability, since it significantly simplifies refurbishment. Secondly, since liquid fuels are used, in theory the first stage can be landed, filled up and launched again.

The first two stages are driven by a turbo pump. These are normally highly complex and costly devices. Therefore a choice was made to design pumps driven by an electric motor and batteries, instead of the gas generator cycle. This might increase the mass of the pump, but keeps the complexity and thus cost at a minimum.

The propellant tanks are made of aluminium-lithium alloy, which was chosen for its high strength and low density. The chamber and nozzles will be made out of 3D-printed steel, to keep production cost at a minimum. A regenerative cooling cycle will be used, where the liquid methane cools the combustion chamber and nozzle before it is burned as fuel. It was found during the design process that the mass of the upper stage nozzles would be high if they are completely made out of steel. Therefore, after an expansion ratio of 40 the steel is replaced by an ablatively cooled carbon fibre reinforced nozzle extension skirt. This increases the expansion ratio and thus specific impulse while minimising the mass.

Payload integration

To make the Quantum Launch System a flexible launcher, it should be able to launch different payload combinations and configurations. An universal payload adapter is part of the upper stage, on which all the payloads can be mounted in the same way. This eases payload integration. The possible payload configurations are: a package of fourteen 3U-CubeSats, a microsatellite with a maximum mass of 60 kg, or a combination of a smaller microsatellite and less cubesats.

Furthermore, if the miniaturisation of satellites continuous a system can be developed that carries large amounts of picosatellites (mass between 100 g and 1 kg) at the same time.

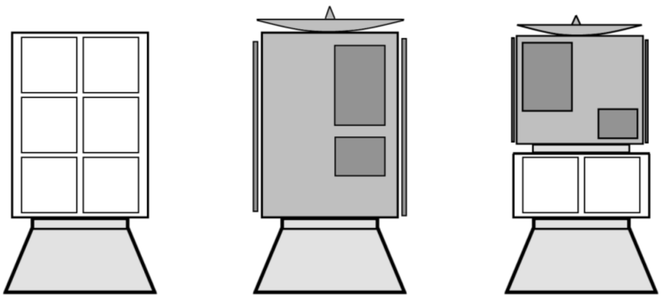

Figure 4.2. Three different payload configurations: (a) multiple nanosatellites; (b) single microsatellite; (c) combination of both

Recovery

To save costs, the first stage will be recovered. This is done by a boost back burn at an altitude of 54 km, followed by a parachute descent and vertical landing using specially-designed landing legs. After the boost back manoeuvre, the engines will turn off. When the first stage has reached an altitude of 12 km, a drogue parachute will be deployed to decrease the velocity of the first stage from its supersonic terminal velocity to a subsonic velocity of 77 m/s. At an altitude of 7.6 km the main parachute is deployed, which reduce the velocity to 5 m/s. At an altitude of 100 m the landing legs are deployed using three actuators. The main parachute is disconnected from the rocket to prevent sudden winds from inflating the chute and tipping over the rocket during the landing. The landing legs will prevent the nozzles from hitting the ground. Impact velocity will be absorbed by the shock absorber inside the three landing legs, while the outer structure of the landing legs and supporting legs will carry structural loads.

ADCS

To control and stabilize the rocket during launch, it will make use of Thrust Vector Control. This means that the nozzles can be rotated to provide control moments. This is a form of active stabilization: the rocket is not passively stable. The first stage contains one gimballed engine in the centre plus three hinged nozzles in a triangle pattern, so pitch, yaw and roll control is possible. The second and third stage only have one gimballed nozzle each. This will only allow yaw and pitch control; therefore, the third stage will also contain cold gas thrusters for roll control. These cold gas thrusters are also placed on the first stage, for the flipping turn manoeuvre for the recovery of the first stage.

4.4 Cost analysis

As rockets have a tendency to scale poorly when making them smaller, cost reduction strategies have been used at multiple points to reach the launch price target. Firstly, the engines for all the stages will be made using additive manufacturing ('3D-printing'), which can reduce the production cost of rocket engines by as much as 75%. Next to that, only one facility will be used to build and test all the subsystems, so productivity is kept as high as possible. Furthermore, the different stages will be made according to the same design principles, only the length will differ. Since they are made in the same way, the same tooling and manufacturing methods can be used for the entire rocket, significantly simplifying the production process. These strategies, combined with the reusability of the first stage, results in an average cost per launch of $2.57 million, or $42,850 per kg of payload.

4.5 Sustainability strategy

Sustainability has been one of the main design criteria during the entire design process of the reusable launcher. The focus on sustainability culminates in the recovery subsystem of the first stage. The sustainability strategy focused on three main points: minimizing waste, reusing the launcher, and using non-toxic and low emission propellants. Firstly, the use of additive manufacturing wherever possible is proven to reduce production waste by as much as 70% compared to traditional machining. The choice of using parachutes to decelerate the stage before landing was done to decrease the use of the propellants. This will not only reduce the amount of propellant required on board of the rocket, which would increase the mass of the rocket, but it also provides a sustainable and environmentally friendly solution because no emissions are expelled in the atmosphere and the main parachute itself is reusable as well. The first stage will land close to the launch pad, which will decrease transportation costs and emissions. During landing of the first stage the surrounding environment will not be harmed or polluted since the propellant system is not used. The retractable landing legs can be reused multiple times for future launches. Most of the parachutes can be reused for future launches. Finally, space debris will not be an issue for the QLS because the target orbit is at an altitude of 350 km, which is a 'self-cleaning' orbit. This means that the lifetime of the spacecraft will be four to five months on average, thus not causing space debris or aggravating the Kessler Syndrome.

4.6 Conclusion

The system can launch many different payload configurations, making it flexible and addressing a larger market. The reusability of the first stage will not only enable the reuse of materials and subsystems but also significantly reduces the cost. QLS also features an innovative propellant combination and will make use of novel manufacturing technologies. A render of the final system is shown in figure 4.3. To conclude, the QLS provides a cost-effective and sustainable solution to launch small payloads into LEO.

Figure 4.3. The final layout of the Quantum Launch System

5. SALSA: Family of Aerobatic Light Sport Aircraft

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5.1 Background

The Federal Aviation Authority issued the LSA class in July 2004. The significantly lower certification costs involved in certifying S-LSA (Special LSA), compared to a type certified aircraft, mean that more pilots can afford to fly. However, only few of the available LSA are capable of aerobatic flight. An aerobatic LSA, certified according to the ASTM F2245-15 standards, provides a means for pilots with a sport pilot license to compete in intermediate category aerobatic competitions.

The number of certified aerobatic light sport aircraft on the market is relatively small. Therefore, the American Institute of Aeronautics and Astronautics (AIAA) issued a Request for Proposal to design an aerobatic LSA family, consisting of a one-seat and a two-seat aircraft

variant. The proposed family of LSA should target this market gap, and offer competitive aerobatic performance at an affordable price.
To achieve a low price, emphasis should be put on the commonality between the two aircraft family members. The aircraft should display at least 75% part commonality in weight.

5.2 Project objective and design requirements

The objective is to propose a design to the AIAA request for proposal. The product required to fulfil this goal is nicely described by the mission need statement.

Mission need statement:

"A competitive and sustainable, two-member family of aerobatic light-sport aircraft for the U.S. market, with entry into service in 2020."

The design phase is started with identification of all requirements. A large number of requirements originate from the Request of Proposal, FAA certification standards. Additional requirements followed from the market analysis.

The following requirements are the primary requirements determining the design of the SALSA family.

- The aircraft shall have level 1 handling qualities.
- The aircraft shall be able to operate in controlled and uncontrolled airspace.
- The aircraft shall accommodate 95% of human pilot sizes.
- The aircraft shall provide space for 30 / 30 pounds and 4 / 6 cubic feet of baggage for ferry missions.
- The single-seater shall be competitive in the IAC intermediate category.
- The gross take-off weight shall be at most 1320 lbs.
- The stall speed shall be at most 45 knots CAS in clean configuration.

- The climb rate of the one-seat variant shall be at least 1500 fpm at sea level, at International Standard Atmosphere (ISA) + 10°C
- The negative limit load of the one-seat variant shall be at most -5G / -3G (for the two-seater), with 230 lbs / two 200 lbs pilot(s), 15 lbs / two 15 lbs parachute(s) and 1.5 hours / 1.5 hours of fuel.
- The positive limit load of the one-seat variant shall be at least 6G / 6G, with 230 lbs / two 200 lbs pilot(s), 15 lbs / two 15 lbs parachute(s) and 1.5 hours / 1.5 hours of fuel.
- The roll rate of the one-seat variant shall be at least 180 degrees per second at the maximum cruise speed or 120 knots CAS, whichever of the two is lowest.

From these requirements it can be seen that the single-seater and the double-seater SALSA should be designed to perform well in respectively aerobatics competitions and training. To be competitive in the IAC intermediate category requires the aircraft to perform all aerobatic figures of that competition. Additionally, visibility from the cockpit is important to perform well in aerobatics. Crisp stall behaviour is also appreciated during the series of manoeuvres that make up the competition. Finally, sharp edges on the aircraft make it more visible to the judges. This enables them to assess the performance of the pilot more accurately and can potentially result in a higher score. These things have to be taken into account in the design of the family and should reflect themselves in the final concept.

5.3 Market analysis

First, a market analysis is performed to investigate the market opportunity of the SALSA family. In 2013 the special light-sport aircraft (SLSA) category in the USA consisted of 2,056 registered aircraft. The predicted average annual market growth is 4.1% per year from 2013 to 2034. At the entry-into-service (EIS) of the first SALSA variant, in 2020, the volume of the market is expected to be 3,080 SLSAs. The number of registered SLSAs is predicted to be 4,880 in 2034, an increase of 1,800 aircraft, which are assumed to all be resulting from new aircraft sales. This is deemed to be a realistic assumption considering the immaturity of the LSA category and its aircraft (established in 2004), compared to the average age of general aviation (GA) aircraft of 40.7 years.

To increase potential sales, foreign markets are assessed. Potential markets are suitable if they meet the following requirements: their respective countries must have (or plan to have) a regulatory framework similar to that of the USA and they must be large enough to justify the additional certification costs. The following markets are selected for future expansion: Australia, Brazil, Canada, China, Europe, New Zealand and South Africa. Including these markets, more than 1200 aircraft can be sold over a period of 15 years.

To be competitive in the market of aircraft that is capable of competing in the IAC intermediate category, the SALSA single-seater should cost less than $ 150,000. Additionally, it should be able to handle higher loads than specified in the request of proposal. The required loads of the SALSA family should be at least +8G/-6G for the single-seater and +6G/-5G for the double-seater.

5.4 Concept design and trade-off

The design space is bounded by multiple requirements, such as: using an unpressurized cabin, one engine, fixed landing gear, max. two seats, propeller-driven, and a fixed-pitch propeller. Five main configuration options appeared to determine the aircraft layout mostly are:

- Wing position
- Engine position
- Tail type
- Seating arrangement
- Landing gear arrangement

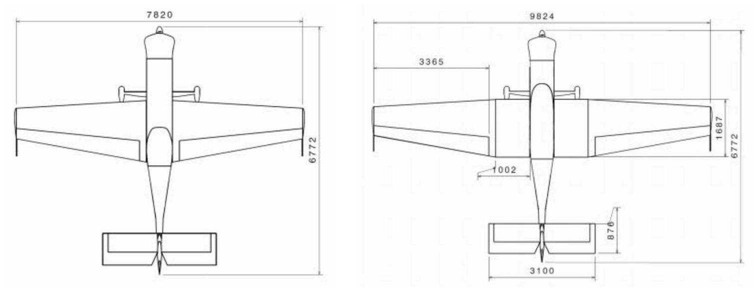
A monoplane and a biplane configuration were investigated in more detail. A trade-off was performed between these configuration options based on performance, pilot visibility, useful load, risk of concept, versatility and cost. With no clear performance difference between the two concepts, the deciding factor in the trade-off was visibility from the cockpit. Visibility from the aircraft is of utmost importance when flying an aerobatics competition.

5.5 Continued conceptual design

To reduce the production costs to a minimum, the commonality in parts within the SALSA family was emphasized. The difference between the one-seater and two-seater is most apparent in the wings. The wing features a NACA 0012 aerofoil with relatively high maximum lift coefficient and sharp stall characteristics. A symmetric aerofoil is required for similar aircraft performance when flying inverted during aerobatics. However, the wings have to differ in size because of the difference in maximum take-off weight between the aircraft versions. The two-seat aircraft has a larger maximum take-off weight and as such requires more wing surface area to comply with the stall speed requirement of 45 kts. The extra wing area is added through root extensions. This is favoured over a tip extension, since it results in a wing with a lower structural weight. Adding an extending element, instead of producing a different wing, retains the family's commonality in parts.

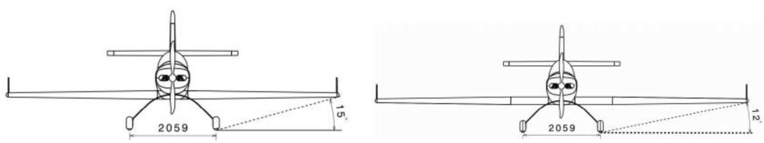
Aircraft overview

The figure below shows the three views of the SALSA family. The aircraft have an identical fuselage, with the only difference visible in the canopy of the aircraft. Furthermore, the aircraft make use of the same empennage. This results in a very high part commonality, which saves significantly in production costs. What makes the SALSA family special are the detachable wings and horizontal tail. This enables SALSA owners to save on storage costs and offers them more flexibility in operation.



(a) Top view of the one-seat variant (b) Top view of the two-seat variant

Top views of SALSA family members



(a) Front view of the one-seat variant (b) Front view of the two-seat variant

Front views of SALSA family members

Figure 5.1: SALSA family top and front view

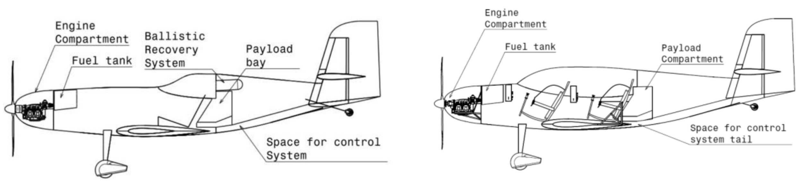


Figure 5.2: Side views of the SALSA family

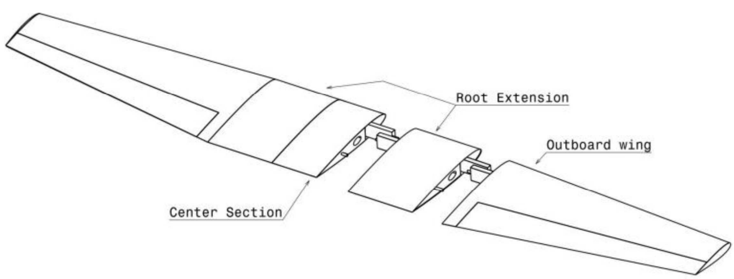


Figure 5.3: Detachable and extendable wing of the SALSA aircraft

Structural design

The load carrying structure in the wing consists of the glass fibre and foam composite skin, a carbon fibre composite main spar at 25% chord to carry most of the bending loads, and a second spar at 62% to introduce aerodynamic loads of the aileron into the structure and carry shear and torsional loads. The spar material was selected to be CFRP for the spar caps and foam-core CFRP for the web. The high specific strength and modulus of the CFRP is desired as spars carry the majority of the loads in the wings.

The end of each spar is tapered, to allow the corresponding spar end of the following wing section to easily slide alongside each other and into the adjacent wing section. A metal insert is laminated into the end of each spar section, to transfer the loads of the spar cap to the shear pin at the end of the spar stubs. This transfers the load to a bushing insert on the adjoined wing section. The figure on the next page shows the construction.

The semi-monocoque fuselage is constructed out of glass fibre and foam composite sandwich. It is reinforced with bulkheads and frames, joined with longerons. Extra reinforcement is required at the points where engine, wing, empennage and aircraft parachute loads enter the fuselage structure.

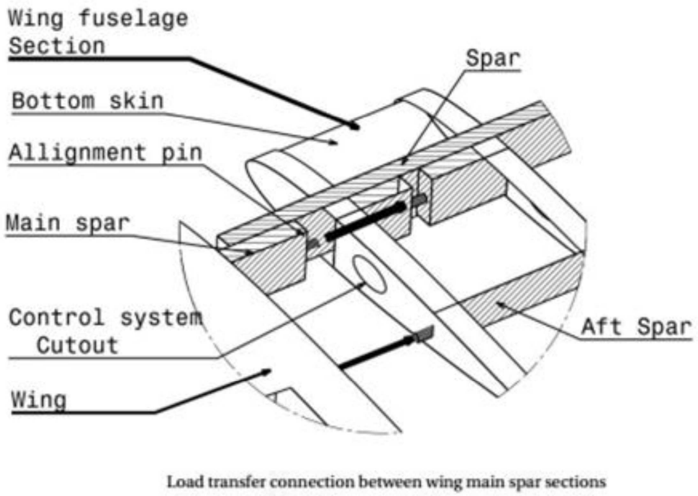


Figure 5.4: Wing structure

The bending moments and shear loads of the empennage surfaces are carried by a two spars. The rear spar also introduces the loads of the control surfaces. The horizontal tail plane is detachable for the owner's convenience, as depicted in the figure below. This enables storage and transport on a trailer if desired. Average maximum trailer width across the United States is 102 inches, so without a detachable horizontal tail, this would not be possible. The complete empennage is constructed out of glass fibre composite.

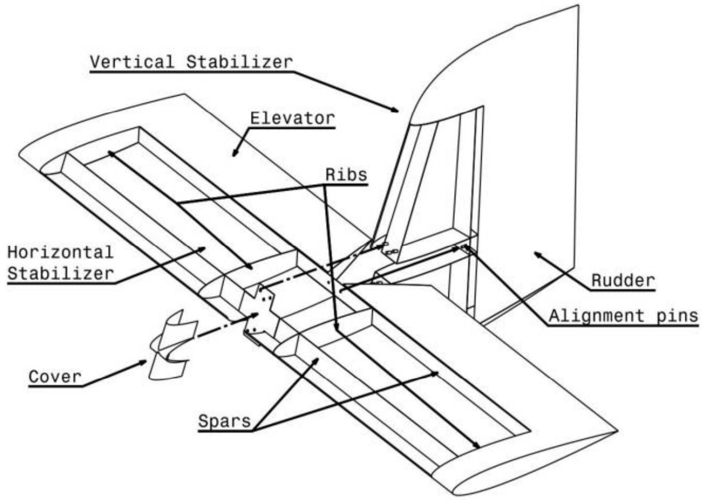


Figure 5.5: Empennage structure

Propulsion

The requirements state that the engine should meet the LSA ASTM standards, which implies that the engine has to be certified. Secondly, both aircraft and thus their engines have to be capable of sustaining inverted flight for at least 5 minutes. Finally, constraints for the design point have to be implemented, especially the minimum power required of 75 hp / 66 hp and the engine has to be available before the entry into service in 2020.

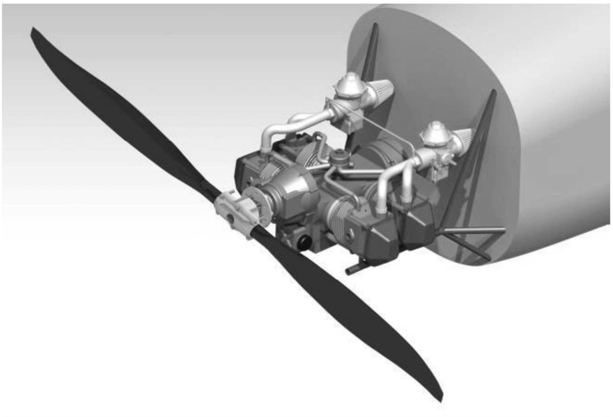


Figure 5.6: engine and propeller installation

The Rotax 915 iSc was selected as the most suitable engine for the SALSA family. It produces 135 hp of power and runs on Mogas. This makes it more attractive than competing engines, because the Rotax engine has lower emission and is cheaper to operate. It is combined with a 2 Blade Rotax Ground Adjustable Propeller from Sensenich.

Stability and control

The empennage, wing position, landing gear position and pilot positions were iterated to come up with two stable and controllable aircraft designs. Extra care had to be taken to not diminish the part commonality within the SALSA family. The aircraft were determined to comply with all level 1 handling requirements, as well with the minimum stick force requirements from the ASTM standard.

Subsystems

The fuel and oil systems are constructed with a flop tube to ensure fuel and oil flow, even when flying inverted for extended periods of time.

Avionics of the SALSA family follow the trend towards a "Technically Enhanced Cockpit" or glass-cockpit, instead of the standard mechanical gauges. Generally, glass cockpit screens are more detailed and accurate, when compared to mechanical dials. As LSA certified aircraft are not required to have any certified avionics. This significantly lowers the cost. When flying in controlled airspace, a certified radio and transponder have to be added. An impression of the glass cockpit is given in the figure below.

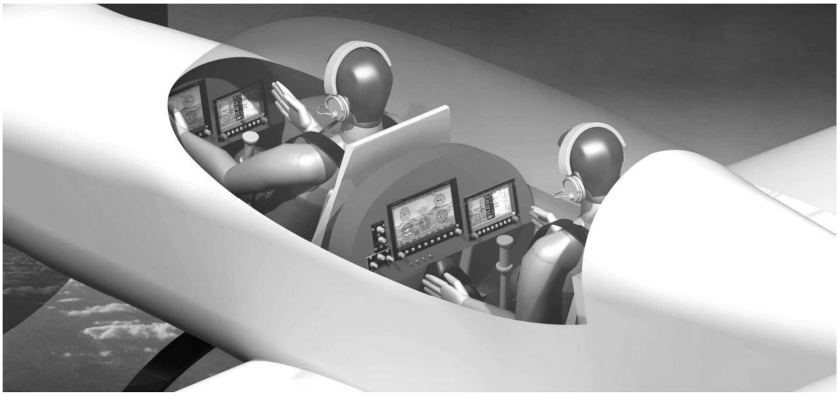


Figure 5.7: Technically Enhanced Cockpit

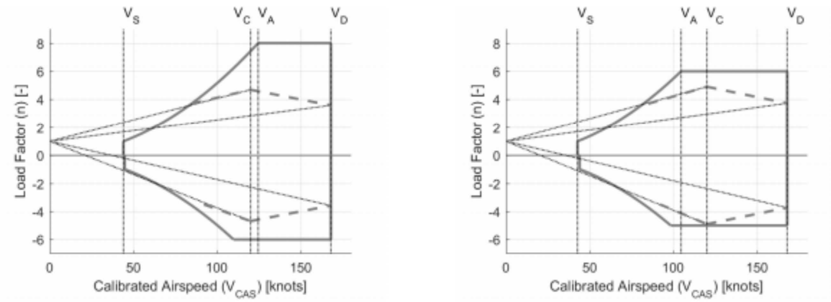
Performance analysis

LSA landing and take-off performance requirements are easily met by the SALSA family. SALSA reaches take-off and landing distances of 220 meter and 280 meter for the single-seater and 335 meter and 325 meter for the two-seater. Moreover, the ferry cruise requirements were met at MTOW for both aircraft. This means both aircraft can cruise substantially further than required, with ferry ranges of 500 nm and 450 nm for the single-seater and two-seater respectively. These ferry ranges give the SALSA family a stronger position on the current LSA market.

Climb performance can be of importance during timed aerobatic sequences. The climb rate of the SALSA family is determined to be 11 m/s for the single-seater and 8.4 m/s for the two-seater. This performance is comparable to competitors on the market and adequate for aerobatic flight.

V-n diagrams have been constructed according to the ASTM standards for LSA. It appears that the gust loads are no driving factors for either design. Furthermore, the aircraft cannot reach its ultimate loads at cruise speed (VC), which has a positive effect on safety.

However, the limit load of the one-seater (8 G) is a reachable target while flying at cruise speed.



(a) V-n diagram of the one-seater.

(b) V-n diagram of the two-seater.

V-n diagrams of the SALSA family.

Figure 5.8: V-n diagrams for the SALSA family

Realization

During the design of the production plant of the SALSA family, extra care was taken to reduce costs. The total non-recurring Research, Development, Testing and Evaluation cost for the project is $2,8 Million USD. For a total production volume of 1200 aircraft, the result is an average of $2,400 USD of development costs per aircraft.

Operating costs of the SALSA family should be low. Assuming 240 flight hours per year for a flight school, the direct operating costs are calculated to be $35 USD/FH for the one-seater and $45 USD/FH for the two-seater. These are very reasonable operating costs for LSA. Indirect Operating Cost consists largely of airport servicing. It should be noted that storage/hanger costs are considerably lower due to the detachable wings of the aircraft.

5.6 Conclusion

To conclude the SALSA family met all requirements that were imposed upon its design. The design can be seen as feasible and it should be able to perform very competitively in the IAC intermediate category.

The design approach included minimizing manufacturing and development cost by maximizing the re-use of major airframe components for both the one-seat and two-seat variants. This was achieved by keeping identical fuselage, empennage, landing gear, fuel system. The wings of the one-seat variant are used as outer wing section of the two-seat variant with root extensions. The proposed SALSA family achieves 80% commonality by weight. Additionally, in designing the structural components of the aircraft it was attempted to produce the aircraft using as many straight components as possible. Their simplified production will lower production costs even further.

The manoeuvrability of the aircraft is another major design objective. The designed one-seat variant is capable of performing all manoeuvres in the IAC Intermediate category. The roll rate that can be achieved with SALSA is slightly above the roll rate that is generally found in aircraft competing in the IAC intermediate category. Extensive analysis of the stability and control characteristics show that all level 1 handling qualities are satisfied. The flight controls meet ASTM standards. The aircraft can be trimmed in any flight condition and the spin stability conditions are met as well.

The performance of the aircraft is an important design driving factor. The wing features sharp stall characteristics which are favourable for aerobatics.

The wings stall completely within a small angle-of-attack range and reattaches readily. The take-off and landing requirements in the most critical condition, at MTOW, are satisfied. For performing aerobatic manoeuvres, the climbing performance of the SALSA has paramount importance and both aircraft meet the minimum requirements.

A design philosophy of minimizing costs hand-in-hand with maximizing performance and manoeuvrability makes SALSA at $135,000 for one-seat and $145,000 for two-seat variant feasible and competitive in the market.


Figure 5.9: Artist impression of the SALSA family

6. WiFly – An Emergency Communication Network for Disaster Areas

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6.1 Introduction

During the past decades, mobile communication has become more and more important in everyday life. A lot of people are dependent on mobile communication, feeling the urge to check emails, send texts by other means than the Internet or simply be able to communicate instantly. This communication is dependent on a proper and reliable infrastructure.

If the infrastructure is damaged, service may be impaired. Current climate development has already shown that extreme meteorological situations are occurring more often and researchers expect this development to continue. Extreme weather situations like storms, tsunamis and floods, as well as other natural disasters like earthquakes can cause great damage, up to the complete failure of the communication infrastructure. If this happens, human beings will be thrown in a situation which they are not accustomed to. They will be in a disaster area, feeling the urge to get information and to communicate even more, but without being able to do so.

The feature of receiving fast information for rescue parties and victims increases the likelihood of survival. This calls for the creation of an emergency communication network. The WiFly system is an ambitious project developed by 10 students at TU Delft in 2016. The mission objective of WiFly is:

"to provide a reliable emergency communication network in the disaster area by means of a swarm of UAVs."

This summary will first highlight the requirements and constraints of the WiFly system. Then the potential design concepts are described and the trade-off criteria for finding the most optimal design are offered. The chosen concept will then be described in detail and finally a conclusion and recommendations for the WiFly system are given.

6.2 Design requirements and constraints

It was important to first establish the design requirements for the WiFly system in order to have a well-defined starting point for the technical design and set the borders of the design space. The functional requirements were that the system:

- needs to be set up within 24 hours and it shall operate for at least 24 hours at the disaster site.
- shall reach the disaster site within 1.5 hours if it is located closer than 300 km away from the storage location of the WiFly system and within 5 hours if it is located between 300 and 1,500 km away from the storage location.
- is required to withstand winds of 10 Bft (102 km/h).
- shall cover an area of at least 100 km^2.

- shall provide Internet access for all cell phones with a data rate of at least 9kbps.
- shall allow for communication between the rescue teams and people in distress.

All these requirements had to be met within certain constraints which were mainly dominated by cost.

A prototype UAV shall not cost more than 50k€ while the whole system shall not cost more than 3M€. Furthermore, the annual maintenance cost is constrained to 100k€.

6.3 Concept study and related trade-off

One of the primary steps in the design process was defining the UAV concepts which can perform the mission under the imposed requirements and constraints. Five different classes of UAVs were defined as potential candidates for the WiFly system. Afterwards, a trade-off matrix was defined, including twelve different criteria used for grading the design concepts. In table 6.1, an overview of the whole trade-off procedure is shown. The concept which scored the best is highlighted in the table.

Table 6.1: The trade-off table between potential concepts. Scores from 1 to 5 (lowest to highest)

Criterion	Weight	Rockets	Balloons	Airships	Rotorcraft	Fixed Wing
Performance	25	1.6	4.2	4.2	2	3.8
Operation	20	3.6	2.2	2.8	3.4	4
Aerodynamics	10	4	2	3	3	5
Structures	10	2.7	4	3	2.7	3
Maintenance	7.5	4	4	4	4	4
Stability & Control	5	1	2.5	3	3	4
Sustainability	5	1	4	4	2.4	3.2
Propulsion	5	5	1	2	3	4
Costs	5	1	4	4	3	3
Modular design	2.5	1	4	4	3	3
Certification	2.5	1	3	3	2	4

Technological maturity	2.5	2	3	3	2	4
Weighted Average	100	2.6	3.2	3.4	2.8	3.9

Due to their high dependence on the mission capabilities, the performance and operation were considered the most important factors in the grading process. The rest of the properties are specific to the subsystems of each concept and the grading was based on literature and previously used systems. At the moment of the trade-off the cost was just a gross estimate obtained from statistical data only for the sake of comparison between the concepts. The grading for certification criterion was based on the number and the complexity of the regulations with which the concept must comply. Technological maturity is a property of the design that offers an idea of how popular the concept is based on past experience. The fixed wing won the trade-off and it was used further in the design process.

With the concept fixed, another trade-off had to be implemented for comparing different fixed wing designs. Based on literature, three concepts were chosen to enter the selection process. These were the conventional wing-tail configuration, the flying wing and the canard, see figure 6.1.

Figure 6.1: the three main UAV design concepts. From left to right: standard horizontal tail configuration, flying wing and canard configuration.

A new trade-off between the three final concepts was implemented. Seven main criteria were used in the grading process, each one with its specific weight. The most important factors in the trade-off process were the performance and the efficiency of the system. These were calculated based on the aerodynamic analysis of each concept together with the anticipated fuel consumption obtained from literature. The operational empty weight for each configuration came from the class

II weight estimation. The complete trade-off matrix is offered in table 6.2. The concept which scored the best is highlighted in the table.

Table 6.2: The trade-off table between fixed-wing concepts. Scores from 1 to 5 (lowest to highest)

Criterion	Weight	Wing-tail	Flying wing	Canard
Performance	20	4	4	3.5
Efficiency	20	3.4	4.6	2.4
Stability & Control	15	4	1	3
Costs	15	3.5	4	3
Reliability	15	5	1	2.5
Structural weight	10	4.5	3.5	3
Sustainability	5	4	4	4
Weighted Average	100	4.01	3.17	2.96

The result of the trade-off concluded that the wing-tail configuration was the most suitable for the designed system, scoring more than 25% higher than the other candidates. With the concept fixed, the design process moved further to its final stage.

6.4 Final concept design

An example of what a typical mission would look like is hereby given. The mission would start with choosing whether to move the base and all the equipment closer to the disaster area, or leave from the current site. A 24 hour endurance is only guaranteed within an operational radius of 300 km: anything further than that will reduce the endurance.

The UAVs are then assembled. This operation should require around five minutes with a team of three people. After having been assembled and refuelled, each UAV is launched using a catapult system, which requires 3 minutes with a team of two. Three catapults are available, so the UAVs will be launched in batches of three, at a rate of a batch every five to eight minutes.

Each group of three UAVs will climb to 4 km altitude, at which point they will assume a "V formation" (to minimize drag) and begin cruising at a speed of 200 km/h (without wind). Direct contact with the base can only be ensured up to 200 km distance, after which point it will be necessary to place a UAV between the swarm and the base to act as a relay. One UAV will be required for every 20 km above the 200 km limit.

Once they have arrived at the disaster area, the UAVs descend to altitudes between 2 and 3 km. When contact is first established with the target area, an initial situation assessment will be carried out. The UAVs will spread out evenly, so that the swarm will cover the whole area with its emergency network.

At this point, the system estimates how many users are located in the sector of each UAV. After a user distribution map is created, the swarm will rearrange itself in order to provide optimal coverage. The UAVs can distribute themselves in different layers, between 2 and 3 km altitude. This allows one to tailor network coverage to the density of each different sector (for low density areas, the UAVs should fly higher, covering a larger area, while for high density areas they should do the opposite).

The UAVs will then loiter for maximum 24 hours, performing their mission, constantly rearranging the swarm configuration in order to provide optimal coverage. When an aircraft needs to return to base, it can either be swapped with a spare one, or an "underloaded" UAV can take over the connections of the leaving one. Thanks to the swarm functionality, the system has "virtually infinite" endurance.

Once the mission has been performed, the UAVs will climb to 4 km, cruise back to base, and land on the skyhook system. Three skyhooks will be available, making it possible to recover three UAVs every three minutes using three people (starting from the moment of impact against the skyhook). The high automation of the system will allow to use a small amount of controllers for the flight operations, whose job will be restricted mostly to supervision, as well as making "swarm level" decisions, and only relinquishing manual control in case of emergencies.

The description of the design is split into all the different subsystems, which will be illustrated in the following order: Communications, Aerodynamics, Structural design, Propulsion and power, Stability and control.

Communications

The communication system allows for communications among UAVs within the swarm, between the swarm and the base, and between UAVs and users on the ground. Concerning the latter, the WiFly system was designed to create an emergency communication network for users in disaster areas. This was achieved by means of a payload communication system. This system provides five different services to users in disaster areas. All of them are aimed at either providing emergency information to the users, allowing them to communicate with the external world, or aiding the rescue effort.

SMS-Broadcast allows to send SMS messages containing crucial safety information to all users on the ground. A Web Portal gives access to specific Internet content (disaster-related), which has been uploaded and cached onto the UAVs. The WiFly system also provides voice calls for rescue teams on the ground, allowing them to communicate with each other anywhere within the disaster area, using a simple mobile phone. Capped SMS messaging is also provided, meaning that all connected users can send a limited amount of SMS to any phone number.

Finally, a situational awareness map is also generated for the rescue forces. This is a map of the disaster area which illustrates the mobile phone distribution in the sector, hence giving an overview of how people are distributed in the area. These services will be provided by a swarm of 45 UAVs to around 300,000 people distributed over an area of 100 km^2. To support the required data-rate and channel capacity of the emergency network, 14 transceivers are to be integrated in the payload communication system. These transceivers communicate with the ground by means of a directional antenna.

The non-payload communication system handles all communications among UAVs, and between the UAVs and the base. Each UAV carries four 2.4 GHz radios, four amplifiers, four omnidirectional antennas (two medium gain and two low gain) and one piece of mesh node hardware (namely the MeshDynamics MD4000). The latter is responsible for managing all communication links, making sure that no UAV is ever left isolated from the swarm, and that the swarm is always in contact with the base.

The UAVs can communicate with the base up to 200 km away, and with each other up to 20 km away. If the base is further than 200 km, a chain of relay UAVs allows for base communications, at the cost of one UAV per every 20 km of extra range. The system has been designed to provide a reliability of 99.9%, keeping into account all possible worst case scenarios (heavy rains, high banking angle, etc.). Hence any communication link will be active at least 99.9% of the time.

Aerodynamics

Concerning the Aerodynamics of the UAVs, the aerofoil choice revolved around ensuring very high endurance capabilities. This resulted in the LA203A shape being selected. The 3D shape of the wing was also designed in order to maximize this performance parameter, yielding a high aspect ratio trapezoidal wing, which was also designed to be aerodynamically efficient, given the Oswald factor value of 0.85. The total wing area is 2.13 m^2, with a root chord of 0.54 m, a tip chord of 0.195 m and a wing span of 6.52 m, for an aspect ratio of 20, which ensures excellent loiter performance.

For the control surfaces, a V tail was selected, since it outperformed all other tail configurations in most of the trade-off factors. The V tail configuration performed particularly well in wing wake handling, drag interference, stability and compatibility with take-off and landing methods (namely catapult and skyhook). Moreover, since the

V tail configuration is used on a very large number of UAVs, it can be considered a "tried and true" design. The aerodynamic model of the whole UAV includes the influences of wing, fuselage and tail, and yields a maximum C_L of 1.85 and a C_{D0} of 0.026, which in turn results in a maximum endurance factor of 21.76 and a maximum range factor of 23.13.

Structural design

The structural design revolved around fuselage and wing design. These were designed with three critical load cases in mind: take-off, flying through a gust and landing. The objective was that upon these loadings neither buckling nor permanent material deformation occurs. Due to the relatively small loads acting on the fuselage and the constraint on the minimum allowed material thickness, magnesium was chosen as the main material for the fuselage for its lightness. To allow for undisturbed propagation of electromagnetic waves, the regions surrounding the communication antennas were made out of the radio-transparent glass fibre. The wing can store all the fuel needed for the mission and can withstand collision with a foreign object. As higher aerodynamic loads are experienced by the wing, this will be made out of aluminium 6061-T6.

Propulsion and power

Concerning the propulsion and power subsystem, the WiFly system has to cruise at a low subsonic speed and loiter for 24 hours. Given the stringent endurance requirement, it was important to choose a propulsion method that was as efficient as possible. Propeller engines offer the best fuel consumption for traveling at low subsonic speeds for very long periods of time, while also resulting in lower manufacturing and maintenance costs. The Rotron RT300 EFI LCR piston engine was selected to power the propulsion system. It has 32 hp and an output power of 20.3 kW.

Its small dimensions allow for easy fitting within the fuselage. The engine makes use of conventional aviation gasoline, which is widely available. For the propeller, it was chosen to adopt a pusher configuration, allowing to move the propulsion system in the back of the fuselage, leaving the nose of the UAV for the communication system. A pusher propeller results in a series of advantages, such as lower aerodynamic interference with the airframe and delayed flow separation. The lower drag is however counteracted by the lower net trust produced, due to the flow being disturbed by the wing.

All things considered, the pusher configuration provides the same efficiency as a tractor one, with the advantage of leaving the nose of the aircraft free for the communication system. A 99 cm (in diameter) three bladed variable pitch feathered propeller was implemented in the design. The variable pitch allows for efficient operation at all flight speeds and engine settings, while the feathering considerably improves the gliding distance in case of a one engine inoperative condition. For a typical mission involving 6 hours of cruise at 200 km/h and 24 hours of loitering at 130 km/h, 43.6 litres of fuel are required, or 31.5 kg. It is also possible to fit an additional 10.0 kg of fuel if necessary.

Stability and control

Concerning stability and control, ailerons provide roll control, while "ruddervators" provide pitch and yaw control. Ruddervators are control surfaces placed at the trailing edge of both V tail surfaces, and they are a combination of elevator and rudder. The aileron size was picked in order to allow a 60 degree roll in four seconds. It was found that an aileron of 0.054 m^2 was needed, with a mean chord of 0.065 m and a length of 0.82 m, positioned 1.5 m away from the wing root (with respect to the aileron's centre). By means of a scissors plot, it was found that the tail surface should be 0.22 of the wing surface, resulting in a tail size of 0.329 m^2 per fin. Both fins are oriented at a 45 degree angle. Each ruddervator was given an area of 0.164 m^2 and a span of 0.64 m, while the tail fin has a span of 0.91 m. Finally, the ruddervator chord measures 0.26 m, while the fin chord is 0.36 m long. Cost, mass and power budgets for each UAV and for the whole system are illustrated in the tables below.

Table 6.3: power budget of each UAV

Payload	398 W
Communications	303 W
Avionics	40 W
Actuators	48 W
Total	789 W

Table 6.4: mass budget of each UAV

Propulsion system	20.4 kg
Airframe	37.0 kg
Avionics and Actuators	4.0 kg
Communication system	3.9 kg
Payload	18.8 kg
All else	6.4 kg
Total	90.5 kg

Table 6.5: cost breakdown per UAV

Propulsion system	8.0 k€
Airframe	5.0 k€
Avionics and Actuators	10.2 k€
Communication system	2.8 k€
Payload	38.1 k€
Total	64.1 k€

Table 6.6: Cost breakdown of the whole WiFly system

Launch Mechanism	150.0 k€
Landing Mechanism	150.0 k€
UAVs	2.88 M€
Ground station	200.0 k€
Total	3.38 M€

6.5 Conclusion

The WiFly system meets most of the requirements established at the beginning of the project. By equipping the WiFly UAV with a 32HP rotary engine and an optimized propeller, it is able to cruise 600 km at 200 km/h and loiter for at least 24 hours at the disaster site. Thus, it fulfils the requirement to reach the disaster site within 1.5 hours if the WiFly systems location of storage is not further away than 300 km from the disaster site.

The requirement to reach the disaster site within 5 hours can be met by designing the whole system for easy assembly and disassembly. This way it can be stored in containers and transported by an airplane in order to reach far destinations on time.

The aim to provide an emergency communication network with the WiFly system is implemented by using a swarm of 45 UAVs, each with an maximum take-off weight of 132 kg. Furthermore, a smart control algorithm is implemented which makes the swarm operation autonomous and adjustable to a non-uniform distribution of people in the disaster area. The aforementioned number of UAVs can provide network coverage to an area of at least 100 km^2, supporting up to 300 thousand users. People in distress will be able to send SMS and access a web portal for emergency information. Rescue crews will be able to carry out voice communication with each other, as well as having access to situation assessment maps, and send SMS broadcasts to all users in the disaster area.

There are two challenging requirements which were not entirely met in the WiFly system. The first one is the requirement to cruise with a 10 Bft wind. It has been relaxed, since headwind increased the speed of the UAV in the cruise phase and the airplane would become too heavy for a feasible design. Hence the UAV can now cruise at lower speeds in such wind conditions, instead of having to cruise at full speed. The second requirement is the permanent data rate of 9 kbps for all users of the system of the ground. The reason for this being not feasible is that it would require too large an amount of data to be transferred between the swarm and the base. In theory it is possible, but an excessive number of UAVs would be required.

The constraints for the WiFly system were unfortunately not met. The system cost is 3.38M€ which means that the budged is exceeded by 0.38M€. One of the reasons for this is that one UAV costs 64.1k€, which means that the price per UAV is exceeded by 14.1k€.

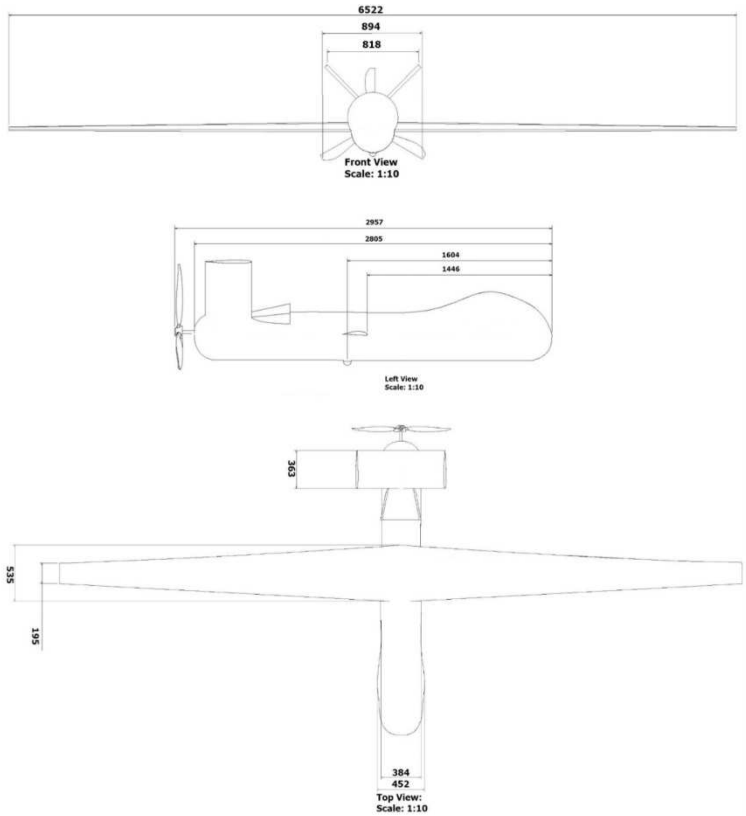

Figure 6.2: A 3 view drawing of the UAV design, showing relevant dimensions

6.6 Recommendations

The main area in which the WiFly system needs some improvement is the communication system, which could not fully meet some of the main mission requirements. This can for example be achieved by designing custom made transceivers which would decrease the system cost by a huge amount. Furthermore, the design can be analysed much more in detail in comparison to the design stage at the moment. Therefore, one should consider a full CFD analysis, which gives more detailed results about the aerodynamic forces. This enables in turn a more accurate structural analysis, which shall be performed by means of a FEM analysis.

7. STRATOSPHERIC AEROSOL GEOENGINEERING AIRCRAFT

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7.1 Introduction

Global warming is advancing faster than models from just a decade ago predicted. Ongoing emissions of carbon dioxide and other greenhouse gases have caused a significant rise in the average Earth's temperature over the past century and might lead to positive feedback effects which further exacerbate the consequences of global warming. Recent efforts to counteract climate change focus on solving the problem in the long term, by reducing the emissions of and capturing greenhouse gases. However, the possibility of unacceptable temperature increase before long term solutions become effective necessitates the development of a system capable of managing global temperatures as a means of temporary risk control, whilst long term solutions are being implemented.

An active way of adjusting the climate, should it be necessary, is presented by geoengineering. Geoengineering is an overarching term for the intentional manipulation of the environment, particularly manipulation intended to reduce the undesired effects of anthropogenic climate change. A geoengineering method to counteract the global temperature increase is solar radiation management (SRM). SRM counteracts warming of the Earth's surface by reducing incident solar radiation. Several possibilities to reduce solar radiation exist, including e.g. space mirror placement, marine cloud brightening and stratospheric aerosol injection. This summary provides a possible implementation strategy, detailed analysis and design of a stratospheric aerosol injection platform using aircraft, the Stratospheric Aerosol Geoengineering Aircraft (SAGA) program.

7.2 Mission objective

The aim of the SAGA project is to propose an aircraft-based method for the delivery of aerosol material to the stratosphere, in order to establish an Earth-covering stratospheric cloud of particles that scatters part of the incoming sunlight. The project goal is stated as follows:

"To perform the preliminary design of an aircraft suited for delivering aerosols to altitudes above 20 km, as well as estimations of the initial and operating cost of a fleet of such aircraft capable of delivering 5 million metric tons of aerosols per year."

The context in which geoengineering is justified needs to be considered carefully. In figure 6.1 three possible global temperature scenarios are shown. The dashed line shows a scenario in which mitigation strategies are not effective fast enough. In this case geoengineering can prove to be beneficial to keep temperatures below the - labelled as dangerous - 2 K increase with respect to pre-industrial levels. When temporary geoengineering is implemented in combination with mitigation strategies, geoengineering helps to keep the temperature below the threshold until greenhouse gas concentration is sufficiently reduced. When geoengineering is implemented, and no mitigation is applied temperatures will rise very rapidly upon halting geoengineering – the moral hazard if geoengineering is wrongfully seen as a solution.

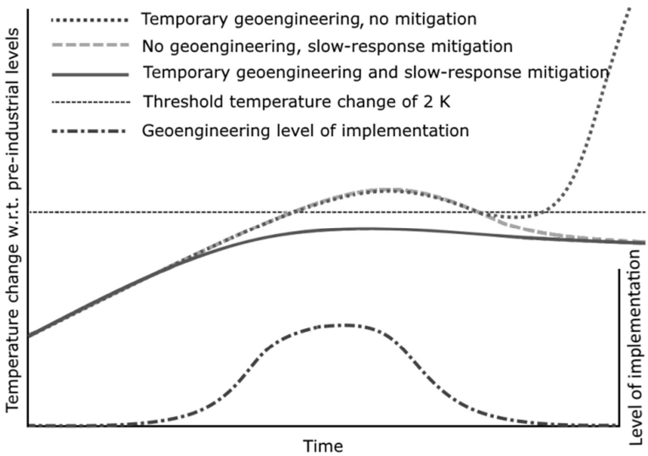

Figure 6.1: Possible temperature scenarios with and without implementation of geoengineering

7.3 Design drivers

The design of the mission is governed by many requirements, of which the most driving ones are discussed in this section. Firstly, the mission requirements and their main implications are addressed, followed by a short description of the commercial aspects of the SAGA program, leading to cost requirements. Finally, the requirements concerning sustainability of the mission goal and the mission implementation are discussed.

Aerosol delivery to the stratosphere is an unconventional and highly demanding mission. The most driving mission requirements concern the payload and the altitude. Annually, 5 million tons of aerosol must be delivered to the stratosphere, which results in high payload requirements. Additionally, the requirement to deliver this aerosol to stratospheric altitudes, meaning altitudes above 18 km, poses stringent constraints on the design.

The payload itself consists of the aerosol material - sulfuric acid (H_2SO_4) -, which must be dispersed in gas form. This acidic compound governs requirements on safety as well as operations, as corrosion resistance must be guaranteed and sufficient thermal insulation and heating systems must be present. Concerning safety and reliability, the unmanned control concept necessitates high levels of reliability and redundancy.

Geoengineering has global impact and is therefore considered to be a global responsibility. The magnitude of the SAGA program calls for an international consortium, likely consisting of governments and institutions with most influence worldwide. In addition, the global responsibility SAGA embodies renders it a program with no intention for commercial profits. Therefore, no clear market is identifiable. The importance of the environment and the absence of commercial interest imply a large budget could be available to implement SAGA. An initial development and acquisition budget of $57 billion and a yearly operating budget of $11 billion are assumed to be available.

SAGA is a mission with the goal to contribute to sustainability and help solve the problem of climate change. As a consequence, every aspect of the mission shall be compliant with this sustainable goal. The effects of the injection of aerosol into the atmosphere must be acceptable in terms of environmental impact. Furthermore, the environmental impact of the delivery system in terms of emissions shall be minimal to ensure the positive effects of aerosol injection are not offset by the negative effects of the delivery of the aerosol.

7.4 Operational concept

Aircraft operations

The operational concept is established with the main goal to spread the aerosol material as efficiently as possible. Dispersing the aerosol in the tropical region is most effective, resulting in its spreading towards the poles by stratospheric motion. The aerosol will be dispersed in the tropical region between 30° North latitude and 30° South latitude at an altitude range of 18.5 to 19.5 km. For aerosol lifetime, stratospheric altitudes between 18 and 25 km are optimal, however, the performance characteristics of the aircraft, discussed below, limit the feasible altitude to the range given. The trajectory of the aircraft is characterized by departure and arrival at the same airport, thus flying out, turning around and returning to base. This operational concept was selected as it provides the possibility to completely cover a circle with a radius equal to half the range of the aircraft. Circles with a radius of 3,500 km allow coverage of the entire tropical region with a relatively small amount of 7 airports, which represents the most economical solution.

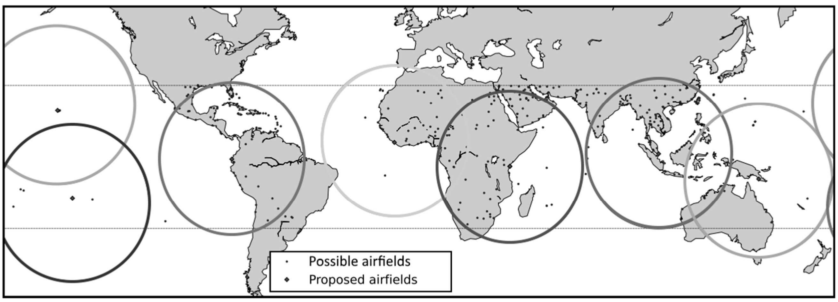

Figure 7.2: Airports and operational radius used by SAGA

The mission profile is constrained by the performance requirements of the aircraft. Due to the weight loss over the course of the mission, high altitude flight is most efficient at the end of the mission when the weight is lowest. For performance reasons it is therefore chosen to perform the outgoing cruise leg to the end of the circle at a low altitude of 13 km, after which a climb to 18.5 km is performed and the aerosol dispersion starts. The return leg consists of three segments during which aerosol is dispersed, at 18.5, 19 and 19.5 km altitude, respectively. The mission profile and the 7 airports with circles showing coverage are presented in figures 7.2 and 7.3, respectively.

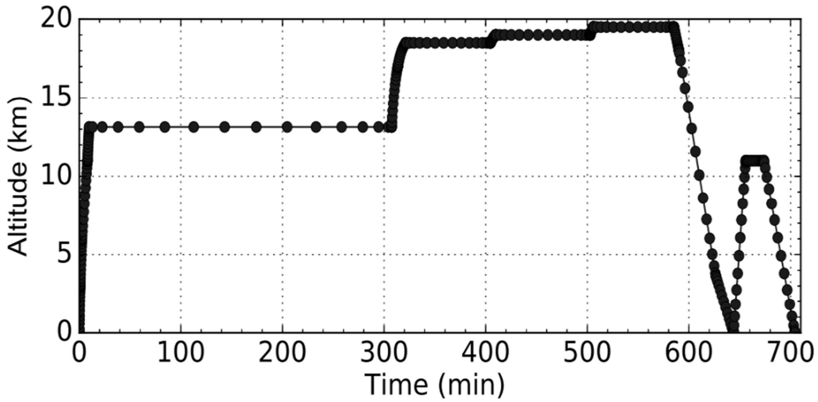

Figure 7.3: Mission profile of the SAGA mission

The 5 million tons of aerosol to be delivered annually, govern the operations in terms of aircraft payload, range and fleet size. For missions performed at this altitude, a payload of 40 tons is cost effective, however, performance characteristics make a payload of 35 tons most fuel efficient. The aircraft range consists of twice the circle radius of 3,500 km plus a diversion cruise, resulting in 8,000 km. The outgoing cruise speed is 191 m/s and the dispersion cruise speed at higher altitude is 210 m/s. The range and these speeds result in a mission time of approximately 11 hours, which implies two missions per aircraft per day can be performed on a constant schedule. In combination with a fleet availability of 80% to account for maintenance and emergencies, these operational parameters result in 572 flights per day and a fleet of 344 aircraft.

Aerosol operations

The aerosol material chosen is sulfuric acid. This compound is selected based on effectiveness, technological readiness level, environmental impact and cost. The main advantage of sulfuric acid is the presence of a natural analogue - volcanic activity. Volcanic eruptions are known to reduce global temperatures due to the ejected sulfuric compounds and in addition, their possible detrimental effects on the environment are known. Due to this natural analogue, the reliability and safety level of sulfuric compounds as aerosol material are high. The specific compound sulfuric acid is chosen as this compound, when dispersed in gas phase, forms aerosol particles of most effective size. The requirement to disperse the aerosol in gas form introduces the need for an evaporation system to evaporate the aerosol - transported in liquid phase - just before injection into the stratosphere. A small amount of aerosol will therefore constantly kept boiling in an evaporation tank. Addition of energy and aerosol influx into the evaporation tank ensure an aerosol dispersion rate of 2.1 kg/s.

7.5 Design

Three concepts were defined and analysed to determine their eligibility to perform the SAGA mission: a conventional aircraft concept, a blended wing body (BWB) and a twin-body aircraft (TBA). The conventional configuration is a proven design allowing for reduced risks and uncertainties and enabling accurate, detailed examination. BWBs have good aerodynamic characteristics due to their smooth shape. TBAs have good structural characteristics and enable a less concentrated load at the fuselage locations, reducing bending moment.

The concepts were graded using four criteria: applicability to SAGA (high altitude performance, aerosol storage capability), sustainability (noise), cost (operational cost, development cost) and risk. The applicability to the SAGA mission is determined to be of largest importance for the selection of the concept, as a concept that does not perform well in terms of applicability will hinder development and fast implementation.

The final result of the trade-off is the decision to proceed with the conventional concept, primarily because of its design maturity and the high accuracy level achievable in the next steps of the process. As a common conventional concept cannot provide a high enough aspect ratio for the SAGA mission, the option of using a strut-braced wing design is included in further development steps.

The final design is completely shaped by the high altitude and high payload requirements. An overview is given in table 7.1, and a three-view is presented in figure 7.4.

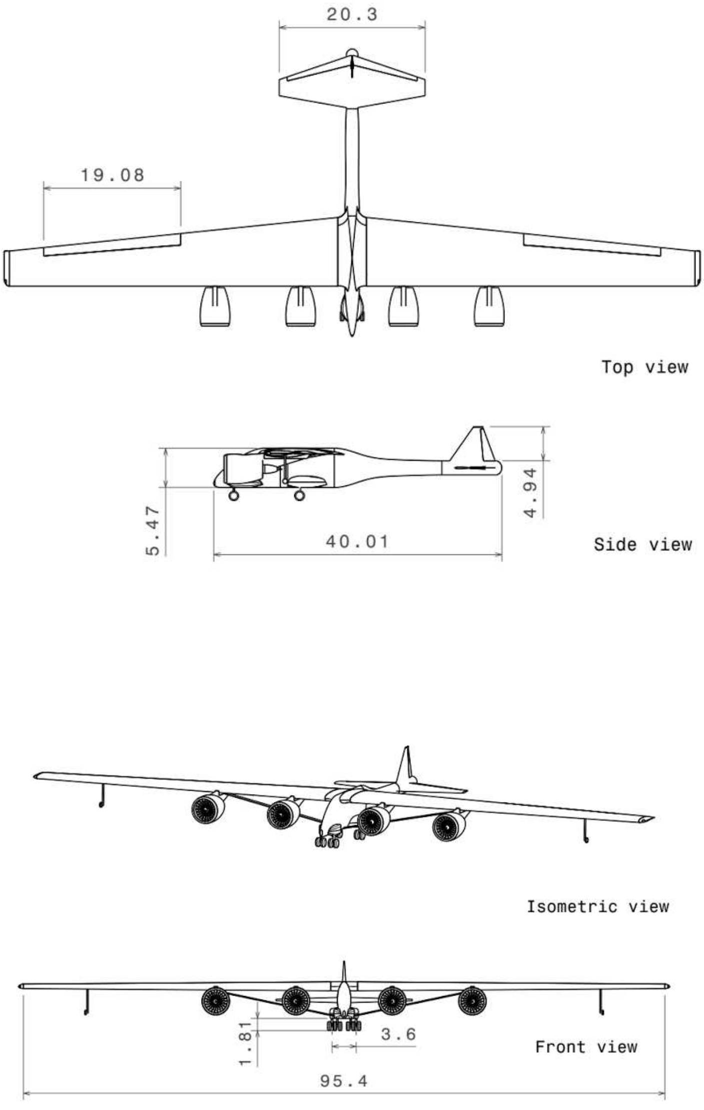

Figure 7.4: Three view technical drawing of the final SAGA design

Table 7.1 Final design parameters of SAGA

MTOW [kg]	207,000	Wing surface [m^2]	700
MLW [kg]	165,600	Wing span [m]	95.4
OEW [kg]	118,882	Aspect ratio wing [-]	13
Cruise Mach [-]	0.71	Taper ratio wing [-]	0.5
Cruise altitude [km]	18.5-19.5	Maximum sustainable thrust at 19.5km [N]	110,000

First of all, high lift must be generated in a very low density environment. The associated induced drag is minimized by employing a high aspect ratio of 13 and by designing for a high Oswald factor of 0.95. Even with a wing surface of 700 m^2, a lift coefficient of 0.81 to 1.01 during cruise is still required. The required lift coefficient cannot be reduced by increasing the speed, since the aircraft flies in the so called “coffin corner”: the airspeed range where stall Mach number and drag divergence Mach number almost coincide. Increasing the sweep angle to postpone drag divergence is not an option either, since the lift coefficient of the wing decreases with increasing sweep.

The long, slender wing is made feasible structurally by using a boron fibre composite laminate for the wing box, and by alleviating wing loads by employing a telescopic strut, which reduces structural weight by 18%. The strut only carries tension, as it becomes very heavy when sized for compression buckling, which eliminates its weight saving benefit. Even though total drag is increased, the strut facilitates a 500 kg fuel weight reduction. Furthermore, the chosen aerofoil reduces the structural weight to a minimum, while also providing a sufficient lift coefficient and a high enough drag divergence Mach number.

Another implication related to high aspect ratio is the increasing probability of the occurrence of aero elastic effects, such as torsional divergence, control reversal and flutter. When not carefully accounted for, these effects can lead to structural failure. Therefore, it is ensured the airspeeds at which these phenomena occur are well out of the flight envelope. A sufficient torsional stiffness of the wing box is provided by stacking the boron fiber composite plies in such a way that shear modulus is maximized, while a sufficiently high elastic modulus is maintained.

A second major issue related to low density is the rapid decrease of available thrust. A total thrust of 110 kN at 19.5 km is required, comparable to the thrust of an F16 with full afterburner. Six GE90 engines can provide this thrust, however, it was found this amount of thrust can be provided more efficiently by employing four purpose designed engines. To ascertain the feasibility of the engine design, it is ensured all engine specifications are achievable with state-of-the-art technology, except for the unprecedented engine diameter of 3.5 m, which is, however, only slightly larger than the current largest diameter of 3.1 m.

Thirdly, the low temperature in the stratosphere introduces the risk of freezing of the aerosol. In addition, a high aerosol temperature is beneficial regarding the fact that the aerosol will be evaporated. Therefore, the aerosol is preheated on the ground to 516 K, after which a thermally insulated wing tank, fabricated of corrosion resistant cross-linked polyethylene ensures the temperature remains well above the freezing point during the mission. In addition, keeping the aerosol at an elevated temperature reduces the power required for evaporation, which occurs just before dispersion. Aerosol evaporation requires a maximum power of 1.6 MW, comparable to the power of a wind turbine, which is provided by the four powerful custom engines.

7.6 Cost

The costs of SAGA differ from the costs of a conventional aircraft due to several aspects of the design. These aspects are aerosol dispersion, the aircraft systems, the wing design and the airport operations.

The most important effect of the aerosol payload is the fact that the aerosol is an extra cost instead of a revenue. This extra cost amounts to approximately $1.5 billion per year. The fact that no passengers will be transported has a positive influence on the costs, resulting from the fact that all aircraft systems related to passenger transport, e.g. air conditioning, pressurization and entertainment systems, do not have to be fitted in the aircraft.

Several aspects of the design which differ from most passenger aircraft are the usage of struts, composite materials, aerosol tanks and the custom engine. In addition, the large wing span introduces challenges to the ground operations of the aircraft. The implications of these aspects include high development and material costs and expensive maintenance programs and ground operations. Lastly, unmanned aircraft require a higher level of redundancy than manned vehicles and the infrastructure to operate autonomously. This results in higher system costs compared to manned aircraft. However, the reduction in cost due to the absence of pilots offsets the increase in system cost.

The total costs of SAGA are estimated to amount to an initial capital cost of $93.9 billion and a yearly operational cost of $11 billion. The initial cost exceeds the specified budget, however, compared to development programs of similar size, the initial cost of SAGA is not extraordinarily high.

7.7 Sustainability

Geoengineering by means of injecting aerosols into the stratosphere is a technology to be used for an environmental purpose. The proposed technology presents a temporary intervention to avoid unacceptable temperature increase. For an optimal implementation of geoengineering, sustainability is integrally included in each step of the process. Sustainability plays a considerable role in the preliminary design choices. An optimal combination of fleet size and payload is chosen based on fuel consumption and therefore on sustainability. In the selection of the aerosol material, environmental considerations, e.g. the chemical characteristics of the aerosol and its effect on the environment are a crucial criterion. In the trade-off between the concepts sustainability is an important aspect. This includes fuel consumption, engine characteristics and emissions, production and manufacturing methods and materials.

In the more detailed design choices, saving weight and reducing fuel consumption play an important role. For example, the composite materials resulted in a decrease in wing weight of approximately 15% and the strut-braced concept in a reduction of 18%. For propulsion, design for environment strategies are implemented by making the engine as efficient as possible. In the future, the use of bio-fuel can be investigated in order to reduce emissions even further. The ground facilities are equipped with systems to generate green energy. For the end of life strategy several solutions are considered. These include the reuse of the aircraft for scientific high-altitude measurements or modifications for private space travel. Also, recycling of parts and materials is proposed.

In order to make the mission successful, it is of utmost importance that global carbon dioxide emissions are cut down in parallel to the dispersion of aerosol. The additional CO_2 emissions due to SAGA are investigated to quantify the effect of this project, in order to show its potential. It is found that SAGA will cause an additional 2.0% fuel consumption in aviation and an extra CO_2 exhaust of 1.5%. Considering that aviation is responsible for 2% of the global fuel consumption, the worldwide fuel consumption is only increased by 0.03%.

7.8 Recommendations

In future iterations and design stages, the most highly constrained aspects of the concept design can be improved. The use of composites and a strut-braced design enable higher aspect ratios than 13 to be achieved. An increase of aspect ratio, however, introduces the need for more in-depth aero elasticity analyses to ensure safe operations within the flight envelope. In addition, the weight margin present in the current design possibly enables higher altitude flight for higher aerosol effectiveness or higher payload, reducing operational cost and emissions. However, detailed investigation of uncertainties in parameters is recommended, and (part of) the weight margin is reserved for required improvements identified in detailed design.

In order to quantify the environmental impact of the injection of sulfuric acid or other aerosol compounds into the stratosphere, thorough experimenting and modelling is required. Ozone depletion, deposition through precipitation and climate effects other than temperature reduction will inevitably affect the environment. Before a geoengineering intervention using stratospheric aerosols is implemented, examination of these aspects is required to guarantee geoengineering is a safe operation.

7.9 Conclusion

The design of an aircraft delivery system for stratospheric aerosol geoengineering has provided insights in the practical aspects of geoengineering. SAGA represents a feasible aircraft-based SRM method by the injection of aerosols into the stratosphere. The design approach focused on feasibility and practicality is intended to augment the current research on climate engineering.

The high altitude and high payload requirements drive the design of all aspects of the SAGA mission. The operational scenario, employing a fleet of 344 unmanned aircraft, enabling 572 flights per day, is established with focus on most efficient delivery of the required 5 Mt of sulfuric acid aerosol. Sulfuric acid will be ejected from the aircraft in gas phase to facilitate efficient aerosol particle formation. Aerosol dispersion will occur in the tropical region at altitudes between 18.5 and 19.5 km. The high altitude demands efficient lift generation and considerably high thrust. The need for efficient lift generation results in a design featuring an aspect ratio of 13 in combination with a wing surface area of 700 m^2. The structural integrity of this long and slender wing is ensured with the help of a strut-braced wing design. Purpose-built engines are proposed to efficiently provide thrust and power to SAGA aircraft. Four engines, each providing over 600 kN thrust at sea level, facilitate completion of the constraining mission profile and provide a sufficient amount of power to the aerosol systems.

The costs of SAGA are estimated to amount to an initial capital cost of \$93.9 billion and a yearly operational cost of \$11 billion, which are acceptable amounts considering the costs of global warming. Although aircraft are not identified to be the most economical method for stratospheric aerosol delivery, the high technological readiness level enables fast and reliable implementation. The context in which geoengineering possibly will be applied calls for a safe and reliable method, provided by SAGA.

8. Capture a Small Asteroid and Change its Orbit

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8.1 Mission statement

The mission is focused on the design of a satellite mission to reduce the risk caused by near-Earth objects. This may either be achieved by performing an observation mission, or executing a mission to directly deflect the asteroid from its path.

Since the origin of the solar system, many small bodies have collided with Earth. A large number of small bodies including minor planets and asteroids orbit the Sun between Mars and Jupiter in the so-called inner asteroid belt. In addition, there are small bodies with orbits closer to Earth, originating either from this belt or from the early solar system. Such a body is called a near-Earth object. Potentially hazardous asteroids, which are of significant size and on a trajectory that passes Earth, are monitored and analysed for NASA by the Jet Propulsion Laboratory of the California Institute of Technology. In recent missions, several spacecraft have visited some of these known bodies, for example the Rosetta and Dawn missions.

The mission statement for the proposed mission was interpreted as follows:

"Design a mission aiming to demonstrate the technology required to reduce the risk of a near-Earth object impacting the Earth."

An important aspect of this statement is that technology is to be demonstrated. Currently, there is no significant threat posed by known asteroids, but this could change in the future as more bodies are discovered. The selected asteroid for this mission was 99942 Apophis. This small body was selected because it was estimated to be a large threat to Earth from its first observations. However, later observations showed the probability of impact to be significantly smaller. An advantage of the earlier apparent threat is that the asteroid has been studied quite extensively. As an impact of a body with the size of Apophis would significantly influence the current civilisation, the asteroid was selected as a suitable candidate for the mission.

There were two options to reduce the risk of impact. The first option is an extensive observation of the asteroid to understand more of its composition, build-up, orbit, and rotation. This knowledge leads to decreased uncertainties pertaining to the asteroid, and a better evaluation of its risk. In contrast, a direct deflection mission could be executed in order to alter the trajectory of the asteroid to temporarily or permanently prevent the body from impacting Earth. The latter option was chosen, as it implied the most innovative goal, because the deflection of an asteroid has never been performed in the past.

In light of this, the most promising deflection method had to be selected. From a literature study, it was found that many different solutions have been proposed. The most plausible methods were selected, and a trade-off was performed to assess their feasibility. The three remaining concepts were to be extensively analysed, with an emphasis on their implications on the design, which would lead to a final selection of the deflection method.

Several hard constraints were posed on the mission. Firstly, the total budget could not exceed 1 billion euros, covering both development as well as the mission operation. Secondly, the designed satellite must be launched within 10 years, and be operational for at least 30 years. In addition, an existing launcher is to be used. Data acquisition was to be handled by the Deep Space Network. Finally, the sustainability and reliability aspects of the mission are of key importance.

In addition to the method selection, the spacecraft functionality was to be analysed, resulting in a system breakdown into subsystems with specific requirements and constraints. The first steps in the design, involving selecting design options for the subsystem components, were performed concurrently with the method selection. After the final concept was chosen, the spacecraft could be designed on a subsystem level.

8.2 Concepts studied and related trade-offs

Out of the nine preliminary deflection methods, three methods were studied and analysed in further detail. The first two concepts are based on heating up Apophis' surface, sublimating the material, and thus creating a thrust force due to the expelled mass. The last one only uses gravitational forces to perform the required deflection.

Solar mirror concept

The solar mirror concept makes use of mirrors, which redirect the solar light onto a small spot on the asteroid surface, sublimating the material. It was found that the proposed concept for this particular mission would involve four spacecraft hovering above Apophis, at a distance of 1,200 m. Each of these spacecraft carries a solar mirror with a diameter of 15.1 m. The estimated launch mass (including all four spacecraft), and deflection time are 621 kg and two years, respectively.

Laser ablation concept

The laser ablation concept is similar to the solar mirror concept. However, it is based on heating up the surface of Apophis using a laser rather than reflected sunlight. The difference lies in that the optimal wavelength for power absorption can be used with this concept. The laser is part of the payload of the spacecraft and is continuously directed towards the asteroid during the deflection period. The spacecraft would be hovering at a distance of 1,500 m from Apophis and would deflect it in six years using a laser with an output power of 500 W. The launch mass of the spacecraft would be approximately 300 kg.

Gravity tractor concept

For this concept, the spacecraft does not need to bring an extra payload on board in order to fulfil the mission, since the spacecraft itself will be the means of deflection. The concept is based on the gravitational attraction between Apophis and the spacecraft. The gravity force acting on the spacecraft is to be counteracted by continuously thrusting away from Apophis, in order to prevent the spacecraft from crashing into the asteroid. The result is the acceleration of both bodies in the direction of the generated thrust. The spacecraft needs to have a total mass of about 300 kg, which was found to be enough to deflect the asteroid in six years.

All these concepts require hovering above the asteroid surface for the complete duration of the deflection period. The solar mirror concept has a low technological readiness level as well as a high risk of failure, as its complexity is quite high. It also has the highest mass. The main drawback of the laser ablation concept is its high power usage to continuously power the laser. Furthermore, its overall performance in terms of the mission risks is worse than for the gravity tractor concept. A high level of uncertainty for the ablation methods is the fact that the composition of the asteroid is not fully known. The current knowledge of Apophis is based on Earth-based measurements, and uncertainties remain. The gravity tractor provides a method independent of the composition of the asteroid. It can be concluded from a trade-off based on these considerations that the gravity tractor concept is the most promising, safe and the most viable solution.

8.3 Detailed description chosen deflection method and scientific payload

There is always a gravity force present between two bodies of mass, and the magnitude of this force depends on the mass, m, of the two bodies, as well as the distance, d, between them. This gravity force also scales with the gravitational constant G. This is shown mathematically in equation 8.1.

$$F_g = G\, m_1\, m_2 / d^2 \tag{8.1}$$

The spacecraft will hover at a certain distance from Apophis, and there will be a gravity force present between the asteroid and the spacecraft. This causes the asteroid to move towards the spacecraft, but the spacecraft will also be attracted towards the asteroid, which will eventually lead to a collision between the two bodies. In order to prevent this, a constant thrust in the opposing direction must be provided in order to counter the pull of the asteroid on the spacecraft.

This causes the entire system, composed of Apophis and the spacecraft, to move in one direction, thus changing the orbit of the asteroid over time. The gravity force that must be countered by the spacecraft is low compared to what is experienced on Earth, because the mass of the two bodies is relatively small. This small thrust is however constantly produced throughout the entire duration of the deflection, resulting in a small but significant change in the asteroid's orbit. After performing a trade-off between various thruster types, it was found that ion engines are the most suitable for this purpose.

One point that must be considered regarding the thrusters is that, according to Newton's third law, to move the spacecraft away from

Apophis, the thrust direction must be towards the asteroid. However, the thruster plumes must not hit Apophis, as this would counteract the effect achieved by the deflection method. For this reason, the thrusters must be positioned in a way that their plumes do not hit Apophis and are instead tangent to the surface. This is shown graphically in figure 8.1, where αt is the angle of the plumes expelled by the thrusters.

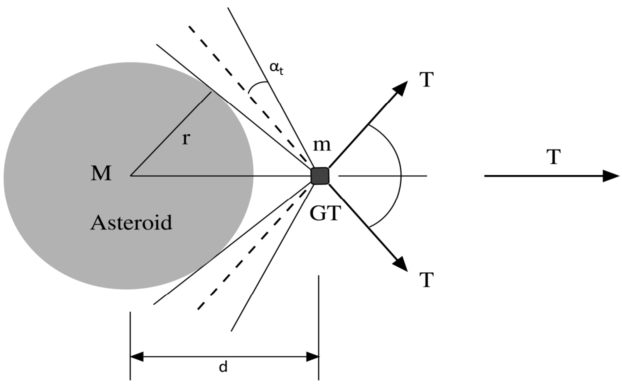

Figure 8.1: Graphical representation of the angle of thrust

Originally, the main focus of the mission was the deflection of Apophis. However, as there is still much to discover about asteroids, and more specifically Apophis, it was decided to include an extra scientific payload. The main objective for this payload is to determine different characteristics of Apophis, to reduce any uncertainties that might influence the design of future deflection missions.

The main characteristics of interest are the surface composition, the albedo, the temperature, the dimensions and the shape, the presence of a magnetic field, the topography, and the tumbling rate of Apophis. Obtaining more information on these characteristics can aid future deflection methods, even those with other target asteroids.

Seven sensors are included in this payload: an X-ray and infrared spectrometer, a polarimeter, two magnetometers, a laser altimeter and a framing camera. At the start of the mission, during a period of approximately six months, the spacecraft will orbit Apophis at a distance of 1,500 m and will use these sensors to measure the previously mentioned characteristics of the asteroid. This observational period consists of three phases: the three-dimensional mapping of the surface using the framing camera, obtaining asteroid characteristics using the spectrometers, polarimeter and magnetometers and, finally, performing crater investigation with the camera and altimeter to obtain more information about the asteroid's history and its origin.

In addition to this, the selection of the launcher and launch date were also based on a conceptual, optimised, ΔV budget, to prove the feasibility of the mission.

A three-dimensional model of the solar system was made, thus the position and velocity vectors within the selected coordinate system could be determined at any point in time.

By solving Lambert's problem, the trajectory between two positions was found, which is unique for a certain travel time. As all input parameters are known from the solar system model, the initial and arrival velocity needed can be calculated for any departure date and travel time. From these velocities, the transfer trajectory orbital parameters can be calculated. As the velocity vectors relative to the Sun before departure and after arrival are given by the solar system model, the difference between the velocity before and after departure and arrival results in the required ΔV.

Then, different porkchop plots were made, where the value for ΔV is plotted for different departure dates and travel times. In the plot, favourable regions can be identified in which the ΔV value is minimised. This is seen in figure 8.2, where a region of specific interest is shown. In addition the arrival ΔV was considered, as this would minimise the injection thrust requirements.

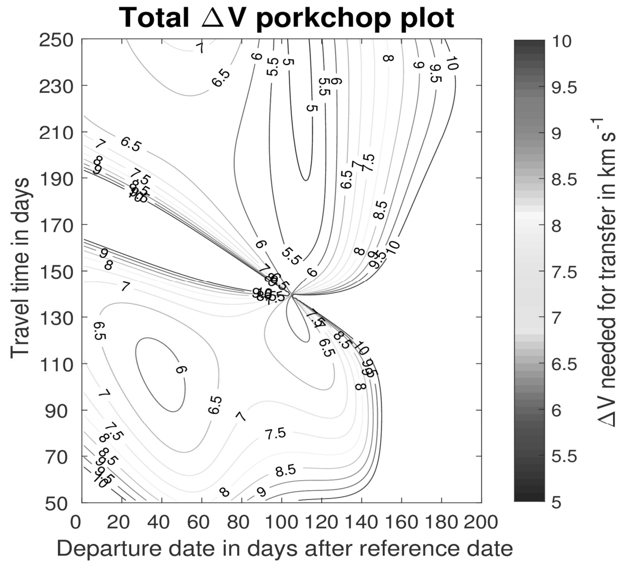

Figure 8.2: The total ΔV for different departure dates and travel times. Reference date is January 1, 2021

The final launch date was determined to be the 21st of April 2021, with a travel time of 245 days, the total ΔV of the transfer was found to be 4.82 km/s. The selected launcher, the Soyuz Fregat, was capable of delivering the velocity increment and sending the spacecraft on the determined trajectory. It is the least expensive and the most reliable option for the current mass of the spacecraft. After the deflection sizing, the scientific payload sizing, and the launcher selection, a conceptual design of the spacecraft could be performed.

8.4 Conceptual design of the spacecraft

The spacecraft is sketched in figure 8.3, which the high gain antenna and solar panels expanded.

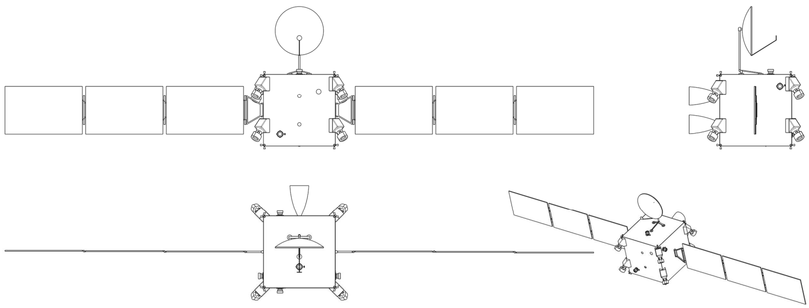

Figure 8.3: The spacecraft viewed from different positions

The final design of the mission can be summed up with the characteristics seen in tables 8.1 and 8.2. The deflection range is due to the uncertainty regarding the mass of Apophis.

Table 8.1: Mission characteristics

Mission cost	$400 million
Observation time	3 months
Deflection time	5 to 6 years
Deflection distance	11.8-15.3 km
Launch date	21-04-2021
Date of arrival at Apophis	22-12-2021
Launcher	Soyuz Fregat, Arianespace

Table 8.2: Spacecraft characteristics

Spacecraft mass	503 kg
Spacecraft dimensions	1.5 x 1.5 x 1.5 m
Maximum power demand	2,084 W
Solar array area	9.65 m^2
Thrusters	Ion engines (8x) Chemical ADCS thrusters (24x) Chemical orbit insertion thrusters (2x))
Payload	X-ray spectrometer, infrared spectrometer framing camera, magnetometer, laser altimeter, polarimeter

The orbit insertion propulsion system was chosen to be a liquid bi-propellant system, whilst the hovering propulsion system uses ion thrusters. For the command and data handling system, the CPU chosen is the RAD6000, with an SSD data storage system. The communications subsystem consists of two transponders, two power amplifiers, a diplexer, a high gain parabolic antenna and three low gain horn antennas, amongst other components. These were selected based on required communication times and distance from the ground station on Earth. The thermal control system was sized according to the required component temperature ranges, using a combination of passive and active control. Passive control systems include Teflon, Beta cloth, Basotect foam, and passive radiators. On the other hand, active control consists of thermoelectric coolers, deployable radiators and patch heaters.

The power system consists of a multi-junction solar array (sized to 3.62 m^2) and a battery with a capacity of 1,344 Wh. Finally, the structural bus was sized to have a final mass of 43.2 kg, an outer thickness of 0.5 mm and a propellant tank thickness of 2 mm, leading to a spacecraft volume of 3.34 m^3.

8.5 Conclusion and recommendations

The final design of the spacecraft was well within the top constraints of the mission. The selected launcher is the Soyuz Fregat, to send the spacecraft with a total mass of 503 kg on its way. It is designed to deflect the asteroid with a distance of 11.8 km, in a time of 5 to 6 years (depending on nominal or worst case estimations of the asteroid mass). In order to meet the mission requirements, the observation time was restricted to six months and based on this, the scientific instruments were selected. It will observe the asteroid using a framing camera, an X-ray and infrared spectrometer, two magnetometers, a laser altimeter, and a polarimeter.

As the mission design is still in an early phase, some recommendations can be made to improve the design in the future. First of all, every model used can be elaborated on and treated in more detail to get an even better overview on the design conditions and constraints. Next, it would be interesting to have a look at shorter mission times and optimise the design for time, mass and other important parameters. During transfer, a barbecue mode could be implemented, which basically means the spacecraft is continuously rotating in order to make the thermal control more efficient and not have the heat concentrated on one side.

More detail could be put in the instrument and component selection, for all subsystems as well as for the scientific payload. There might be more suitable components available in terms of mass, power and other specifications, which were not found due to time constraints. For the orbit insertion, the spacecraft uses two thrusters, which could possibly be reduced to only one. This can be investigated in a later design phase. As the total cost of the mission is still under the budget, the spacecraft could be given extra functionality.

A good example would be implementing a way to actually gather a material sample of the asteroid for further analysis and examination. The scientific community is extremely interested in actual samples, so this could be an interesting opportunity for further design.

9. MICRO - PAVING THE WAY FOR INTERPLANETARY CUBESAT MISSIONS

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9.1 Introduction

Spacecraft development has evolved over the past two decades not only in terms of technology, but also in terms of size and mass. The purpose of cubesat development is to provide a standard for the design of micro-satellites to reduce development costs and time, increase accessibility to space missions and sustain frequent launches. The miniaturisation technology and its benefits have been demonstrated by cubesats orbiting Earth. However, this technology has not been demonstrated yet for interplanetary space missions.

The Piggyback to Mars project explored the capabilities of extending the cubesat technology to perform a scientific mission on Mars. The scientific interest in Mars is steadily increasing, with many missions underway in the next five years, making Mars the de facto option for piggybacking.

9.2 Requirements

To properly design the cubesat, a set of requirements had to be fulfilled. These can be found in the list below:

- The cubesat shall communicate with a data relay satellite
- Launching the cubesat shall not jeopardize the host mission
- The cubesat shall comply with the COSPAR recommendations on planetary protection
- The cubesat shall have a size of 6U
- The cubesat structure shall comply with cubesat standards
- The mission shall have a total lifetime cost of at most 10 million euros
- The mission shall provide unique scientific measurement

Due to the little amount of constraints, the design had an unexpected level of freedom. For the group, this meant that first a suitable scientific mission had to be selected. After thorough research in the field on Martian sciences, the group came up with four options. These are shown in the section 9.3.

9.3 Mission concepts

In the first phases of the project the science mission was undefined and up to the group to decide. After coming up with multiple mission concepts in the beginning of the project, three of them were developed further. The trade-off between them was made in the Mid-term report based on the science case and performance of the mission. A brief description of each concept is given below.

Methane analysis

There are several reasons for methane to occur on a planet: volcano eruption, comet impact, hydrogeochemical landscape or from a microbial source. On Mars, the only two possible sources are from hydrogeochemical (carbon-12) or microbial activity (carbon-13). Mapping the distribution and isotope of methane provide insight in the source and location of methane.

Mapping of the Martian moons

There have not yet been dedicated missions to the moons of Mars, Phobos and Deimos. Due to this, even their origin is still uncertain. An analysis of their internal composition would give clues about the origin of the moons, as well as information on the ancient history of the Solar system.

Dust analysis

The surface of Mars is covered with a thin layer of dust. While its effects have been studied for decades, a lot still remains unknown. The atmospheric dust has been found to be about 3 μm in diameter. These tiny particles play a role in the climate system on Mars and possibly pose a threat to future human Mars occupation. Studying the microphysical properties (size, shape and composition) allows for better understanding of the Martian climate and aid in the selection of future landing sites on Mars.

Trade-off

The trade-off between the concepts was made based on four criteria: scientific value of the mission, the required ΔV and the availability of both the payload and a suitable launcher. The moon mapping mission was deemed unfeasible due to the lack of host missions to them which made piggybacking impossible. The dust and methane missions had the same available launchers and the selection between the two was based on their relative ranking with respect to the other criteria. In scientific value, the dust mission was given a higher score than the methane one due to the fact that methane has been studied extensively.

The orbits used for the missions would be identical, leading to a tie for the ΔV criterion. The last criterion, instrument availability, solidified the selection of the dust mission. The instrument selected for analysing dust, SPEX (Spectropolarimeter for Planetary EXploration), has been developed by SRON in the Netherlands and is already available. For methane analysis, the prototype instrument BIRCHES (Broadband Infra-Red Compact High-resolution Exploration Spectrometer) was selected, meaning its availability is worse than that of SPEX. As a whole, the dust mission is as good or better than the methane mission and is therefore selected to be developed further.

Limitation on the volume and mass budgets put constraints on the transfer option to Mars, mainly in terms of a large budget allocated to the propulsion subsystem. Transferring to Mars without the piggyback option does not leave enough room to house the other subsystems. In order to reach Mars in time and still carry the subsystems, the piggybacking option is chosen. Piggybacking will happen with the Mangalyaan 2 mission, from the ISRO (Indian Space Research Organisation) with the GSLV-III launcher. The launch of the Mangalyaan 2 mission is planned for 2020. Therefore, MICRO has to be ready by that time.

9.4 Mission description

There is a need to analyse the dust particles on Mars to aid in the selection of future Mars landing sites to protect the prospective astronauts and obtain knowledge on the properties of dust. MICRO, or Mars Interplanetary Cubesat and Research Orbiter, first has to be assembled, integrated and tested in TU Delft. The final assembly of MICRO has to be transferred to the GSLV III launcher in July 2020 to piggyback with the Mangalyaan 2 mission. Mars will be reached between January and March 2021.

The cubesat will be deployed from the host in a Sun-synchronous orbit in an altitude range of 200-500 km. MICRO will transfer to a Sun-synchronous circular orbit of 400 km due to the payload operational orbital range. Dust is one of the main contributors to the climate system on Mars. It effects for example cloud formation and cloud lifetime. Mapping the dust on Mars provides quantitative information on its effect on the weather on Mars, which can be used to further understand the Earth's atmosphere. On Mars, dust storms occur frequently: these can cover the whole planet. These make the missions to Mars riskier due to reduced solar radiation received and increases potential damage caused by dust particles. These dust particles are small and might contain silicates which can be chemically reactive, inhaling these particles may pose health problems. Furthermore, the particles can be statically charged, making them sticky. Along with the small size, this means that they can get stuck inside of the electronics, causing damage.

The primary mission goal is to support the selection process for future Mars landing sites, adapted from existing NASA and ESA missions. These landing sites are located between -30º and 30º latitude around the equator. By analyzing the microphysical properties of dust at these locations, along with their spatial distribution and temporal change due to dust storms, a recommendation can be given on the most favourable landing sites. In addition, due to the configuration of the scientific payload selected, information is provided simultaneously on the ice cloud properties due to the two limb viewing angles. After performing the primary mission, more regions on Mars can be measured. Or scientific results obtained from rovers and landers on the surface can be re-measured and validated.

The properties of dust can be derived from the information on the linear polarisation of sunlight as it is scattered by the dust in the atmosphere. SPEX measures the degree and angle of linear polarisation along with the flux of the scattered light. To analyse the dust properties, the degree of linear polarisation is plotted against the scattering angle of the sunlight for each measured location. This requires accurate location and lighting condition knowledge that has to be collected by the attitude and orbit determination subsystem. There are seven ground viewing angles that measure dust properties and two limb viewers for analysing the ice clouds. These angles allow for measuring each location at seven different scattering angles as the cubesat flies along its orbit. This is shown in figure 9.1.

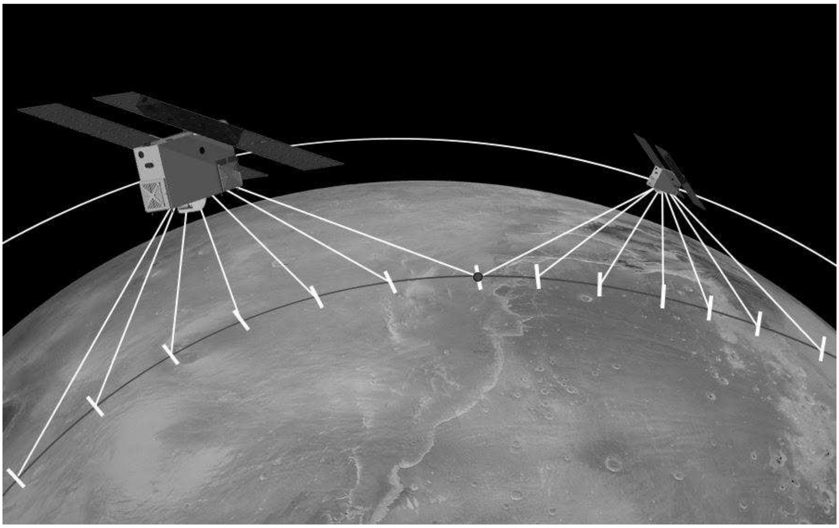

Figure 9.1: The viewing angles of SPEX shown with MICRO at two different locations

9.5 Subsystems and configuration

Subsystems

MICRO has on-board capability to determine its attitude with three star trackers along with an inertial measurement unit. Attitude control is performed using four reaction wheels. The position along the orbit, on the other hand, is determined by detecting Martian landscape features. The propulsion subsystem in MICRO is a dual unit mono-propellant system, that uses a green propellant to provide a specific impulse of 240 s. The two units have a combined capability to provide a ΔV of 248 m/s. One of the most fundamental requirement to the mission is the ability to generate and store power. The MICRO power subsystem uses high performance quadruple junction solar cells and has a total solar panel area of 0.273 m^2. The power subsystem uses an array drive actuator for active sun tracking and maximum peak power generation. The addition of two batteries gives the ability to perform during eclipse, as well.

MICRO obtains data using SPEX which is then processed by the on-board computer. The data collected is transmitted back to Earth with the aid of the Mars Reconnaissance Orbiter, acting as relay. Communication is performed using five micro strip patch antennas, operating in the UHF band, which enable communication in all directions. The 6 unit cubesat uses a custom made monocoque for load distribution and subsystem integration. This also protects from radiation, using graded shielding. MICRO has a passive thermal control system, and capability to maintain a working temperature in the range of -5 to 27 ºC.

Configuration

All the components listed above are arranged in the configuration shown in figures 9.2 and 9.3.

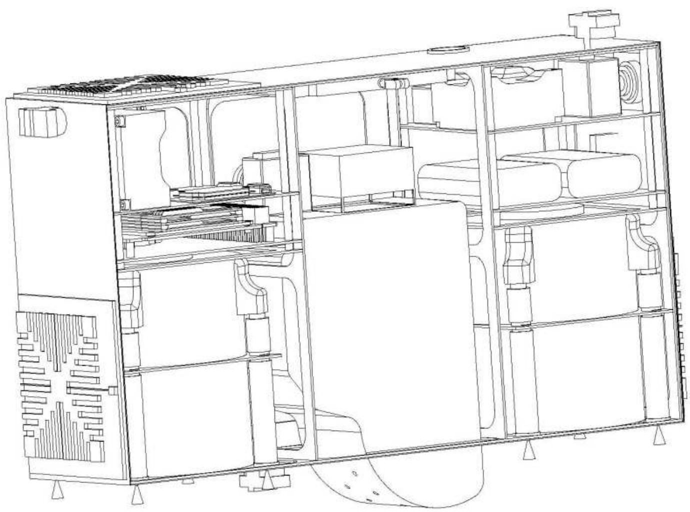

Figure 9.2: Side isometric view of MICRO cubesat

In figure 9.2 one can see all the subsystems fit in the cubesat, leaving some extra space at the top, for a possible secondary mission to be added in future phases of the design. Furthermore, in figure 9.3 the exploded view of the cubesat can be noted.

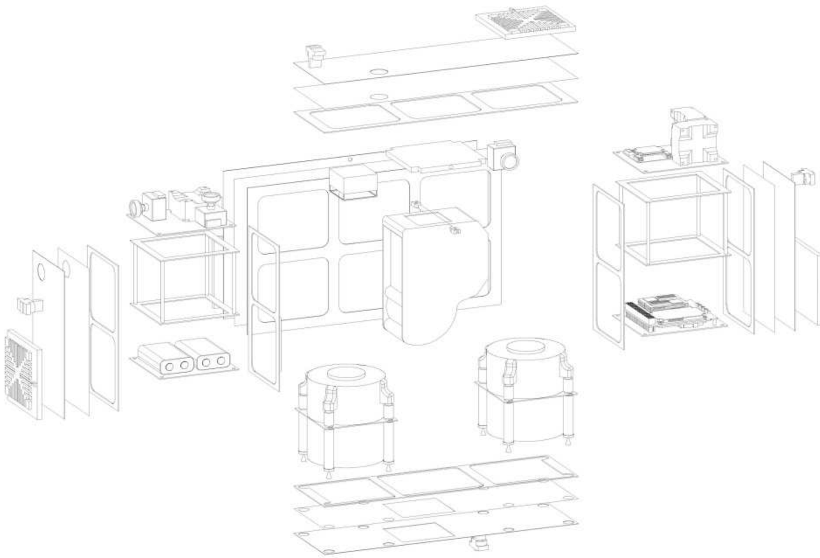

Figure 9.3: Exploded view of MICRO cubesat

9.6 Sustainability

The MICRO mission addresses sustainability by means of using commercial off-the-shelf components and green propellant. In addition to the sustainable design solutions, 3.16% of the budget is allocated for protection of the Earth's environment, by planting trees to compensate for the carbon footprint imposed at launch.

9.7 Conclusion and recommendations

The main conclusion that can be drawn from this DSE project is that interplanetary cubesat missions are feasible. MICRO would perform a scientific cubesat mission that provides unique science measurements while orbiting Mars. Apart from the data it would provide, it would demonstrate new cubesat technologies. Nonetheless, it is important to emphasise that the mission success would depend on external factors. A launch opportunity should be provided and the mission success would partially depend on the use of MRO. Furthermore, the mission would comply with all the requirements set at the beginning of the project.

Based on the last conclusion, it is recommended for the design of future interplanetary cubesat missions to first find a launch opportunity before initiating the design itself. This is essential because interplanetary cubesat missions or piggybacking options are non-existent at the time of writing. Finally, it is recommended for space agencies and private space companies to invest in interplanetary cubesat missions and take the next step towards the expansion of interplanetary space exploration. This way, the advantages of miniaturisation already proven for Earth-bound cubesats, can be extrapolated to interplanetary space missions.

10. WINGS FOR AID

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10.1 Introduction

Traditional means of transportation are often inadequate in inhospitable areas where humanitarian aid is required, such as in a warzone or following a natural disaster. One of the biggest challenges that humanitarian aid organizations face is the lack of infrastructure and logistical organization in the affected areas. These challenges make delivering of sufficient aid in a timely manner extremely expensive. The Wings for Aid initiative aims to deliver aid to those in need in disaster areas quickly, efficiently and at a low cost through the use of a set of rapidly deployable unmanned aerial vehicles (UAVs).

The aim of these UAVs is to be able to transport five 20 kg packages over a distance of 500 km. The packages will contain food, water and blankets sufficient to provide relief to victims on the ground for several days. These goals resulted in the following need statement and mission statement for the wings for aid UAV:

Need statement:

"There is a need to provide fast, low cost and accurate humanitarian aid in areas which are hard to reach by conventional aid transportation on a short notice."

Mission statement:

"Wings for Aid will deliver accurate emergency humanitarian aid with a UAV within a 500 km range, in areas hard to reach for conventional transportation, with a lower operating cost than a helicopter and a unit cost price less than €15,000."

10.2 Stakeholder requirements

In several discussions with the customer the following stakeholder requirements were agreed on:

- The UAV shall be able to perform take-off and land on rough terrain in 150 m.
- The unit cost of the UAV shall be less than €15,000 based on the bill of materials and unit production cost.
- Operating cost shall be less than that of a UH-1 helicopter.
- The round trip distance shall be higher or equal than 500 km with 100 kg payload.
- The UAV shall fit in a 20 ft container.
- The UAV shall be designed for replacement of broken parts.
- The UAV shall be equally or more stable than a Cessna 172.
- The payload shall be dropped in 5 different packages of 20 kg each.
- The UAV shall be able to deliver a payload of 100 kg at least 3 times within 24 hours.
- The UAV shall be able to perform in similar conditions as a UH-1 helicopter.
- The UAV shall be able to deliver a package within a square of 25x25 m.
- The delivery capacity of the UAV swarm shall be at least the same as a C-130 Hercules.

10.3 Ground operation design and trade-off

Due to the large amount of aid which has to be delivered in a short time, it is required that many UAVs are operational at the same time. Therefore, a smooth and fast ground operation has to be designed. Because of the size of the operation it is too large to operate from an airport. Therefore, another field will be used. This section shows the considered concepts and the trade-off.

Ground concepts

For the ground operations four concepts were considered. The first concept is called the Brodie system. This system was developed in the Second World War by Capt. James H. Brodie. It can be used for operations where a normal runway is not present. The system contains four poles and a set of cables. The Brodie system is shown in figure 10.1. The UAV is attached to a cable by a hook on the top of the fuselage which is hooked to a sling. The start of the take-off is at the back and once the UAV is winched to the other end the cable detaches and the aircraft continues its flight. The aircraft will also land on this system by flying the hook into a sling.

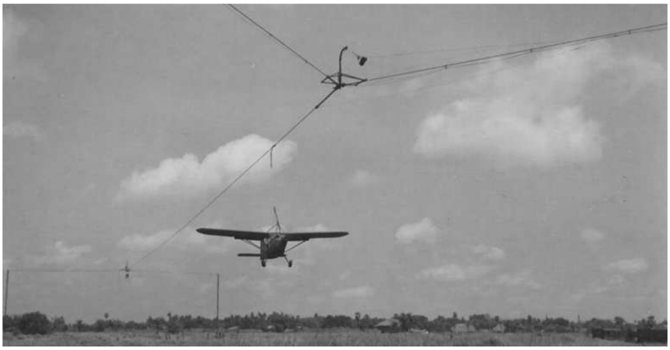

Figure 10.1 Brodie system

The second considered concept is the winch and arresting wire system. The winch system is used for take-off in the same way as it is used for glides aircraft. Furthermore, the arresting wires are used for landing as used on aircraft carriers. The UAVs will land on one side of the field and then it is transported to the loading and maintenance site of the field. Once the drone is ready to go it is attached to the winch again and it starts the next mission.

The third considered system is the catapult launch and airbag landing system. The logistical plan of this system is first to land on the airbag by a post stall manoeuvre. Then the UAV is removed from the airbag and placed on a cart. This cart can be rolled to the loading and maintenance site. After the UAV is reloaded is it placed on the catapult and launched into its next mission.

The last considered concept is the conventional take-off and landing concept. This means that the UAV will take off and land just as normal fixed wing aircraft do, however it will likely not be able to land and take-off on a football field. Once the UAV has landed it will taxi to the reload and maintenance site, and after reloading and checks it will take off again.

Trade-off
For the trade-off each concept will be evaluated on the take-off and landing distance, system and operating cost, guidance complexity, UAV capacity, constraints on UAV, and setup time and complexity. From the trade-off with these criteria the conventional ground operation had the best overall score. Therefore, the conventional operation is chosen.

10.4 Configuration concept trade-off

With the ground operation concept known, the configuration of the UAV itself was chosen in another trade-off. This trade-off was between a conventional configuration, a blended wing body and a canard. The chosen concept was a conventional set up. This is due to the simple design to lower cost. The large centre of gravity range allows packages to be placed behind each other in the fuselage.

10.5 Final design

Configuration

A conventional aircraft configuration came out the trade-off as the most suitable solution to the problem. A high wing position was chosen to have better lateral stability. Furthermore a high wing position provides the most place for the packages and the wings structure means no obstruction for the dropping of the packages.

A puller propeller configuration was chosen to avoid stones launched by the wheels hitting the propeller, and to allow packages to be loaded from the rear. The tail cover can hinge as can be seen from figure 10.2 and a rack with payload packages can be pushed into the back. The fact that all 5 packages can be loaded at once significantly decreases the turnaround time. Moreover also the fuel tank will be integrated in this payload rack. This allows the fuelling and preparing of packages to be performed in a designated safety zone.

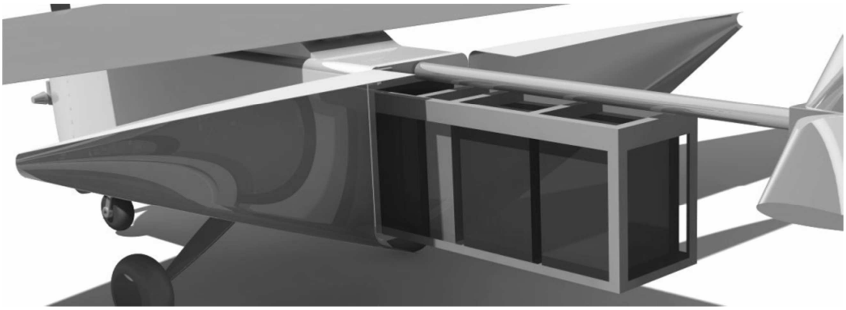

Figure 10.2 Reloading of the UAV

Furthermore a nose wheel landing gear set-up was chosen to be able to easily slide in the payload rack from the rear, prevent propeller strikes and to simplify the take-off and landing procedure that has to be programmed into the avionics software.

Structure

For the fuselage structure a trade-off was executed between a main beam structure, a truss structure, and a load carrying skin structure. After the trade-off the third concept was chosen for the structural design of the fuselage. This structure is very lightweight and makes easy manufacturing possible, because its possibility to reuse almost all components.

After a trade-off of different structural concepts, a sandwich structure was selected as the final wing concept. This structure would be made with expandable PU foam as the core material, and aluminium as skin. The core will be formed by letting the foam expand in a mould, and then glued to formed sheets. Plastic inserts, manufactured by injection moulding, will be used to connect the wings to the fuselage and the modular wing tips.

The structure for the wings of the empennage is the same as the structure of the main wing. For lower production cost the vertical wing has been designed such that it is the same as the horizontal tail wing. Therefore, each product in the tail wings can be made the same.

The landing gear has a conventional tricycle configuration. The landing impact energy is taken up by the elastic deflection of the main gear's struts, making a more cost effective construction.

Modularity

The UAV will be made modular in the tail and wing. This means that two UAVs can be stored in one 20 ft container. Furthermore the transport is easy as the UAV can be rolled out of the container. If the landing gear and propeller are detached as well, four UAVs can fit in a 20 ft container. However, this will make loading more difficult. Apart from the easy accessibility, the modularity additionally will ease the maintainability at the ground site. Broken modular parts can be replaced by spare parts.

Specifications

The most important characteristics of the UAV are given in table 10.1.

Table 10.1: UAV specifications

Weights		Dimensions	
Maximum take-off weight	350 kg	Wing taper ratio	0.4
Operational empty weight	212 kg	Wing aspect ratio	7
Payload	108 kg	Wing area	9 m^2
Fuel weight	32 kg	Horizontal tail area	1.65 m^2
Performance		Propulsion	
Runway length	139 m	Engine	Victor 1 Super
Cruise height	150 m	Maximum power	40 kW
Cruise speed	43 m/s	Propeller diameter	1.52 m

The main physical dimensions of the UAV are given in the technical drawing in figure 10.3. The dimensions are in meters.

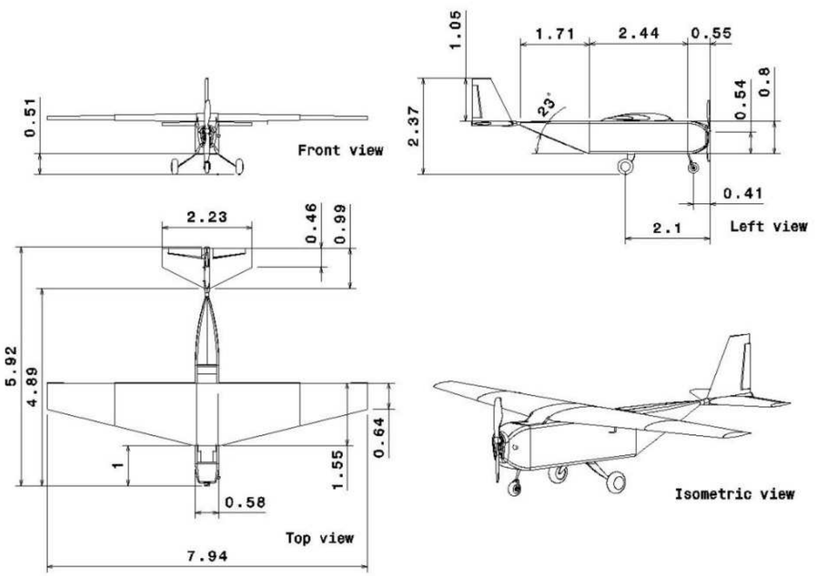

Figure 10.3: Technical drawing of the Wings for Aid UAV

Unit cost

The unit cost was minimized by using a simple design with as much commonality as possible. In practice this resulted in using standardized structural elements and off-the-shelf components to decrease the unit cost. An open fuselage structure allows easy accessibility. This eases assembly and therefore production cost. Furthermore commonality in the horizontal and vertical tail planform and control surfaces and a single stiffener cross-section in the fuselage decreases the production costs. This resulted in a unit cost of €14,790.

Dropping

For the dropping manoeuvre two different scenarios were considered. Starting with the warzone scenario. In this scenario the UAV has a cruise altitude of 1000 m at which it flies to the drop zone and starts to circle around the drop zone waiting for a go or no go signal. Once the go signal has been received it circles down to 150 m where it determines the wind direction and then descends to 15 m to drop the package while flying slightly above its stall speed. In a non-warzone scenario the cruise altitude will be 150 m to stay out of the way of the conventional airspace. In this case the UAV will simultaneously determine the wind direction and wait for the go signal while circling at 150 m and then proceed to complete the same drop manoeuvre as in the warzone.

Take-off and landing

To obtain the short take-off and landing distance various measures were taken. Simple flaps were used for their low cost, while still providing a good amount of lift. Furthermore the ailerons were deflected down 10° to increase the lift even further and the engine was slightly oversized. This resulted in a take-off distance of 129 m at sea level, and a landing distance of 139 m at sea level on a dirt and gravel surface. Additionally the landing gear was designed with a large stability margin and large tire sizes were used to be able to operate on rough terrain.

Operating cost

The operating cost were determined by combining the fuel use, maintenance cost and ground operation cost. The fuel use is 35.9 €/mission for a natural disaster and for a warzone mission 45.6 €/mission. The maintenance cost consist for the most part of the engine maintenance cost which is based on the maintenance schedules of the engine and will be €32.40 per mission for a natural disaster and €38.5 per mission for a warzone . The ground operation cost consist of the required personnel and the food for the required personnel. The food of the personnel is estimated to be 1.7 €/mission and the personnel is divided in three groups. There are specialist, trained locals and volunteers. They get paid respectively €40, €3 and of course the volunteers are not paid at all. Combining these values results in a ground operation cost of €37.76 per mission. When all the value are added this result in an operating cost of €107.8 per mission for the natural disaster and €123.6 per mission for the warzone.

Operations

For rapid deployment to the disaster area the UAV was designed to fit in a 20 ft container. This was done by using a modular design where the wing, wing tips, empennage, landing gear and propeller could be removed. With all parts separated, four UAVs could fit in a single container, meaning that the entire swarm can be shipped with 11 containers. Furthermore the UAVs are designed such that the payload bays can be replaced with fuel increasing the ferry range to 2,572 km. Using six main hubs the most common disaster areas can be reached in one ferry trip as shown in figure 10.4. Here the large circles around each hub are the ferry ranges with extra fuel tanks and the smaller ones are without these tanks. The crosses are historical disaster areas.

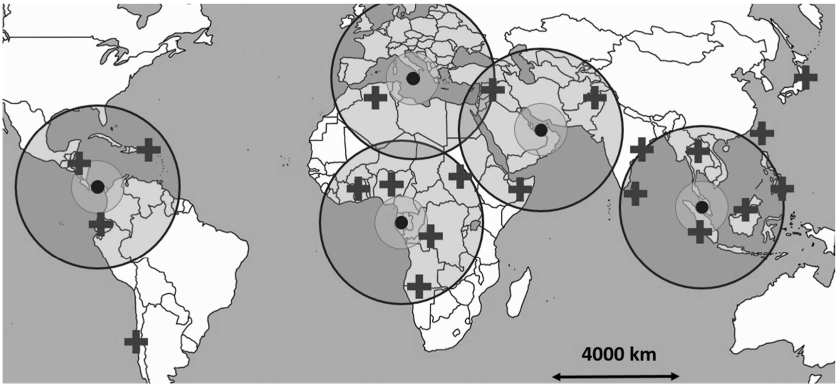

Figure 10.4 Locations of the logistical hubs with historical disaster areas

10.6 Conclusions and recommendations

Conclusions

Different concepts for both the operational aspect and the UAV aspect were considered to come to the design presented in this report. The Wings for Aid concept can operate at full capacity within 72 hr from any location within 3800 km of the central logistical hub using the ferry concept to transport the UAVs. The turnaround time of every UAV during operation is 29 min but because an assembly line strategy is used a UAV can take off and land every 6.4 min. This means that for a warzone mission a minimum of 50 UAVs and for a regular mission a minimum number of 43 UAVs is required to reach the required capacity requirement of 22,500 kg delivered aid supply within 24 hours. The operating cost for the UAV are €28,000 to deliver 22,500 kg of goods to a location 500 km away from the ground site. This means that the Wings for Aid UAV can compete with the C-130 Hercules (€32,678) and the UH-1 Bell Huey (€897,000) from a cost perspective.

Furthermore, the UAV units cost is below €15,000 at €14,790. This leads to the conclusion that the Wings for Aid concept is a financially feasible concept that will be a dominant competitor in the aerial aid delivery market by outperforming other competitors on aspects like delivery time, operating cost and effective distribution of aid supplies.

Recommendations

There are of course some recommendations which can be made for future development of the Wings for Aid UAV. These recommendations are listed below.

- Performing CFD analysis of a complete UAV model to get better values for lift and drag coefficients, as well as determining the actual effectiveness of the control surfaces and stability derivatives for the stability and control of the UAV.
- Performing field tests to determine the actual efficiency of the engine and propeller, to test the landing, take-off, climb and cruise performance in a representative environment and to see if the loading can actually be done pretty easily.

- Taking a better look at the connection between the main structural elements to see if the loads can be transferred between them. And to perform the structural trade-off for the wing and fuselage at a later stage after other concepts have been worked out more into detail.
- Take a better look at operational circumstances which can prove difficult for the UAV, like lightning strikes, communication with a ground station during ferrying or cruise conditions which require de-icing systems and how to cope with these difficulties.

11. DESIGN OF A UAV BASED SYSTEM FOR NAVAID CALIBRATION AND TESTING

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11.1 Introduction

One of the issues with flying longer distances is being able to find your way through the immense sky. Therefore, navigation of aircraft has become an important topic. One of the systems used at airports to provide accurate navigation data is the instrument landing system (ILS). This system will allow aircraft to fly to the runway in all conditions, where the runway might not be visible at all. Depending on the accuracy of the ILS system it can be used up to zero feet decision height and 150 feet visual range (CAT IIIb), only limited by the ability to visually drive off the runway. In order to provide such a high accuracy, the navigation equipment needs to be calibrated on a regular basis. Currently, inspection of the equipment is performed mostly by twin-engine business jets or small four seat propeller aircraft. These aircraft are expensive to acquire, operate and maintain. The low altitudes needed for flight inspection means suboptimal fuel use and high noise levels. Moreover, because of the limited amount of flight inspection aircraft in some regions, inspections can only be performed after careful planning and leaves little room for flexibility. At the moment, the industry of UAVs is growing and therefore, a new solution in the form of a UAV has been investigated and designed.

Figure 11.1. Final concept design

For this project, the following mission need statement has been defined:

"Accurate testing and calibration of aviation navigation systems, such as ILS, DME, VOR, TACAN and NDB, needs to be done in a more cost effective way than current systems, with special emphasis on producing less noise and causing less interference with aerodrome operations."

The aim of this project was to develop a UAV based system that is able to replace the aircraft that are currently used for the calibration of aerial navigation equipment. The aircraft that is currently used for calibration in the Netherlands is a Cessna Citation II. This Citation is fitted with a multitude of extra sensors and systems such as a high-accuracy Global Positioning System with phase tracking, a multichannel digital data acquisition and recording system and a telemetry system. The goal of the design was to be more efficient, to be more sustainable, to reduce the interference with aerodrome operations and to produce less noise, while still meeting the same performance requirements.

11.2 Mission analysis

In order to calibrate navigation equipment, certain profiles need to be flown that enable the UAV to pick up signals radiated by the different systems. The ILS positions an aircraft within the proper approach path. The boundaries and centreline of this system therefore have to be calibrated by performing several approaches and a number of partial orbits at different distances from the antenna. Most of the other systems can be calibrated by flying a full orbit around the beacon at a given distance. All of these profiles are performed at a maximum altitude of 5,000 ft above the ground with typical airspeeds around 180 kts. Depending on the availability of the airspace, flight time ranges from 1 to 5 hours with a maximum range of 800 nm.

11.3 Design requirements and constraints

The requirements and constraints as set by the customer are listed below.

Performance

- Range > 800 nm
- Endurance > 4 hours
- Maximum cruise speed > 200 kts
- Service ceiling > FL100

Safety and reliability

- Pre-programmed and in flight updateable automated flight
- Manual control systems
- Crash rate < 10^{-7} per flight hour
- Compliant with relevant EASA standards

Sustainability

- 20 dBA lower sound exposure level than a Cessna 550 on a 3° glideslope
- More than 80% of empty weight should be recyclable

Engineering budgets

- Weight < 500 kg
- System (air and ground support) must be transportable in an LD6 container

Cost

- Production volume: 100 units
- Air system unit production cost < €250,000

Other

- Flexible payload for possible future missions
- Ground based positioning system with accuracy of less than 1 m

11.4 Concept selection

After an initial analysis in the requirements and functions of the system, 24 concepts were created. A first trade-off process was used to eliminate non-feasible or inferior concepts from this list of 24 concepts. By doing this, four concepts were chosen for the conceptual design phase. In the trade-off, the concepts were scored against a comparative benchmark (the conventional piston engine fixed-wing aircraft) on all criteria. The four concepts that were chosen to be developed in more detail for an extensive trade-off were a piston-powered fixed-wing aircraft, an electric-powered fixed-wing aircraft, an electrically propelled airship and a turboshaft propelled tiltrotor aircraft.

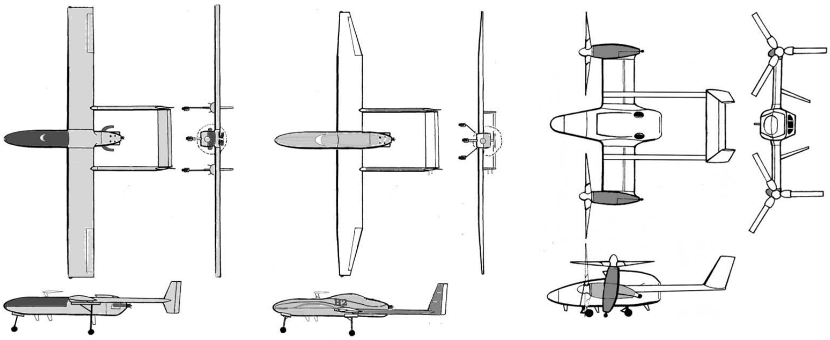
The concept design phase started with a general design which was applicable to all four chosen concepts. First of all, a basic investigation had to be performed on what must be included as payload for this mission. By doing a technological survey, a detailed list was created stating examples of the antennas, avionics sensor units, signal processing units and telemetry systems. The second part of the general concept design was to define the mission profile. In the specific concept designs, the mission profile could be used to determine the fuel fraction for an initial weight estimation. After the general part, the concept design of the four chosen concepts started.

The piston-fixed wing concept was the conventional and proven concept. The preliminary configuration was a tail plane configuration with a twin-boomed tail system utilising a pusher propeller to increase the effectiveness of the horizontal stabiliser. It used a high-wing design for ease of access to the payload bay. The MTOW was estimated to be 450 kg of which 140 kg is fuel. The difference between the ferry and harmonic range is 254 nm, which is representative of its payload flexibility. The span and chord was estimated to be 8.2 and 0.74 m, respectively. These dimensions mean that the aircraft wings must be detached to fit within the LD6. The take-off and landing distances are 300 and 550 m, respectively, with a stall speed of 61 kts.

As a very innovative concept, the electrical fixed-wing concept was challenging the use of fossil fuels and the increasing emissions associated with this. Using initial calculations, it was concluded that the hydrogen fuel cell has the best energy density. The endurance of a battery would be too low, while solar cells would have a too low power output. The electrical fixed-wing concept features a conventional tail with a low-wing configuration. The aircraft would be propelled by a pushing propeller placed after the fuselage and in front of the tail. This was decided in order to have less interference with the payload, which is placed in front of the fuselage. Using reference aircraft in the same configuration, a Class I weight estimation was done, resulting in a MTOW of 417 kg. Now that the initial sizing and weight was known, other characteristics are determined in order to do a proper trade-off with the other concepts.

To be independent of runway lengths for take-off and landings, a tiltrotor concept with a twin boom tail was analysed. For this concept, the UAV was equipped with two tiltable rotors at the end of each wing. This enables vertical and conventional flight, which requires two different control systems. During vertical flight the attitude of the UAV is controlled by a swashplate installed on each rotor and tilting of the rotors. During conventional flight it uses normal control units such as an elevator, ailerons and a rudder. However, since a tiltrotor has two distinctly different ways of performing flight, it is not optimised for either flight condition. The tiltrotor is therefore less efficient both structurally and with respect to fuel use. By doing a class-I estimation based on reference tiltrotor aircraft, a MTOW of 666 kg and a fuel weight of 230 kg were estimated.

To start the concept design of the airship, the aircraft configuration was defined. For the lighter-than-air envelope, both the shape and the lifting gas had to be chosen. It was decided to use hydrogen because of its superior lifting capabilities and moderate cost. A conventional, ellipsoidal envelope shape was chosen because of its good overall performance. The power source was chosen to be a hydrogen fuel cell, which powers both the payload as well as the two main propeller engines mounted at the side of the airship. The initial design estimations yielded an aerostat length of 30 m and a diameter of 5.6 m with an OEW of 250 kg. This OEW included the engine weight, internal framework weight and skin weight. Clearly, it was not feasible to create such a massive structure that could withstand all loads in flight weighing only 250 kg. Besides the weight, the transportability of such a framework is terrible, and disassembling the structure would take a very long time for it to fit in an LD6 container. Because of these reasons, it was decided to stop the development of the airship concept at this stage and only continue with three concepts. These three concepts are depicted in figure 11.2.



a) Piston fixed-wing b) Electric fixed-wing c) Turboshaft tiltrotor

Figure 11.2 Drawings of the three evaluated concepts

The second and final trade-off process again started with defining the trade-off criteria and method. Since the concepts were designed in more detail at this stage of the project, the criteria could also be assessed and determined in more detail. The newly defined trade-off criteria were cost, size and transportability, mass, mission time, RAMS characteristics, producibility, mission diversity, operational flexibility, development risk, noise, emissions and sustainability. After the determination of the trade-off criteria, the trade-off method and the criteria weights, the trade-off was performed. The concept with the highest score in this final trade-off was the piston-powered fixed-wing aircraft which appeared to be 11% better than the electric-propelled fixed-wing concept and 86% better than the tiltrotor concept.

11.5 Final concept design

After the final concept was chosen, a detailed concept design phase started. In this phase, the final concept design was determined. A three-view drawing of this final design is shown in figure 11.3. The main system specifications are given in table 11.1.

Table 11.1. Main system specifications

Parameter	Value
Max. Take-Off Weight	484 kg
Max. Cruise Speed	204 kts
Service Ceiling	FL190
Endurance	19 hours
Range	1,700 nm

The piston-powered fixed-wing concept features a pusher propeller engine with a three-propeller, variable pitch configuration, which delivers 121 bhp to enable high cruise speeds. The use of a variable pitch propeller ensures maximum efficiency throughout the entire airspeed range. A dish antenna enables constant contact between the ground station and the UAV through a satellite communication link.

The fuselage design features a bulge at the front of the aircraft to accommodate for this dish antenna. Within line-of-sight, a camera streams live 720x1080 video to the ground station at 30 frames per second for visual flight operations and calibrating PAPI/VASI. Furthermore, the UAV is fitted with numerous antennas to receive and calibrate ILS, NDB, VOR, TACAN and DME signals, obtain navigational information and communicate with the ground station.

This enables the system to perform the flight inspection automated while also enabling ground controlled operations if needed. The landing gear features substantially large tires to enable operations at various runway surfaces. During flight, the main landing gear is stowed into pods mounted on the wings while the nose landing gear is stored in the nose of the fuselage. Finally, the wings and empennage are dismountable to enable shipment all over the globe in a standard LD6 aircraft container.

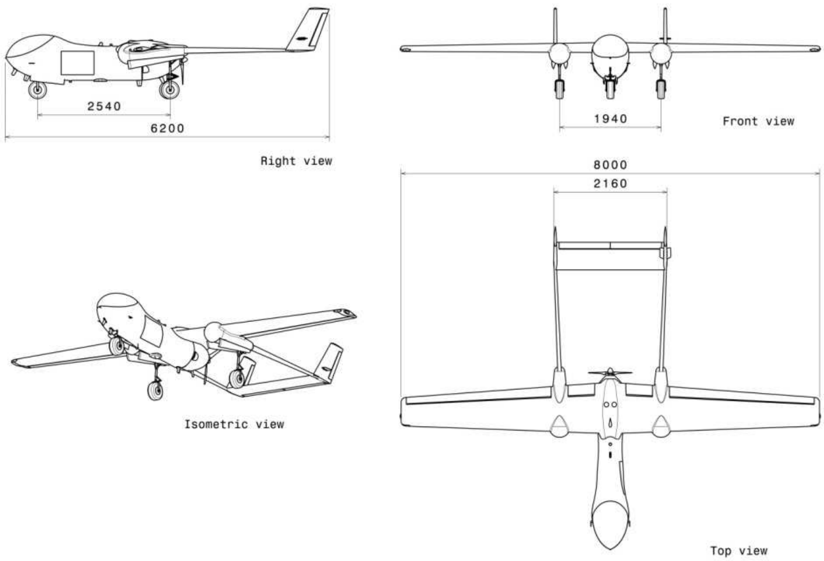


Figure 11.3. Technical drawings of the final concept

11.6 Conclusions and recommendations

The aim of this project was to investigate and design a UAV based system able to replace the aircraft that are currently used for the calibration of aerial navigation equipment. The UAV is equipped with multiple navigation receivers, as well as communication antennas to receive commands. Because of this, the system is able to calibrate aerial navigation equipment without a pilot and co-pilot and only requires a flight inspector on the ground. This significantly reduces the operational cost of the system compared to current flight inspection systems. Because the system is smaller and specifically designed for the altitude and speed at which flight inspection is performed, it is able to perform the flight inspection more efficiently, more sustainable, while creating less interference with aerodrome operations and producing less noise and while still meeting the same performance requirements.

The main requirements of the system were described by the customer. Most of these requirements have been met, however, some are not and some have been altered in consult with the customer. At this point, the crash rate of less than 10^{-7} could not be analysed quantitatively, therefore no precise crash rate has been presented and the compliance with this requirement could not be proven. Furthermore, the sound exposure level was required to be 20 dBA lower than a Cessna 550 on a 3° glideslope. Reaching this value was deemed impossible after analysing the characteristics and the final realised value of 15 dBA was accepted by the customer. The production cost of at most €250,000 could also not be met. First estimations calculated the production cost at €1,270,000, however, this seemed perfectly reasonable after further debate with the customer. Finally, the production volume of 100 units is, according to first estimations, not sufficient to compensate for the high development cost of the system; therefore, a wider market is required in order to be profitable. To achieve this, the UAV could be repurposed for other missions, which is already enabled by making the payload removable.

As can be read in the mission need statement, the main driver for the design of the UAV based system was to perform flight inspection in a more cost effective way than current systems, with special emphasis on producing less noise and causing less interference with aerodrome operations. It has been proven that the noise intensity of this system is a factor 32 lower compared to the Cessna Citation II. The interference to aerodrome operations can be reduced by decreasing the mission time of the system. The airspeed of the UAV during flight inspections is comparable to a Cessna Citation II. The better turning performance, however, enables faster transition between different required flight profiles and therefore decreases the total mission time. The Cessna Citation II has a minimum turn radius of 275 m, while the UAV at least has a radius of 110 m. This decrease significantly reduces the overall mission time for calibrating ILS systems. Finally, the operational costs of the final concept are estimated for a typical flight inspection mission consisting of one VOR beacon orbit and 15 ILS approaches from 5,000 ft. Based on this mission, the total operational cost for a Cessna Citation II was estimated to be €6,100, while the operational cost for the UAV based system is only €470.

Some recommendations on the final design have to be made. First of all, more dynamic load cases should be analysed and specifically at points of load introduction as these cause local stress concentrations. In addition, an aero elasticity analysis should be performed on the tail to determine the impact of its low stiffness on the aerodynamic performance, as large deflections impact the local angle of attack. Furthermore, a detailed design of the fuselage should be made, especially with considerations regarding composites.

To reduce the structural loads due to gusts, which are currently driving in the V-n diagram, an iteration of the aerofoil should be performed where a lower lift gradient should be opted for to reduce its sensitivity. This will be beneficial for possible high stability requirements for imaging payloads. Alternatively, gust alleviation systems may be analysed to reduce the accelerations caused by gusts and to improve fatigue life by reducing the load amplitude. Finally, because the requirements ask for a relatively small aircraft with a high cruise speed, the size and weight of the engine drive the overall design to a large extent (the engine weight is 25% of the OEW of the UAV). Researching new engine technologies for more compact and lighter engines could therefore significantly improve the overall design and performance of the system and should be looked at before further development of the system.

12. RELOAD – RELIABLE LOW COST AIRCRAFT DESIGN

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12.1 Introduction

Rather than starting anew, the current trend in the aviation industry is to redesign an existing aircraft to increase its efficiency. However, with the ongoing tightening of environmental sustainability requirements and demands for cost reduction, these designs are reaching their limit of improvement. An innovative aircraft is needed that can fulfil the future demands. This was what the project RELOAD aimed to do: to develop a REliable LOw-cost Aircraft Design for a narrow-body aircraft. The RELOAD mission statement was defined as follows:

"Design a narrow-body aircraft with a 30% reduction in direct operating cost, compared to 2015 competitors such as the Boeing 737NG and the Airbus A320, for market introduction by 2030, with a group of 10 students in 11 weeks time."

A high amount of innovation had to find its way into the final design to be able to meet the stringent requirements. Therefore radically new concepts and technologies were chosen to lower the direct operating cost and to increase sustainability. The goal of this project was to design a completely new aircraft complying with the following product mission statement:

"Perform at similar levels as current competitors, with at least a 30% reduction in direct operating costs and a 20% reduction in emissions, while at the same time maintaining equal development and production costs."

By 2035 the aircraft market will have grown considerably, especially in the new emerging areas such as Southeast Asia and Latin America. Within this increasing market, the largest demand is the narrow-body segment with 26,730 aircraft until 2035. By innovating the existing narrow-body design, the team is confident to successfully reload the market.

12.2 Requirements

As can already be concluded from the product mission statement, the RELOAD aircraft had to comply with several requirements. To make sure that it fulfils every expectation, all requirements were analysed. The driving requirements are listed below:

- The aircraft shall have a seating capacity of 175 in a typical single class configuration.
- The aircraft shall have a 6,500 km range with maximum payload.
- The direct operating cost shall not exceed 3,125 $/h.
- The return on investment shall be 5% after 5 years.
- The aircraft shall be introduced to the market by 2030.
- The aircraft shall have a 99.7% technical dispatch reliability.
- The aircraft NO_x emission shall be lower than 51.28 g/kN.
- The aircraft CO_2 emission shall be lower than 0.05472 kg/chair/km.
- The aircraft cumulative noise shall be lower than 239 EPNdB.
- The aircraft shall comply with the CS25 regulations.

12.3 Conceptual design

In order to fulfil the requirements and the market need, four concepts were generated. The first step in this approach was a brainstorm, resulting in design options on different subsystems. Several options were considered on lift and drag, propulsion, stability & control, communication and structures. Afterwards, a selection of the most feasible options were made and combined into concepts, as shown in this section.

Conventional baseline concept

The first concept was a conventional, low-risk configuration. This concept was used as a baseline to compare the performed analyses to and was very similar to current narrow-body, single aisle aircraft. Small improvements were made on the design to meet the requirement, such as a geared turbofan engine, improved winglets and morphing high lift devices. This concept was thus rather straight-forward, but had benefits in certification, reliability, routine maintenance and operations.

Blended wing body concept

The Blended Wing Body (BWB) is the second concept that was considered. This configuration is currently largely investigated for its high lift-over-drag ratio. Combining this with efficient split winglets, a twin tail and zap flaps, will reduce fuel consumption and emissions even more. Further features were the use of biofuel and the reduction of noise by the twin tail. Besides, the fuselage had a double aisle and elliptical shape to use the available space in the wing.

Three lifting surfaces concept

The third concept featured three lifting surfaces, namely: a canard, a main cantilever wing and a T-tail. A three lifting surface aircraft was chosen since high lift-over-drag ratios can be reached with this configuration. It was equipped with spiroid winglets, morphing high lift devices and a hybrid propulsion system. For this, a boundary layer ingestion turbofan engine at the back was combined with wing-mounted electric engines. The fuselage design was similar to a conventional layout.

Braced wing concept

The last concept featured a braced wing. The extra wing strut allowed for high wing aspect ratios, which increased the efficiency due to the reduction of induced drag. Furthermore the geared turbofan engines used biofuel and made use of boundary layer ingestion. Finally, plain flaps and Krueger slats were installed. The layout of the fuselage was conventional.

Concept trade-off

After the generation of the concepts, a class II analysis on their overall performance and layout was executed. A trade-off table with selected criteria and weights on their importance was developed. For several criteria, a numerical analysis was not possible, which is why a lower weight was given. The best performing concept was graded with a ten and the rest was normalized with respect to the superior concept. The results of the trade-off can be observed in table 12.1, from which the BWB was selected as most promising option.

Table 12.1: Concept trade-off table

	Weight	Conventional	BWB	Three Lifting Surfaces	Braced Wing
Fuel fraction	5	9.80	10.0	9.83	9.99
Max. take-off weight	4	9.04	9.62	10.0	9.20
Lift-over-drag ratio	5	8.48	10.0	8.67	9.87
Induced drag	2	4.47	10.0	4.78	5.49
Ease of maintenance	1	7	4	5	6
Ease of development	1	5	4	3	6
Ease of operations	1	7	4	5	7
Total		156	170	155	166

The same approach was taken on the trade-off for the engines and high lift devices, as they appeared to be interchangeable between the concepts. In this way, the BWB concept was combined with a geared turbofan, featuring boundary layer ingestion, and morphing high lift devices.

12.4 Final design

The BWB concept was refined during the final design phase. Some major changes were applied to the concept and eventually a well-balanced aircraft outline was generated. This section describes several selected design processes and results. Some technical details are illustrated in figure 12.1.

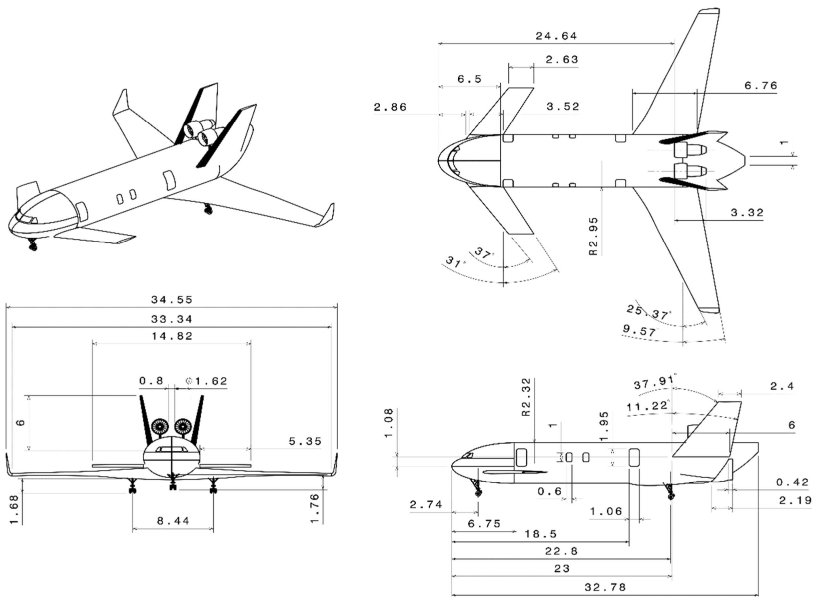

Figure 12.1: Technical drawings of the RELOAD aircraft with dimensions in m

Aerodynamic design

The RELOAD aircraft is required to cruise at a height of 11,000 m and at 0.75 Mach. The wing sizing was initially performed using handbook methods and was later refined using advanced analysis tools. The fundamental wing parameters are listed in table 12.2.

Table 12.2: Trapezoidal wing parameters

Aspect ratio	Wing span	Root chord	Mean aero-dynamic chord	Wing reference area	Taper ratio	Leading edge sweep
7	33.34 m	7.32 m	5.22 m	158.63 m^2	0.3	25.4°

An aerofoil database was analysed and two suitable aerofoils with minimum drag values at the given cruise conditions were selected. These were the "Whitcomb Integral Supercritical" aerofoil and the "Grumman K-3 Transonic" aerofoil.

Afterwards, the high lift devices were designed. In this design stage, the selected morphing slats and flaps were further researched. It appeared that morphing leading edge devices were not feasible yet due to de-icing, bird strike and scaling problems. Sealed slats and morphing flaps were thus applied, minimizing drag and noise. The sizing of these high lift devices was performed using a customized statistical method together with aerodynamic panel methods. The effect of blending the wing into the fuselage could not be investigated in detail and thus remained a recommendation.

Finally, an accurate drag estimation was performed. The cruise drag coefficient was determined to equal approximately 33.41 kN whereas the landing drag was equal to 73.06 kN.

Structural design

Structural integrity is essential to ensure a long and safe lifetime. As the RELOAD features an unconventional configuration, it was decided that both the fuselage and the wing box structure had to be investigated in more detail. As a basis for this analysis, fatigue was investigated using Miner's rule taking into account a spectrum of eleven load cases and a lifetime of 45 years with six flights per day. The canard configuration proved to be beneficial for fatigue in the fuselage since the internal loading during cruise and landing are similar.

Another challenge was the structural design of elliptic fuselage since the shape induces bending stresses in the skin next to the regular hoop stresses. By making the cross-section more circular, the maximum Von Mises stresses can be reduced by 8.5%. It was decided to using 7075-T6 aluminium alloy for both structures because of its high specific strength. This resulted in a structural fuselage weight of 5439 kg and a wing box weight of 7,135 kg.

Stability and controllability design

To provide stability and control to the BWB concept, a tail plane was required. Due to the aft location of the wing and engine, a canard configuration was selected. To ensure stability and controllability for the required centre of gravity range, the RELOAD aircraft features an augmented canard of 43.9 m^2. By including a negative gain feedback in the control system of the canard, this component reacts to perturbations in the aircraft attitude by means of automated corrections. A special feature of the canard is that it can tilt vertically upward when arriving at the terminal, to allow better access for ground operations. The vertical tails were designed afterwards, showing a surface area of 29.5 m^2 each due to the aft centre of gravity. The dynamic behaviour of the aircraft is then determined to be a set of damped and stable eigenmotions, except for the spiral mode. This spiral is only marginally unstable with a doubling time of over 6.5 minutes. Such 'unstable' behaviour can easily be corrected by the pilot or the system itself. To stabilize the Dutch roll motion, an anhedral of 1° was applied.

Furthermore, on-ground stability is provided by means of a tricycle landing gear layout. The nose and main gears are positioned at 8% and 70% of the fuselage length respectively, which ensures compliance to ground clearance and loading requirements under all circumstances.

Engine design

For the optimization of the RELOAD power plant it was decided to use a simulation program that is able to model gas turbines into detail. The software is called GSP 11 and was development by NLR. The

RELOAD engine was based on the high-bypass turbofan from CFM's LEAP-1A engine, the BIGFAN, provided by the GSP tool, and the General Electric GE-90 engine. The final result can be found in figure 12.2. This highly efficient engine has a total fuel cost of $1,237 per airborne hour, this is a total fuel costs reduction of $911 per airborne hour or 42.2%. A 20.42% reduction in direct operating costs compared to competitors is thus achieved, but also a decrease in emissions. In total, the CO_2 emissions were reduced by 39% and the NO_x-emissions even by 60.1% compared to the competitors. By placing the engines on top of the fuselage and between the vertical tails, the noise is shielded and a cumulative noise reduction of approximately 29.6 EPNdB is achieved. To make sure the aircraft can safely take-off, a maximum thrust of 71.16 kN per engine can be achieved.

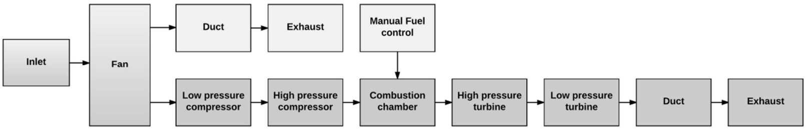

Figure 12.2: Turbofan engine model for RELOAD

Communication design

Under normal conditions RELOAD can be flown by one pilot as it has been equipped with a single pilot operations system. This system consists of two branches; an on-board system and a ground based system. The on-board component fulfils all tasks that were normally executed by the second pilot. The ground based terminal in the airline control centre serves under a live connection with the aircraft and becomes active when the captain calls it or when the ground controller decides the captain is incapacitated. By switching to a one-pilot configuration, the crew cost and thus direct operating cost were reduced by 7.6 For communication inside the aircraft, a fly-by-wireless system was implemented. The advantages of reducing cabling were considerable: not only weight was saved, also costs for development, manufacturing and maintenance were reduced. On the other hand, safety issues with interference and authentication were encountered. These were expected to be solved by 2030 by the use of encryption, a feedback loop and a single redundant cabling system.

Performance analysis

After all the sizing of the aircraft's subsystems, the performance of the complete aircraft was checked. Even with a preliminary class II weight estimation and updated weights for subsystems such as the fuselage and the wing box, the maximum take-off weight (MTOW) remained at 58687 kg, far below its requirement. With this, the aircraft is able to perform an unstable climb at a rate of 26.0 m/s and a steady climb at 29.8 m/s.

A few of the main inputs for the structural design were the load cases. One had to answer the question of when the most critical loads occurs in an aircraft's flight cycle and what the maximum speeds and heights are that it should withstand. To define just that, a flight envelope was constructed according to the CS-25 regulations as shown in figure 12.3. The maximum height was either limited by the engines or by the structural load of the fuselage due to the pressure differences.

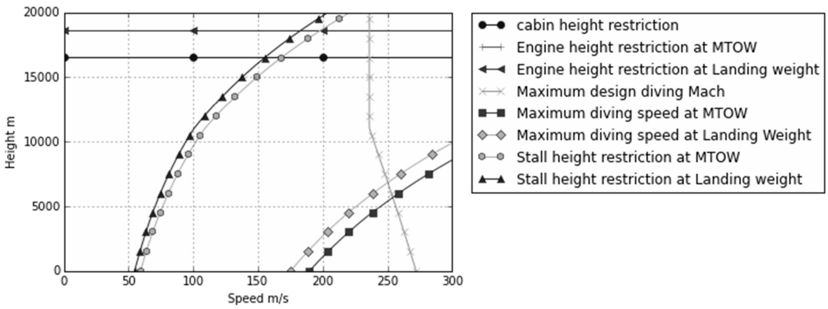

Figure 12.3 Flight envelope

The TLR-DS4 requirement, as mentioned in section 12.2, was marked as a killer requirement, as it would drive the design to an unacceptable extent with respect to cost. A dispatch reliability of 99% was determined, which is in line with reference aircraft.

For airliners to consider buying the RELOAD aircraft, it needs to differentiate itself from competitors. Figure 12.4 shows the position of RELOAD with respect to current and future competitors. RELOAD has a promising position on the overall narrow-body market, differentiating itself from other aircraft mainly on the low direct operating cost. The current range of 6,500 km is currently the only limiting factor compared to competitors.

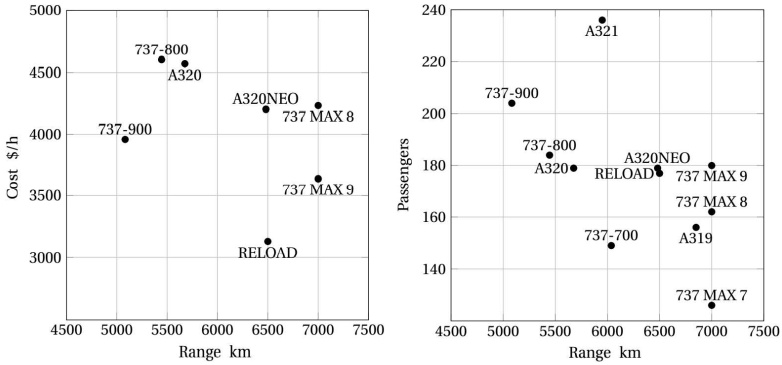

Figure 12.4: Performance of RELOAD in terms of direct operating costs and range compared to the competitors

Cost analysis

The RELOAD mission statement indicated a 30% reduction of direct operating costs. This reduction was mainly achieved by decreasing the fuel consumption and thus the fuel costs. Furthermore it was possible to significantly decrease the crew costs due to single pilot operations. The remainder reduction is achieved from the prognostic health management program and fly-by-wireless for maintenance and from depreciation. In figure 12.5 a bar chart comparing the direct operating costs of the RELOAD aircraft to its current competitors can be seen.

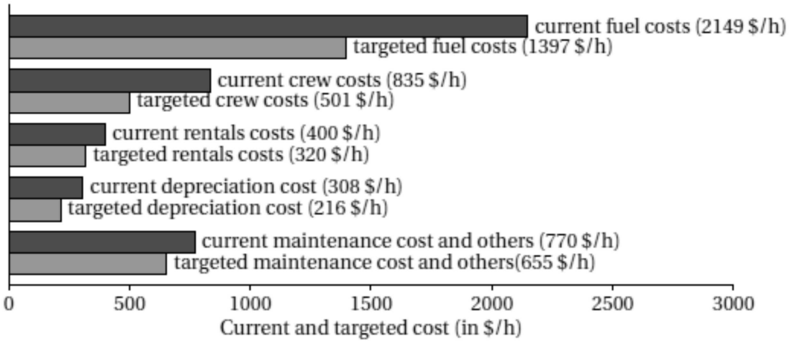

Figure 12.5: Average direct operating costs of RELOAD (targeted) and current competitors

To determine the return on investment the development and acquisition costs were calculated. A conservative calculation based on statistics was done, by taking into account extra costs for the complex morphing high lift devices and augmented canard. Assuming a manufacturing rate of 3 aircraft each month, a 5% return of investment is reached after 5 years by selling 172 aircraft for 90 M$. This would mean that RELOAD only needs to capture 2.57% of the narrow-body market in the period between 2030 and 2034 to meet this requirement. However, due to its good performance compared to its competitors as indicated in figure 12.5, a market share of at least 5% was assumed reachable.

12.5 Conclusion and recommendations

Conclusion

The RELOAD aircraft features a canard configuration, which is augmented for stability. The wing consists out of two aerofoils that vary over the span. The structures of the fuselage and wing box were designed to cope with the different load cases. Finally the engines were placed on top of the fuselage, in between the vertical stabilizers, to meet the noise reduction of 29 EPNdB. The fuel consumption reduction of 42.2% enabled RELOAD to meet their emission requirements. Furthermore it contributed, together with the one pilot operations, to the reduction of direct operating cost of 30.4%. In conclusion, all the requirements were met except TLR-DS4, which was marked as a killer requirement.

Figure 12.6: Final RELOAD design

Recommendations

At the final stage of the RELOAD project, several recommendations for further development were described. The first pertained to the size of the vertical tails. It is recommended to investigate reducing the tail surface, by placing the canard under a dihedral angle such that it also features a vertical projected surface. The second recommendation stated to research changing parts of the aluminium fuselage to composite. Composites show less fatigue problems compared to metals. Besides, a recommendation on the blending of the wing into the fuselage is made. This increases the aerodynamic efficiency dramatically, however due to operational constraints this was postponed for now. Also the maintenance of RELOAD could be further researched by the use of smart materials and an efficient maintenance program. A last recommendation pertained an improvement for the engine. Implementing a geared system and boundary layer ingestion is expected to dramatically increase the efficiency of the engine.

13. ANTI-DRONEDRONE

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13.1 Introduction

Due to the swift pace of technical innovation, boundaries are pushed in a wide variety of technical fields. Specifically, the increased energy density of batteries is proven to be a fundamental impetus driving the booming business of small and micro unmanned aerial vehicles (UAVs), mainly in terms of amateur drones. With it comes an increase in the number of drone incidents in which the vehicle is flown into a restricted area, such as the airspace around airports or government institutions. Therefore, ten students were tasked to design an anti-drone system that consists of at least one UAV that is capable of removing the majority of the existing hobby drones from a restricted area. The following mission statement was obtained:

"DSE Group S08 will design a drone capable of preventing other drones from entering a specified area, whilst doing so in a sustainable way, without entering the area itself."

13.2 Mission requirements

Based on the functional analysis and the needs of stakeholders, a set of top level requirements was established. These requirements are listed below.

- The anti-drone system shall prevent any fixed wing and rotary wing drones, whose mass is less than 2.5 kg and velocity is less than 35 m/s, from entering a restricted circular area with a diameter of 1 km.
- The anti-drone aerial vehicle shall have a horizontal velocity of at least 50 m/s.
- The anti-drone aerial vehicle shall have a rate of turn of at least 16.4 °/s.
- The anti-drone system shall have a reset time after end of engagement of less than 5 min.
- The anti-drone system shall not form a threat to manned aviation.
- The anti-drone system shall not form a threat to life forms.
- The anti-drone system shall have an End-of-Life solution.
- The anti-drone system shall have a mass of less than 10 kg per aerial vehicle.
- The anti-drone system shall have a prototype cost, including development and ground station, of no more than €50,000.
- The anti-drone aerial vehicle shall stay outside the restricted area at all times.

Other things which were taken in mind were the sustainability and legislation. In terms of sustainability, the power requirement should be met with sustainable energy sources and the waste during the entire design and production process should be minimized. In terms of legislation, the anti-drone system should comply with U.S. and Asian regulations, as these regions were most likely to have the largest anti-drone markets.

13.3 Concepts and trade-off

Once the key requirements and constraints for the anti-drone solution had been defined, six concepts were derived from the 27 concepts generated during group brainstorming sessions. Initially multicopters were considered but were traded-off early as these could not meet the speed requirement. An artist impression of the final three concepts are shown in figure 13.1.

Concept 1: HELIADES
HELIADES was a fixed wing UAV including tiltrotors placed at the wing tips. The tilting function allowed the UAV to function as a helicopter including its VTOL capabilities as well as a conventional aircraft with its higher velocities. The removal systems can include a radio frequency jammer, net launcher and kinetic cannon based on customer requirements.

Concept 2: PHAETON
PHAETON was a vertical take-off and landing drone, incorporated with an unconventional design due to its fuselage integrated rotor. Its other two propellers were integrated into the wingtips using tilting ducts, for balancing in vertical flight and increasing the airspeed during forward flight. Removal of threats was done using a net in the guppy mouth at the bottom of the fuselage.

Concept 3: TPS
By combining advantages of a helicopter and aircraft, the compound helicopter concept TPS had been derived. The drone was being propelled by both its coaxial rotor and its push propeller. This allowed for excellent agility and acceptable maximum airspeed for interception. The removal system again consisted of RF jammer and a net launcher.

(a) HELIADES (b) PHAETON (c) TPS

Figure 13.1: Conceptual designs

Trade-off

In order to decide on the final design concept, a trade-off was performed based on criteria which have a direct link to the generated requirements. The main criteria included design characteristics like mass, cost, manoeuvring, sustainability, reliability, availability, maintainability, safety and speed. Every criterion was given a weight based on their importance regarding the mission. Then a sensitivity analysis was performed which was used to verify that the trade-off is unbiased, and that only one concept was selected in the end, instead of having two or more designs with the same score.

Following the trade-off results, HELIADES was selected due to its high overall average among the criteria besides manoeuvring. Thus this concept was being used for the detailed design analysis.

13.4 Details selected concept

A detailed analysis was performed in terms of a technical, operational, risk and sustainability analysis. Furthermore, the current drone and anti-drone market was investigated, where it turned out that drones near airports could lead to potential losses up to dozens of millions of euros. Therefore, the need for anti-drone systems is inevitable.

Propulsion

HELIADES is being propelled by two tilting propeller engines. The nacelles are placed at the wing tips and consist mainly of the gearbox system of the three-bladed propellers. The tilting mechanism allows for varying of the tilt angle from 0 to 110 degrees with respect to the horizon. Therefore, HELIADES is capable of performing a vertical take-off and landing while being efficient in forward flight conditions. The two Li-Po batteries placed in the top compartment of the fuselage provide enough power for propulsion. The gearbox system and propeller are shown in figure 13.2.

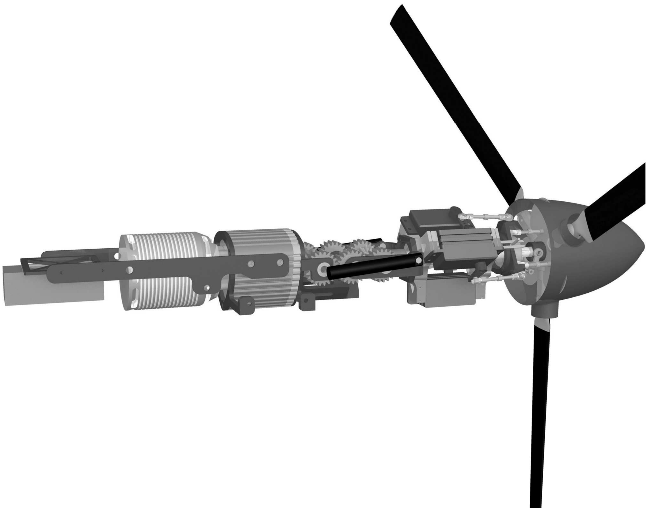

Figure 13.2: One gearbox and propeller of HELIADES

Materials

With the nacelles placed at the wing tips, the most critical condition for the structural loads was performing a 20g turn in forward flight. The respective stresses were analysed and based on the maximum wing deflections a suitable material for the wing and fuselage was picked. After a trade-off, it was decided to use Polylactic Acid (PLA), which is a bio-degradable material, produced from renewable resources such as corn starch. Moreover, this material can be 3D printed, which is favourable in terms of manufacturing. Due to issues with the temperatures near the engines extending beyond PLA's glass transition temperature, it was decided to use other more heat resistant materials in these regions.

Removal system

The removal system consisted of three main elements, all located in the bottom compartment of the fuselage: a radiofrequency jammer, a kinetic gun and a net gun. The jammer is able to jam or spoof frequencies of 2.4 GHz and 5.8 GHz, as this is the common radio frequency for radio controlled systems, as well as 1.58 GHz, the typical GPS frequency. Furthermore, the kinetic cannon is based on a Rossi Cirquit Judge and fires metal pellets. A total of 6 shells can be fired with a maximum distance of 70 m. The net gun, based on the FN 303, has 5 net shells with a maximum range of 20 m. The nets unfolded after firing to their full span of 2x2 m, in which a small drogue chute ensured a smooth landing after capturing the intruding drone. The lay-out of the removal system is shown in figure 13.3.

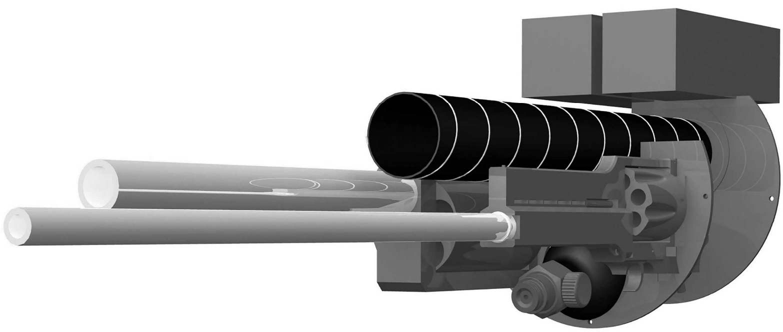

Figure 13.3: Removal systems with battery

Performance

A typical mission profile of HELIADES can be found in figure 13.4, where VTO/L, ROC and ROD represent vertical take-off and landing, rate of climb and rate of descent respectively. The profile starts with a vertical take-off and transition to aircraft mode once a certain altitude is reached, followed by a climb to the cruise altitude of 600 m to intercept the target. Once the target drone is reached, the combat stage ensues. After the drone is captured, HELIADES descents to the transition altitude. Transition is needed as HELIADES should both take-off and land vertically as the rotor blades would touch the ground in case of a conventional take-off and landing.

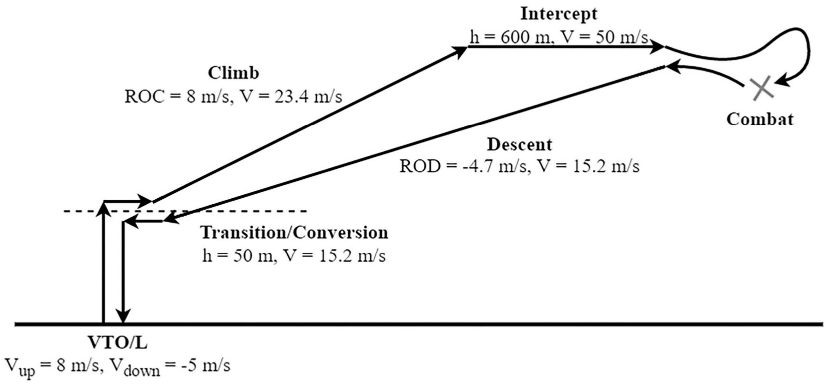

Figure 13.4: Typical mission profile of HELIADES

Operations

As HELIADES has an autonomous aerial vehicle, the remaining tasks for an operator are to swap and recharge batteries, supply ammunition and pick up eliminated drones. Furthermore, the locations of the centre of the restricted area can be easily programmed in the system using the robust laptop of the ground station.

In terms of transporting the system, HELIADES is a modular design. This means that wing, and fuselage can be disassembled and stored in a transport box. The antenna and recharging pods of the ground station can both be stored in another box and lastly the laptop has its extreme weather resistant suitcase. This can be clearly seen in figure 13.5.

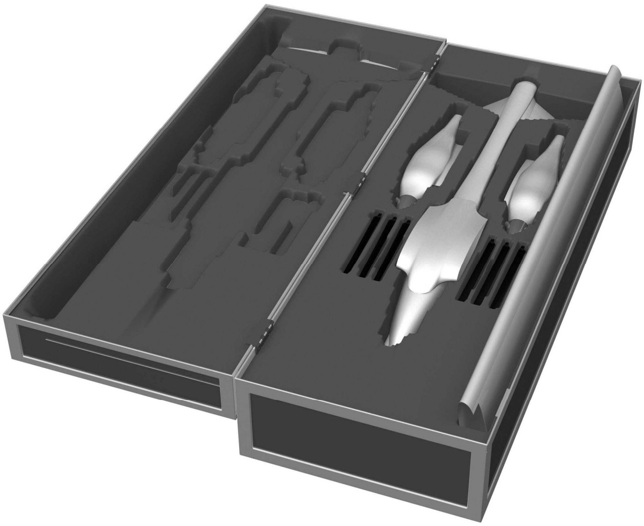

Figure 13.5: Transportation box of HELIADES' aerial vehicle

Lay-out

A visualization of HELIADES' aerial vehicle with its aerodynamic shape is shown in figure 13.6. Note the inverted V-tail for stability and control, both kinetic and net gun in the front of the fuselage and tilted rotors at each end of the wing.

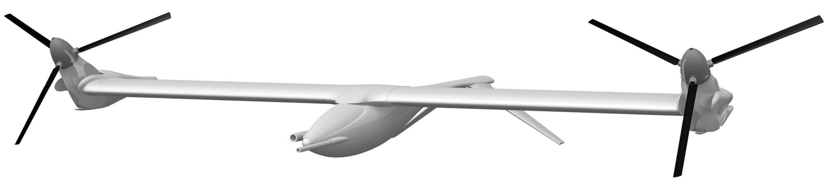

Figure 13.6: HELIADES with tilted propellers

13.5 Conclusions

It was found from the market analysis that there are ample opportunities for anti-drone systems. This was evidenced by an expected market growth rate of 23.89% between 2017 and 2022. As HELIADES operated autonomously and had a significantly greater detection range than existing solutions (5 km), it was expected to be favoured over other products. Moreover, its development cost of

€50,000 was negligible compared to the millions of euros customers, such as airports, could lose due to drone incidents. A final system cost of €200,000 was set, as this way a break even margin was included besides also ensuring remunerative sales.

Furthermore, an elaborate technical analysis was performed to ensure the system met the requirements imposed on it. Attention was paid to manoeuvrability to ensure intruding drones could be intercepted, to control to ensure complete autonomy in flight and to structural design to ensure all loads could be sustained. Regarding the former; HELIADES used tiltrotors to switch between helicopter and aircraft mode, thus allowing for efficient flight whilst omitting the need for a runway.

Moreover, autonomy was ensured through the use of a combination of mostly existing software programs for obstacle avoidance, trajectory planning and target tracking. The latter was done by using a stereoscopic camera that keeps the target in sight and determines when it is close enough to attempt removal. For the latter, a jammer, a net gun and a kinetic cannon were installed, thus providing the system with multiple options to take out the target. This made it effective against the majority of hobby drones available. Furthermore, HELIADES was built using mainly PLA, a biodegradable plastic produced from renewable resources such as corn starch. This allows for the structure to be 3D printed, resulting in a total aerial system weight of 9.84 kg.

Following the technical analysis, the detailed design took place. In this phase, operational aspects and risks were assessed. It was ensured that the system broadcasts a distress message in case of emergency, so as to ensure safety in the vicinity of the drone. Moreover, the design was such that the wings and fuselage can be detached to allow for convenient transport. Note that only one operator was required to keep the system operational. This did not have to be a professional as the operator's tasks are relatively straightforward. The reliability of the system was also analysed, yielding a probability of successful removal of 89% during the first year of operations.

Finally, a preliminary planning for future phases was set up. This identified activities, cost elements and verification and validation procedures. The main focus was on more extensive analysis and testing, so as to obtain a more detailed design that has been reliably validated.

Table 13.1: HELIADES' important design parameters

Parameter	Value	Unit
Aerial vehicle weight	9.84	kg
Ground station and transportation case weight	88.30	kg
Wingspan	1.70	m
Inverted V-tail span	0.66	m
Fuselage length	1.56	m
Maximum velocity	70.8	m/s
Stall speed (aircraft mode)	13.5	m/s
Maximum load factor	20	
Maximum rate of climb	41.3	m/s
Maximum instantaneous turn rate	185	°/s
Maximum sustained turn rate	136	°/s
Rate of successful removal	89	%
Pitch rate	60	°/s
Roll rate	150	°/s
Prototype cost excluding radar	22,200	€
Total system market price	200,000	€

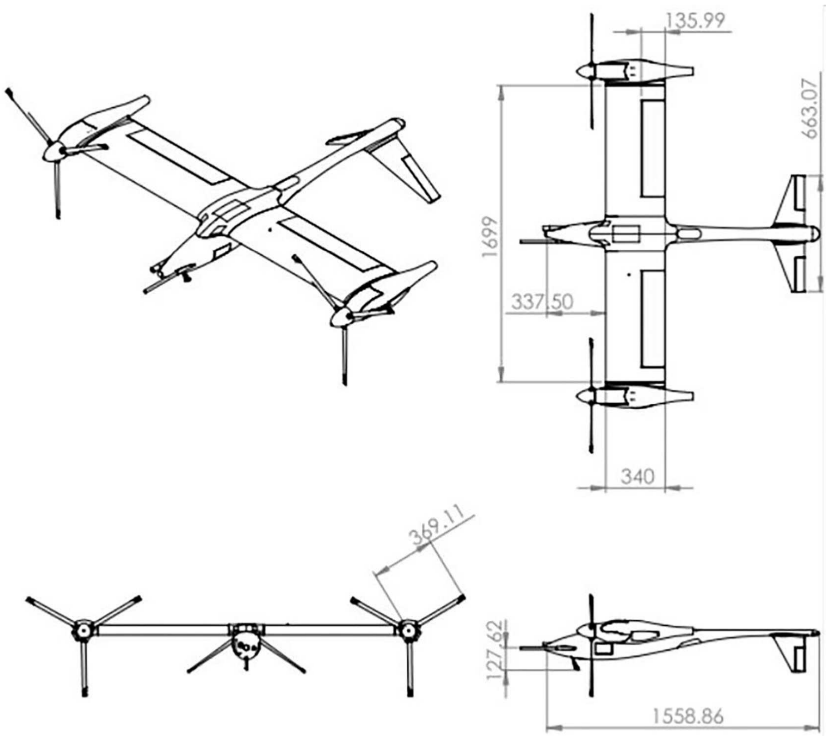

Figure 13.7: HELIADES three-view, dimensions in millimetres

13.6 Recommendations

During the preliminary design process, not all design aspects could be investigated due to them being too complex or detailed to be thoroughly analysed in the time available.

Firstly, a detailed analysis of the transition from helicopter to aircraft mode and vice versa should be conducted. A preliminary analysis had been performed, showing that the transition can be performed safely. Nonetheless, this phase can cause difficulties for stability and control due to the disparity between operation modes. Therefore, a state space model should be set up that accurately models this transition phase and the resultant change in attitude. To this end, validation tests should also be performed to ensure the transition can be safely performed in reality.

Secondly, the removal system should be tested extensively to ensure it functions correctly and provides the desired reliability. Components had been selected or designed, but should be validated both on the ground as well as in the air. The former allowed for general checking of the working of individual components, whereas the latter allowed for checking of the functioning of the combination of jammer, net gun and kinetic cannon when used on an aerial vehicle.

Thirdly, radar system options should be looked into. The preliminary design mainly focused on the aerial system and used the assumption of a functioning radar detection system. The Faculty of Electrical Engineering, Mathematics, and Computer Science at Delft University of Technology was approached to set up an experiment to test the effectiveness of existing radar systems for drone detection. Due to time constraints however, this experiment could not be performed in the allotted time. It should be performed at a later point in time, after which either an existing system should be selected or a customized system should be designed.

Moreover, validation tests should be performed for the used computational models in the preliminary design. Most of the validation procedures in this design phase were conducted using data that was not specific for HELIADES. Although these do allow for an initial verdict on the validity of aforementioned models, they were not as reliable as data obtained from concept specific tests. Examples of the latter included wing bending tests, stall tests, and flight tests. Note that more advanced computational methods, such a Computational Fluid Dynamics and Finite Element Analysis, should be used for detailed analysis. These should also be validated using aforementioned tests.

Furthermore, the developed system remained in a grey area when it comes to legality. This was due to its use of firing mechanisms and jammers, which were not allowed for commercial use. Some legislative requirements had been taken into account, but a detailed legislative analysis should be performed to ensure the drone was allowed to operate as intended in every aspect.

Finally, a marketing strategy should be defined for both sales as well as development funding. As a phase was entered in which profuse tests will be performed, investors had to be found and convinced of the potential of HELIADES to ensure sufficient resources were available for further development.

Summarizing, HELIADES was revolutionary in the sense that it is capable of autonomously performing its mission. Moreover, it outperformed existing solutions in terms of manoeuvrability and cost. Besides, its sustainability was unmatched, as carbon emissions are limited during manufacturing and operations. It was also recyclable due to its biodegradable structure. In conclusion, the group is confident that HELIADES will experience a successful market introduction due to the design's potential as well as the team's drive to deliver a sound product.

14. SCULPTUR - SURVEILLANCE CIVIL UAV LED BY PROPELLER-BASED GAS TURBINE

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14.1 Introduction

Natural resources such as oil and gas are crucial aspects of most of the world's economies. Most countries have extensive networks of pipelines to transport these resources. Pipelines decrease the freight expenses of the transport of petroleum and natural gas drastically. They are out of sight of the general public and are an environmental friendly way to transport these products. There are two main types of energy pipelines: liquid petroleum and natural gas pipelines.

Liquid petroleum pipeline networks transport a range of different liquids, including crude oils and refined products. In the case of crude oil lines, there are two types of lines: gathering lines and transmission lines. Gathering lines are very small pipelines which can be found in the areas where crude oil is found deep within the earth. On the other hand, transmission lines are larger and bring the crude oil from the producing areas to refineries.

Furthermore, refined petroleum products, such as jet and diesel fuels or heating oil, are carried through the pipelines from the refineries to terminals with storage. These refined products are then loaded into trucks that deliver the products to gas stations and homes. Natural gas lines differ from oil lines as they deliver the gas directly to homes through pipelines. In order to prevent accidents, these types of infrastructures need to be monitored regularly. Corrosion and third party damages can lead to incidents and accidents such as leaks and explosions, which have negative impacts on the environment and can generate substantial safety issues.

An example would be the pipeline explosion in Nigeria in 2005 that killed hundreds of people. Another important impact of pipeline failures is the consequence it has on national economies. The economic loss as well as energy loss is very disadvantageous. Such incidents can paralyze entire areas. Hence, it is clear that pipeline monitoring missions are vital to ensure safety for both oil and gas companies and governments.

Currently, these monitoring missions are mainly performed by manned aircraft. However, these manned missions are very expensive and present a potential risk for the safety of the crew. A Need Statement may be found due to the high risk of pipeline leaks and their catastrophic impact on the environment. The need for better surveillance arises from the ageing of current pipelines, both above the ground and buried underneath:

"Pipelines must be monitored to prevent incidents and accidents that may generate substantial safety issues for the environment."

An opportunity to solve this need is the adoption of an unmanned aerial vehicle (UAV). It would permit these missions to be carried out without human risk and to have extended endurance, leading to cheaper missions. However, current UAV technologies use reciprocating engines which cannot support the required endurance. It follows that a new design is needed. To solve this issue, a UAV propelled by a micro gas turbine engine is proposed. This can lead to a higher endurance due to its large power-to-weight ratio. Also, its low noise emissions and low vibrations will allow for better quality of observation. Hence, the aim of this project is:

"To design a micro-gas turbine propelled UAV with a payload of up to 50 kg which can have a range of 1,800 kilometres, as well as having an endurance of 20 hours and an altitude of 1,000 meters at a cruise speed of 100 km/h. The UAV will be remotely controllable at long distance. Moreover, the socio-economic benefits, the environmental footprint and the sustainability connected to the life-cycle of the UAV will be investigated."

14.2 Requirements

Requirements were the basis for the design process as they imposed constraints on possible solutions. The sources of requirements are diverse, as the requirements come from stakeholders, regulations or are simply present to ensure competitiveness with the current market. This section will describe the requirements of the design of SCULPTUR from certification regulations imposed by EASA and ICAO. The requirements are grouped into the sections performance, safety and reliability, sustainability, ground control and cost, which cover the main areas of the design process.

- The UAV shall have an endurance of 20 hours
- The UAV shall be able to carry a payload (e.g. IR and day camera, sensors, etc.) up to 50 kg
- The UAV shall be remotely controlled by a ground crew
- The UAV shall have a cruise altitude of 1,000 m
- The UAV shall have a cruise speed of 100 km/h
- The UAV shall have a maximum flight distance of 1,800 km
- The UAV shall be powered by a mini-turboprop gas turbine
- In case of engine failure, the UAV shall be capable of landing undamaged

- The UAV shall be able to relay flight data information to the ground control station
- The development cost of the UAV shall be lower than current empty weight cost per unit mass (≈3,000 €/kg)
- The aircraft structure shall be able to withstand limit loads without permanent deformations and without deformations large enough to interfere with safe operation

14.3 Concept design and trade-off

When all requirements were determined, many concepts were made and researched for their feasibility. After this conceptual research, three designs were selected as being the most feasible designs. These three designs were:
- Conventional configuration
- Tandem wing configuration
- Flying wing configuration

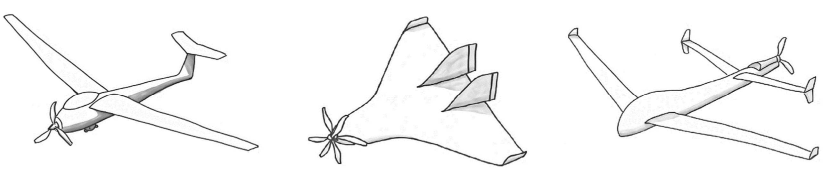

Figure 14.1: The conventional, flying wing and tandem wing configuration

Conventional configuration

The first configuration is regularly used in the current aviation industry. The stability of a mono wing with empennage design is provided through positive wing lift and negative lift generated by the stabilizer during cruise flight. The use of an empennage leads to a relatively large c.g. range in which the UAV is found to be both stable and controllable, which allows for a larger allowed shift. The conventional design will, compared to the tandem configuration, require a larger wing span and a tail, which increases the structural weight of the aircraft. The presence of one wing instead of two decreases the profile drag and having only one integration point decreases the structural complexity of the design. The drag performance of the conventional design is good but the interference of the wing air flow with the tail decreases efficiency and increases drag.

Tandem configuration

The second configuration that was considered was the tandem configuration, which uses two wings of equal size to provide sufficient lift without the need of adding an horizontal tail surface. The absence of a horizontal tail and the addition of an extra wing increases the aircraft lift and decreases the induced drag. Another advantage is that the added wing results in a reduction of the wing's aspect ratio, reducing structural weight. However, the extra wing increases the profile drag. Furthermore, the airflow over the aft wing is disturbed by the flow of the front wing, which can negatively influence the lift and drag production. This disadvantage makes the placement of both wings an important design parameter.

Flying wing configuration

The last configuration considered was the flying wing design in which the "fuselage" is integrated in the wing. The main advantage of this streamlined design is the usage of the complete aircraft to generate drag, resulting in a reduced parasitic and profile drag. The flying wing also allows for a lower structural weight compared to the tandem and conventional configuration, as the lift loads are carried over the whole surface area of the UAV. Problems were found when investigating the stability of this design, as flying wings are inherently unstable on low airspeeds, allowing only a very limited cg-range to operate in.

Design choice

After conducting a detailed research into the three concepts and making a trade-off, the conventional configuration was chosen for the preliminary design phase. The flying wing was the least suitable design due to the difficulties in stabilizing this design. The tandem and conventional configuration were both found to be suitable to complete the mission objective. However due to the limited resources and the high development risk of the tandem configuration, the conventional design was chosen for the following design phase.

14.4 Design, analysis and results

When the concept for the SCULPTUR UAV was defined, the design and analysis phase could begin.

Weight estimation

For the preliminary design of the UAV, Class I and Class II weight estimation methods were used. To use these methods, an optimum design point was determined based on the power loading and wing loading diagrams. Secondly, the Operating Empty Weight (OEW) and Maximum Take-Off Weight (MTOW) were estimated based on statistical analysis of different UAVs with similar range, endurance and propulsion systems. The resulting values were used as a starting point of iteration. Using the subsystem design changes after analysis, the OEW and MTOW were further optimised and iterated.

Wing design

The design of the wing consisted of two major sections: the planform area and the aerofoil. In order to increase the efficiency of the SCULPTUR UAV, the aerofoil was optimized for minimal drag during cruise. The wing planform was optimised for endurance and range, since high endurance and range were the driving requirements for the SCULPTUR UAV. A high endurance and range, together with a low required speed resulted in a wing with a high aspect ratio as the most feasible option. Table 14.1 shows all further wing planform characteristics.

Table 14.1: Wing planform characteristics

Parameter	Value	Unit
Wingspan	12.50	m
Root chord	1.19	m
Aspect ratio	15	-
Taper ratio	0.4	-
Sweep angle	0.0	deg
Dihedral angle	5.0	deg
Incidence angle	3.56	deg

Empennage design

An inverted Y-tail was chosen for the empennage. This tail configuration has a high longitudinal stability performance and very good Dutch roll damping. It also has a good stall performance and due to the anhedral angle of the diagonal surface, the engine exhaust is not influencing the control. Another characteristic of the inverted Y-tail is an increase in the effectiveness of the control surfaces. This increased effectivity is very beneficial, since large control surfaces are required due to low speeds, and consequently dynamic pressure, at landing and take-off.

Propulsion system design

Before the propulsion system design was initiated, an exploration study into reference turboprop engines was conducted. Due to the limited amount of Micro-Gas Turbines (MGTs) in the power range of 20-50 kW, it was decided to perform a preliminary design of a new turboprop MGT, developed precisely for the surveillance mission in Australia. This decision was taken based on the strict requirements set for the design of the UAV. They were obtained from the performance design calculations and ICAO restrictions for maximum fuel flow and harmful exhaust emissions (see table 14.2)

Table 14.2: Engine requirements and results

Description	Required values	Final values	Unit
Thrust take-off	>>1250	1447	N
Thrust cruise	>738	739	N
Shaft power take-off	>21.03	24.61	kW
Shaft power cruise	>7.6	8.4	kW
SFC take-off	<514	493.0	g/kWh
SFC cruise	<572	537.0	g/kWh
Fuel flow take-off	<9.4	3.4	g/s
Fuel flow cruise	<1.25	1.2	g/s
NO_x exhaust emissions	<0.040	0.038	m%

The table shows that the design parameters were only applicable to two performance settings: the maximum take-off and the normal cruise conditions. The former is found to be critical for power output and emissions, while the latter was determinant for the thrust generation and specific fuel consumption restrictions.

For the design of the engine, the Gas turbine Simulation Program (GSP) was used to iterate until the design point has been reached. The final configuration is then run with the Steady-State series simulation feature. Firstly, it was found that to be able to obtain the optimal maximum pressure ratio, a two-stage radial compressor would be implemented with a connection via a diffusion channel and a U-turn channel between the separate stages. Secondly, a lean-burning annular reverse-flow combustor was chosen because of its excellent combustion performance with low flame residence times (given the small size of the engine) and the fact that lean-burning minimizes NO_x and CO_2 emissions.

Moving on to the turbine stages, it was found that because of the high total pressure ratio relative to similar-size MGTs, a high expansion ratio also occurs in the turbine stages. For this reason, two single-stage radial turbomachinery were incorporated on two different shafts; a high-pressure turbine was chosen to drive the compressor stages, while a power turbine was solely developed to drive the propeller via a reduction gear box of maximum gear ratio 48:1. Regarding fuels, two soybean biodiesel blends of 80% and 50% petroleum diesel (B20 & B50), as well as the Hydrogeneration-Derived Renewable Diesel type I (HDRD I) were considered. Because of the high Lower Heating Value, low density, excellent lubricity & viscosity properties, HDRD I was chosen.

After determining the overall configuration and running the simulation for both take-off and cruise conditions, the propeller's type and size was determined, mainly based on the required thrust it needed to generate. It was found that a fixed-pitch, three-bladed propeller having a nominal diameter of 1.91 m would be optimal to meet all thrust, power and torque requirements. It was also found that the designed engine would produce approximately 24.61 kW of maximum power at take-off and 8.4 kW at cruise conditions, 10% more than required to account for elevated altitude effects.

The thrust generation was found to be 1447 N at take-off and 739 N during cruise. Finally, the CO_2 emissions were found to be approximately 35% lower than the required maximum, while the results for NO_x were kept at the upper limit – close to 0.04% of the overall gas composition at the exhaust.

Payload

There were different payloads necessary for the mission: surveillance, communication and emergency payload. For the surveillance, a gimbal turret was chosen with the possible Electro-Optical and Infra-Red camera options that accompany it. The communication should provide a downlink for the flight data information and the surveillance data, and an uplink for the flight controls. The emergency payload should ensure a safe landing. A ballistic parachute was installed to ensure an undamaged landing in case of engine failure. Furthermore, a flight data recorder was required for the emergency payload. An overview of the chosen payload is shown in table 14.3.

Table 14.3: Payload for the UAV

Payload Type	Chosen Model	Mass in kg	Power in W	Size in cm
Gimbal	CM202	8	55	29.5 Diameter = 18.8
EO Camera	Hitachi DISC120R	0.26	11VDC	5 x 6 x 9
IR Camera	Tau 2 640	0.15	5VDC	4.4 x 4.4 x 3
Communication Unit	Gilat BlackRay 1000	13	350	25.4 x 46 x 46
Ballistic Parachute	BRS-6-1050-CAN	22.6	-	55 Diameter = 18
Flight Data Recorder	SLICEMicro	0.028	0.84	4.2 x 4.2 x 8

14.5 Conclusion and recommendations

The aim of this report was to design a micro-gas turbine propelled UAV with a payload of up to 50 kg. As well as having a range of 1800 km and an endurance of 20 hours at a cruise speed of 100 km/h, the UAV will be remotely controllable at long distance. Followed by an investigation on the socio-economic benefits, the environmental footprint and the sustainability connected to the life-cycle of the UAV. The final report introduced the conceptual design phase, focusing on the main components of the various subsystems. With a final MTOW of 403.5 kg and an estimated selling price of only $3,341,000, it is believed that SCULPTUR is an attractive product for all oil and natural gas companies in Australia. As time passes, it could be available for sale also in different countries, as certifications for UAVs develop further.

To conclude, SCULPTUR complies with all the stakeholder requirements as well as with almost all the system requirements set by the design team. It is the first UAV of its kind, being the only one capable of long-range pipeline surveillance. With the conceptual design phase now finished, it is time to move on to the next step: the detailed design. Some recommendations can be given on a subsystem level.

For the structures and materials subgroup, a detailed FEM model can be constructed which would allow for an accurate representation of the loads the UAV has to cope with. For the aerodynamics subgroup, a computational flight dynamics (CFD) analysis and wind tunnel testing are mandatory while the propulsion subgroup should perform further investigation of the engine's characteristics in GSP (normal take-off, climbing and landing conditions). It is also highly recommended to investigate if it would be possible to fly at higher altitudes and still be able to perform the surveillance mission with the available payload.

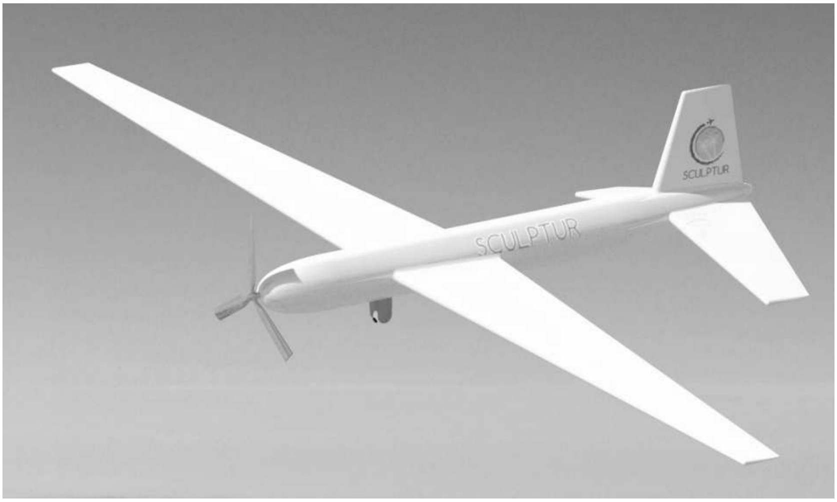

Figure 14.2: SCULPTUR

15. MIRU: MULTI-PURPOSE IMAGING AND RESEARCH UAS

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15.1 Introduction

Air safety investigators have the primary purpose to determine the cause of the accident, in order to improve the safety of aviation. This means the investigators have to document every single piece of information, from the way parts are deformed to the location it was found at the accident site. Not only do they work on the ground, but they also use aerial photography to get a complete overview from another perspective. Unfortunately, aircraft crashes tend to happen in remote areas, which makes locating the wreckage difficult, time consuming and costly.

Currently, investigators are limited to ground based imagery and single purpose aerial solutions to aid in an investigation. MIRU, the Multipurpose Imaging and Research Unmanned aerial system, bridges the gap between the investigator on the ground and the aircraft and satellites at higher altitude. Imagery from ground level does not provide the same overview as an aerial photograph would, while conventional possibilities for aerial photography do not provide the level of detail needed for an investigation. MIRU's aid in locating wreckage and documenting an accident site greatly improves the efficiency of an air safety investigation, even saving potential survivors if a wreckage is located faster. At the same time, MIRU is readily applicable to accidents at airports as well, where documenting the crash site is sped up. This allows regular airports to presume operations faster, and thus resulting in less turnover loss.

15.2 Mission objectives and requirements

The mission objective of MIRU is:

"to aid air safety investigators to perform a faster and lower cost investigation. MIRU is able to aid the investigators both while they are on-site, or at a basecamp close to the accident site. MIRU aids an investigator in both locating and documenting an air accident site."

In order to perform the mission objective, MIRU had to fulfil a number of requirements. Primary aim was to ease the operation of the investigators, such that he is able to focus on his normal duties and perform his investigation as good as possible. The key requirements of MIRU were:

- The design shall fit in a 60 litre traveller's backpack
- The design shall provide a high detail 3D map, to recognise objects of at least 2 cm
- The system shall be able to act as a communication relay
- The system shall be able to locate the main wreckage within an area of at least 1 x 1 km
- The design shall be able to detect toxic gasses, fuel and oil vapours
- The system shall be able to operate autonomously

The requirements listed above are all achieved while the investigator is on site, therefore the system performs a so called on-site mission.

When an investigator has not arrived at the accident site yet, the system performs its remote mission, for which only the last three requirements hold. In a remote mission, the system is able to start out from a basecamp which is located at most 15 km away from the actual accident site. In the meantime, the investigator can proceed to the accident site or concentrate on his other duties.

15.3 Conceptual design process

The conceptual design process was comprised of two phases. In the first phase, various UAV concepts were traded off based on the mission types. This resulted in a number of feasible concepts, out of which the decision was made to combine them to a single reconfigurable UAV.

Trade-offs: UAV concepts

For the UAV platform several concepts were considered, which were: a fixed wing with tail, flying wing, multirotor, tail sitter, tilt rotor, and tilt wing concept. These concepts were all evaluated on several criteria. A first value or estimate was found for each criteria based on reference UAVs or simple calculations. The relative results of this process can be found in table 15.1. In this table, the '+' indicates a good performance of the concept for the criteria, a '-' indicates a low performance, and a '0' shows that the design has no benefit or disadvantage for the criteria.

Two trade-offs were performed to make sure the best configuration was obtained for both missions. As the remote and on site missions have very different requirements, the weights used for the trade criteria differ largely depending on the mission. This way, an optimal configuration is obtained for each mission type.

Remote mission trade-off

Due to the importance of range and endurance it was concluded that a flying wing or fixed-wing with tail was the best option for the remote mission, since the flying with a wing is the most efficient for range and endurance. The advantage above the other configurations is also that there is no need for extra moving parts for transition, like with the tilt wing or tilt rotor, which adds extra mechanisms and therefore weight, thus decreases performance. Therefore, the best UAV for a remote mission to fit in a backpack was a flying wing since this has no tail to be stored. However, this configuration is limited by its take-off capabilities, which is solved by using a tail sitter. This is therefore the second option to use for a remote mission.

On site mission trade-off

A multirotor was the best option for the on-site mission as it is able to hover and fly at low velocities, which allows for very high quality images. Moreover, it has a lower weight than the others as it does not need to carry its wings along, resulting in a longer endurance. It also has a high manoeuvrability as it can control the rotors separately and if it has an even number of rotors, generally four, it is very stable. The other on-site mission aspect is the communication relay, which can be achieved the best by using a flying wing or tail sitter. Thus, a quadcopter was found most efficient for an on-site mapping mission, and a flying wing or tail sitter for a long endurance on-site mission.

Table 15.1: Trade off summary for the different concepts

	Mass	Range	Endurance	Manoeuvrability	Size	Complexity
Fixed wing with tail	0	+	+	0	-	+
Flying wing	0	+	0	0	0	+
Multirotor	+	-	-	0	+	0
Tail sitter	-	+	0	+	0	0
Tilt rotor	-	0	0	+	0	-
Tilt wing	-	0	0	+	0	-

As can be seen from both trade-offs, there is an optimum for each of the separate mission trade-offs. However, when both missions need to be performed by one UAV, a problem arises. For this, another trade-off was devised to obtain a final concept applicable to both missions. The feasibility of using a single reconfigurable UAV or a system of multiple different UAVs was analysed with this trade-off. This was done in order to asses which method will achieve the requirements and missions the best.

Final operational trade-off

To comply with the mission needs, it was possible to either design a system of UAVs that suits all different mission needs, or design a single UAV that is reconfigurable. Designing a system of UAVs would result in a design that is already commercially available, meaning competition is high when expanding the product to other markets. Also, the users have to familiarise themselves with three different systems and has to bring multiple UAVs to the mission site if it is not known what type of mission has to be performed.

Designing a single reconfigurable UAV allows the users of the system to familiarise themselves with only one system. Also, the UAV is easier to be brought in a backpack, as this single UAV comprises all configuration and provide flexibility in the missions that need to be performed. It was concluded that the investigator will have a lot of flexibility when using a single reconfigurable UAV, as air safety investigators often faced with unexpected situations at the accident site.

Based on these findings, final platform chosen was a UAV that can be reconfigured from the primary tail sitter configuration, to a flying wing or quadcopter (see also figure 15.1).

Reconfigurable payload

Of course payload also needs to be adapted to the different missions that MIRU performs. Not all payload options are always needed, and therefore the most efficient system was obtained by deciding to use reconfigurable packages for the payload. This can either be a mapping device or IR camera, with the possibility to combine this with a toxin detector.

There are three payload fairings, one for each UAV configuration. Within these fairings, the different payload configurations are installed.

15.4 Detailed design

The final design is a modular tail sitter, of which the fins can be replaced by small fins to have a flying wing. When the wings of the tail sitter are taken off, a quadcopter is formed. The UAV in its tail sitter configurations weighs 2.44 kg in its heaviest configuration and has a span of 1.3 m. It performs the mission completely autonomously. The design was driven by the requirement to fit in a backpack such that investigators can carry it easily to the accident site. The modularity of MIRU makes it perfectly adaptable to the various tasks. The three configurations of MIRU are found in figure 15.1.

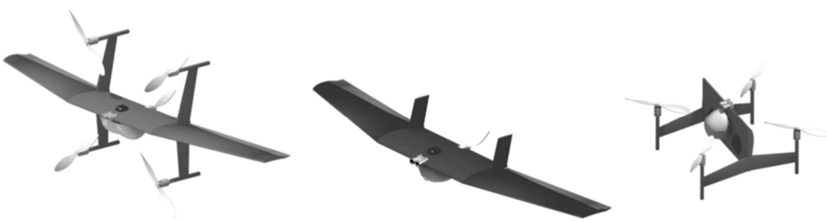

Figure 15.1: The three configurations of MIRU (f.l.t.r.): a tail sitter, flying wing, and quadcopter

Mission profiles

The main mission objective of MIRU is to locate and map the air accident site. The mission has been split in two main missions, each split up into two sub missions. The first mission is the remote mission, where it locates the wreckage, makes a coarse map of it and occasionally performs toxin detection. The second mission is the on-site mission, where it makes a detailed map of the accident site or acts at a communication relay.

- Remote mission: coarse mapping (long range)
- Remote mission: coarse mapping and toxins detection (long range and hover)
- On-site mission: detailed mapping (short range, slow speed or hover)
- On-site mission: communication relay (long endurance)

System characteristics

MIRU is primarily characterised by its modularity, which enables the investigators to use a single system for all four missions, both remote and on site. For all three configurations, MIRU is able to operate up to an altitude of 4500 m and within a temperature range of -20 ° to 45 ° C. The main features of each configuration are explained below, and are summarised in table 15.2.

The tail sitter's primary features are vertical take-off and landing, while at the same time having a long range and endurance, thus allowing to map large areas. Because of the hover capabilities, the tail sitter can also detect toxic gasses, which allows the investigators to check whether it is safe to proceed to the accident site. The compromise of this design however limits the performance, however, for this precise reason the flying wing and quadcopter configurations were devised.

The flying wing configuration is characterized by a long range, thus being capable to map large areas. Because of its long endurance, the flying wing is also a perfect solution to act as a communication relay. The flying wing configuration is obtained by replacing the tail sitter's vertical fins by smaller fins, which provide the right amount of directional stability. Since no vertical take-off and landing can be performed in this configuration, the flying wing should take off using a hand launch, and is able to make a safe and autonomous belly landing.

The last configuration, the quadcopter, is perfectly usable for high detail mapping. The quadcopter is able to hover and fly at low speeds, therefore no image blur will occur when mapping at high detail, with 2 cm accuracy. The downsides of the hovering capabilities are however a low range and endurance, thus the investigator has to replace the batteries if a detailed map of a large area is desired.

Table 15.2: Features of MIRU's different configurations

Parameter	Tail sitter	Flying Wing	Quadcopter
Maximum weight (kg)	2.4	1.7	2.0
Span or width (m)	1.3	1.3	0.42
Range (km)	74.5	114	7.50
Endurance (min)	77	122	19
Maximum mapping area (km^2)	7.3	10.5	0.034
Take-off and landing	Vertical	Hand launch and belly landing	Vertical

Payload configuration

The payload needs to contain devices that are able to map the air accident site for the different mission profiles. For these different mission profiles, different specific payload configurations are needed. The payload contains a visible light camera, an IR camera and a multispectral camera. Using this payload, MIRU is able to locate and map an air accident site during daytime, in the absence of sunlight and when the wreckage is camouflaged by its surroundings. These cameras must be able to obtain images with sufficient resolution for identification and recognition. The UAV also carries an optional toxin sensor in order to perform toxin detection.

In order to perform the mission objectives in the broad range of environmental conditions, the payload contains either the visible light camera (Mapir Survey 2), the IR camera (FLIR Vue Pro R), or the multispectral camera (Tetracam ADC Snap). These cameras are chosen for their high ground resolution performances and compactness. These cameras can be carried on board in tandem with the MicroRAE toxins sensor, in case MIRU has to perform a toxins detection. Detailed mapping is always performed at a height of 15 m from the ground, here the visible light camera and IR camera are used. At these heights, the visible light camera obtains a resolution of 140 px/m and the IR camera 80 px/m, which can provide detailed 3D maps. Coarse mapping is performed at a height of around 100 meters from the ground, dependent on the camera. At these heights the UAV maps the area at the optimum cruise speeds given for the specific payload configuration.

Aerodynamics and stability

The aerodynamic characteristics of MIRU were analysed for the flying wing and tail sitter configuration, which were used to make good calculations of the performance. For proper aerodynamic efficiency and stability, the UAV was split up into three main components, being the body, wings and vertical stabilizers, also known as fins. The body is independent of the configuration, however the fins change in size to obtain a flying wing, and the wings are detachable to obtain a quadcopter. By splitting up the system in these main components, transportation is made easy as well.

The body and wings both have to fulfil different tasks, therefore different requirements were set on the aerofoil selection for the wings and body respectively. The body allows the storage of the various subsystems, such as the Guidance, Navigation and Control (GNC) module and the batteries. The main requirement of the wing is to provide sufficient lift and provide a stable flying condition. Based on these requirements, it was quickly noticed that no single aerofoil profile could meet the necessities for both the wings and body at once. Therefore, a normal cambered aerofoil was chosen for the body to allow good storage of the subsystems, while at the same time providing lift. For the wings, a reflexed aerofoil profile was used to obtain the desired stability and lift. Afterwards, the longitudinal stability of the wing was further enhanced by applying sweep.

Lateral stability is obtained by the small vertical fins. These were sized in such a way that no image blur occurs due to the dynamic motions in the event of a perturbation. The small fins are only used for the flying wing, but for the tail sitter, engines are mounted at the end of larger fins, to allow the clearance of the rotors needed for hovering.

Propulsion

The propulsion was divided into two separate disciplines, one to accompany the hovering performance, and one to accompany the forward flight performance.

Because of the large altitude range, the performance for hovering in the quadcopter and tail sitter configuration cannot be ensured if only one type of propeller was chosen. Therefore, the hovering performance was increased by using three types of propellers for hovering, with a different pitch. One set was chosen for an altitude range of 0 – 2,500 m, the second set, with a slightly larger propeller pitch, was chosen for an altitude range of 2,500 – 3,500 m. The last set is only used when an investigation is done in very remote and mountainous areas, ranging from 3,500 to 4,500 m altitude. When the tail sitter is flying horizontally, the hovering propellers are folded such that drag introduced by the hovering propellers is limited.

For forward flight, only one type of propellers was chosen for the total altitude range from 0 – 4,500 m. The forward propeller was attached to the engine via a gearbox, such that both the propeller and electrical engine operate at their optimal rotational speed.

Ground station and communication

As MIRU is a UAS, next to the UAV a ground station is necessary. The ground station's main function is to facilitate the operations of the mission and to present the data the UAV has sent. This includes a broad range of components; from tablets to present the data received by the UAV to a laptop to process the data gathered by the UAV.

To receive real time imaging taken by the payload and to make this available to the investigation team while the UAV is performing its mission, communication is necessary. There is communication possible with the UAV over a distance of at most 15 km to cover the range necessary for the remote mission. On the UAV, there are three antennas placed: two for short range communications and one for long range transmission. They are all omnidirectional as to omit the use of a heavy antenna tracker. The ground station is equipped with four antennas of which two are used for short range communications and two for long range communications. The long range communication system is directional and thus has to be pointed towards the UAV.

There are two types of ground stations: one portable and one fixed at the basecamp. The basecamp ground station is connected to the long- and short range communication system while the portable ground station consist of a tablet with short range communication. Using the tablet, the investigator can make use of the communication relay. There is one commander tablet which can send area of interest inputs to the UAV to prevent input clashes between the users.

When there is no line-of-sight due to objects between the basecamp and the area of interest, there is no connection possible with the UAV from the ground station.

The only consequence for the UAS is that the UAV does not send any data to the ground station and the user has to wait until the UAV is back at the basecamp. However, as the UAV is completely autonomous in flight and since all the data gathered by the UAV are stored on the internal memory, this does not limit the operations of MIRU.

For the on-site mission, it is advantageous to all investigators to be able to communicate with each other, while being independent of the communication infrastructure present at the accident site as this quality might not be sufficient or the network not reliable or available. The communication relay can be both used for speech and instant messaging including images and does not affect the telemetry and the images sent to the ground station. Investigators within a radius of 1.5 km can make use of the communication relay and a maximum of up to 10 investigators can use the relay for speech.

The user and UAV are connected through the interface of a tablet. The software on the tablets is initialised using an application. Through the application, the user is guided through checklists before and during the UAV performing its mission and can communicate with other investigators using the communication link. Precautionary warnings will pop-up to remind the user of actions to be taken necessary for safe operations. Also maintenance instructions can be accessed to limit the knowledge and technique necessary by the user.

Operations

The UAV has been sized to fit in a travellers backpack. In order to not damage the components during transportation, they are placed in casings. The UAV and ground station components necessary for an on-site mission fit in one backpack and weighs only 10.5 kg in total. All components necessary for the remote mission are sized such that they fit in small transportation boxes which fit in a vehicle such as a car. This makes MIRU easily transportable to any accident site and thus deployable from any site.

Figure 15.2: Tail sitter showing its main components and assembly points

Assembly

The assembly of the UAV is made as simple as possible, for quick and stress-free deployment. As discussed before, the UAV consists of components, separated at locations that benefit the modularity. For the tail sitter, the user has to assemble the body, wings, fins, propellers, payload and the respective payload fairing of the configuration. The main components of this assembly are found in figure 15.2. For the flying wing, instead of the fins, small fins are mounted. For the quadcopter, the wings are detached. Geometric figures indicate the attachment points of different components, as illustrated for the left wing-body attachment in figure 15.3.

In this figure, the geometric figure is a square. The wings are secured to the body using two bolts for each connector from the bottom of the body. Therefore each wing is secured using a total of four bolts. The fins both contain a triangle, as well as on the top of the location of the body where these fins must be connected. They are secured using a single bolt, which is screwed in the internal thread at the back of the male connector. This bolt therefore prevents the fin from sliding backwards. Furthermore, as explained before, there are three sets of four vertical flight propellers available. The propellers are separated using a number of stripes, to show the propeller type.

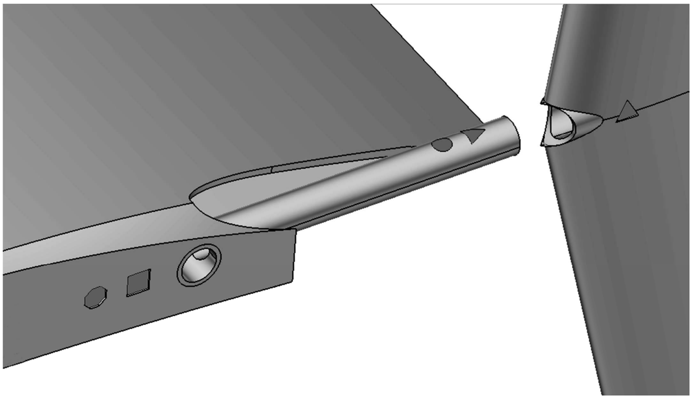

Figure 15.3: Detail of the connector of the vertical stabilizers with the body

15.5 Conclusion and recommendations

MIRU has two main missions to fulfil, namely the remote mission, and the on-site mission, with the primary objective to aid air safety investigators in a faster and lower cost air safety investigation. With the remote mission, MIRU is able to locate a wreckage, create a coarse map for the aerial overview investigators need, and has the capability to perform toxic gas detection. On-site, MIRU can make a detailed map of the accident site. The modularity of MIRU results in a flexible application to any accident site.

The design has been proven to be feasible by preliminary aerodynamic, structural and performance analyses and through extensive trade-offs. Furthermore, from a user standpoint, the design is feasible due to its flexibility and ease of assembly and operation. Next to that, the ease of carrying the system to an on-site mission in a backpack improves the quality of the use of the system.

It is recommended to make the design more weatherproof to even further broaden the operational range. One of the possibilities is to include a temperature management system to be able to fly in the harshest environments. Furthermore, the UAV can also be made completely watertight to withstand submersion in water as well, while currently, the system is only splash proof.

Furthermore, the UAV currently does not have any lights. This could be implemented such that ground teams and other airspace users can easily spot the UAV.

Lastly, to further improve the design for its ease in assembly, instead of regular bolts, quick release lock pins are recommended to use. This would be a great improvement on the user side of the design, as it will cut down the deployment time assuring an even faster response. However, these joints will increase the weight, and therefore this option needs to be further researched in its entirety.

16. Ultra-long Range Business Jet

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16.1 Introduction

As global wealth continues to increase, air travel is becoming available to a larger part of the world's population. While most people travel with commercial carriers, the business jet market is rapidly expanding as well. For companies, business travellers and wealthy individuals, business jets are often favourable in terms of travel time, availability and comfort compared to traditional airliners. In 2014 the global business jet market grew by 6.5% with a total of 722 new deliveries.

Among these, especially long-range business jet indicated an obvious growth in the market share. However, modern business jets have yet to demonstrate the same range capabilities as their long range commercial competitors. Currently, the longest range for a business aircraft in operation is 6,500 nm for the Gulfstream G650ER. New designs from Bombardier's Global 7000 (7,300 nm) and 8000 (7,900 nm) are expected to enter into service in 2017 or 2018. Clearly there is gap in range capability between regular business jets and larger commercial counterparts.

This project responds to the need for an ultra-long range business jet with the Starling 9000 corporate jet that is capable of flying farther than any current business jet in the same large size category. With its record setting 15,700 km of range and capacity for 18 passengers, the Starling 9000 can deliver state-of-the-art technology and outstanding passenger comfort. At a base price of $60M it outperforms foreseeable competition in both acquisition affordability and cost per seat mile. The Starling 9000 is ready for the market in the end of 2020.

16.2 Mission objective and requirements

The mission objective is:

"to design a business jet to travel at least 8,500 nm for 18 passengers, in a comfortable business class seating layout, while cruising at high speed."

The requirements are identified based on the mission objective. The key design requirements of the Starling 9000 are a combination of the stakeholder's demands and regulations. The requirements are listed below:

- Employ a range of at least 8,500 nm
- Carry a total of 18 passengers in regular business class seating configuration
- Maintain a cruising speed of Mach 0.8 or higher
- Be able to take off in less than 2,000 m
- Total cost of 60 million Dollars when producing in series of 600 aircraft
- Comply with the latest safety and environmental regulations

16.3 Concepts and trade-off

Four different concepts were closely examined to find the optimal design for achieving the requirements, as shown in figure 16.1. The final trade-off between these concepts mainly concerns the exterior layout.

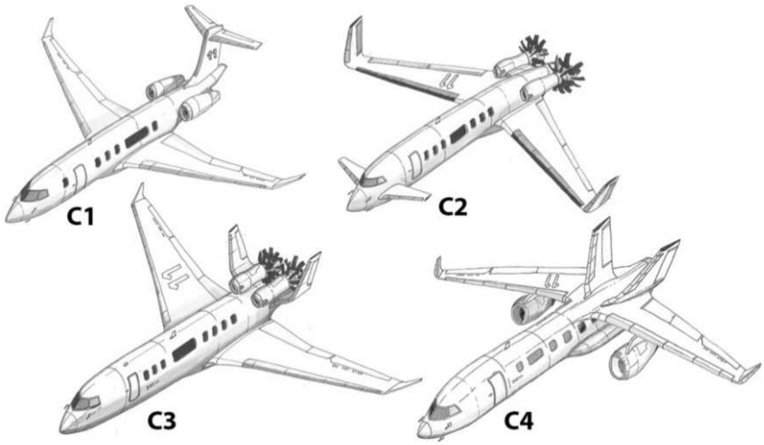

Figure 16.1 The four different concepts

C1 Conventional

The first concept is based on what the competition offers, a mid-fuselage, low mounted wing and a T-tail configuration. This configuration has a clean wing and low cabin noise thanks to the engine placement at the back of the fuselage. The high availability of this configuration proves the viability and provides many reference material. Though improving upon the competition with this design was considered unlikely as it has been optimised for decades already.

C2 Tailless canard

The second concept is radically different from the conventional configuration. The wing is mounted at the back of the aircraft and a canard at the front. The canard functions as horizontal stabiliser and includes the elevator control surfaces. Because both the canard and the main wing provide upward lift during cruise, contrary to the conventional design, it has high aerodynamic efficiency. The lateral stability and control is provided by the super-sized winglets with integrated rudders. Furthermore the design is complemented with fuel efficient open rotor engines.

C3 Pelican tail
The third concept improves upon the conventional design with fuel efficient open rotor engines and a pelican tail. The pelican tail allows for the open rotor engine placement at the back of the fuselage, to reduce cabin noise and provides more safety in case of a blade-off.

C4 High wing
The fourth and last considered concept in the trade-off has a high mounted wing and under wing mounted turbofan engines. The turbofan engines have the advantage of being relatively silent compared to open rotor engines and are inherently safer than open rotor engines. A v-tail is included to reduce drag.

Trade-off
To achieve an objective trade-off, the group members individually assigned weights to the criteria. The average of these weights were then used as the weights of the criteria. Each concept was then judged for each of these criteria. The tailless canard concept outperformed the other three and was chosen as the baseline design for the continued development.

16.4 Final design

Aircraft parameters
The final design has been iterated over the course of several weeks, the final design is described by the following parameters as shown in table 16.1 and figure 16.2.

Propulsion
Open rotor engines consist of a conventional turbojet core combined with two unducted counter rotating propellers as shown in figure 16.3. The open propellers virtually increase the bypass ratio of the complete engine and thereby increase the fuel efficiency. Full scale tests by the FAA have shown that open rotor engines reduce fuel

consumption by approximately 26% compared to conventional turbofan engines at the same thrust setting.

Table 16.1: Aircraft parameters for the final design

General Parameters	
Take-off weight	53,000 kg
Empty weight	27,100 kg
Fuel weight	23,300 kg
Cruise speed	0.8 Mach
Wing Parameters	
Aspect ratio	8.5
Area	136.1 m^2
Geometric span	31.05 m
Mean aerodynamic chord	4.91 m
Quarter chord sweep angle	25.6°
Dihedral angle	3.63°
Fuel tank volume	35.3 m^3
Canard Parameters	
Area	17.7 m^2
Geometric span	12.3 m
Half chord sweep angle	25°
Winglet Parameters	
Area	11 m^2
Height	4.06 m
Leading edge sweep angle	20°
Propulsion parameters	
Engine core	Two BMW-Rolls Royce BR710C4-11
Required thrust	148 kN
Number of blades (front x aft)	12 x 10

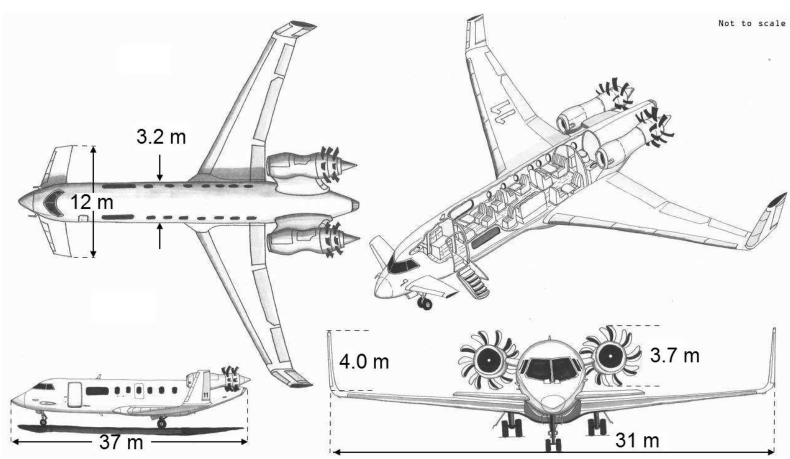

Figure 16.2: Aircraft layout and outer dimension

The reduced fuel consumption also translates into great environmental improvements. Because the propellers are unducted some extra safety precautions are taken in case of a blade disconnection. These precautions include a dorsal shield to prevent cross engine debris, fibre metal laminate materials for extra fuselage protection and strategic engine placement at the back of the fuselage.

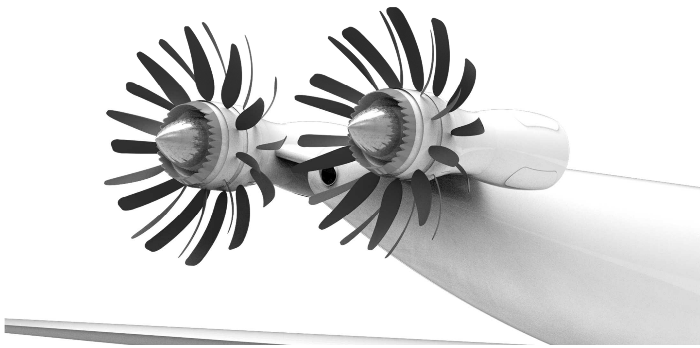

Figure 16.3: Open rotor engines

Materials and production

The material selection has a big influence on aspects or parameters of the aircraft such as: production, maintenance, weight and cost price. The focus was put on different aluminium alloys and carbon fibre reinforced plastics (CFRP) for both the skin and stiffeners' trade-off. Several criteria were examined for the different materials, the first of which includes specific strength. Specific properties are the factors that minimise a component's weight in a load case. Conventional CFRP prepreg sheets were assumed to have a 60% fibre volume and half of the fibres in the longitudinal direction. The CFRP materials outperform the metals in both specific strength and the second criteria, specific stiffness, as can be seen in figure 16.4. This usually leads to 10% to 20% weight reduction. For sustainability, maintainability and cost efficiency the metals outperform the CFRP materials. Because of the high priority on structural weight reduction of the aircraft, the high strength CFRP scored highest in the final trade-off for both the skin and the stiffeners.

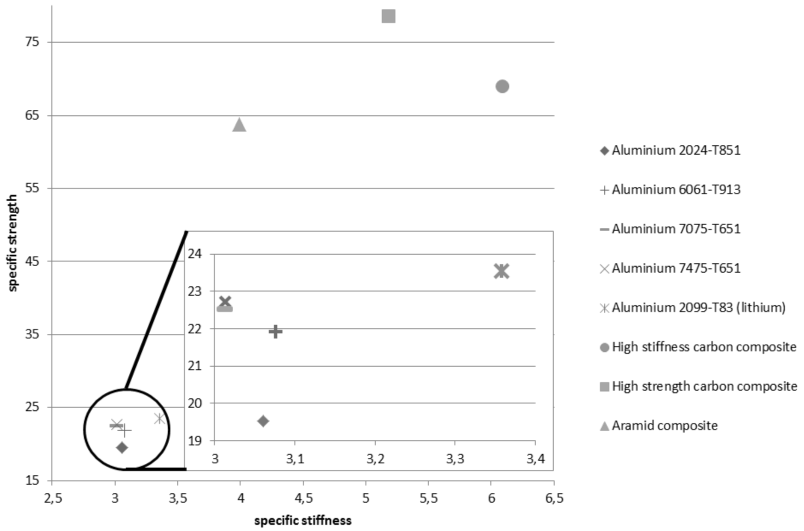

Figure 16.4: specific stiffness $\sqrt{E}/\rho$ and specific strength $\sigma^{2/3}/\rho$ for stiffeners

A production plan was set up, that describes how manufacturing, assembly and integration are organised. The division of the factory into workshops, warehouses, sub-assembly lines and final assembly lines was performed in order to benefit the logistics aspect of the production.

Furthermore parallel activities reduce the throughput time. The activities performed during sub-assembly and final assembly were detailed as a function of station time. This allowed to identify the number of stations needed and calculate various production related parameters such as throughput time, production rate and production run. The throughput time was computed to be 90 days based on a number of 15 stations with a station time of 6 days. The production rate was set at 5 aircraft per month and the production run for 600 aircraft was estimated to be around 11 years. To keep control over manufacturing, over 70% of the components are to be manufactured internally. These include the fuselage panels, the wing box, the wings, the empennage and certain structural components of the landing gear. The remaining 30% of the components are to be obtained through outsourcing and include expensive parts such as engines and auxiliary power units.

Systems

For the cabin interior, the choice was made to use mainly lightweight environmentally friendly composites with natural fibres. These fibres, such as hemp, are carbon neutral and completely biodegradable. The cabin is standard designed with a high luxury interior for eighteen passengers and includes fully reclining seats, a galley and a spacious lavatory with a shower. The fuselage features sixteen windows on each side and one window which is four times larger than the average aircraft window on each side.

The pilots are equipped with a glass cockpit and a bed for long haul flights. Environmental control is achieved with bleed air from the engines, heat exchangers and filters to provide a comfortable 21 degrees Celsius at 4,500 ft. pressure. The wing stows away some of the more technical systems where the first system is the fuel tank with cross feed possibilities for better performance in one-engine-inoperative case. Lastly, the wing also houses the retractable main gears. The placement is balanced between reducing the wing structural weight and complying with different tip-over and ground clearance criteria. The 21 cm wide landing struts are able to cope with worst case scenario landings and the brakes have carbon discs to provide long life time.

Cost

The development of the aircraft is a big investment of two billion USD. After that the design updates can cost up to half a billion. The cost for financing the production will be the biggest cost at 6.5 billion after the production cost of 25.3 billion. The total development and manufacturing of 600 Starling 9000 will add up to 34.5 billion, this averages to 57.5 million per aircraft. At a sales price of 60 million and including the learning curve, the break-even point will be reached in 2027 after producing 385 aircraft.

To the operator of the aircraft it is important to know what the cost of every hour of flying is. This was calculated to be 9600 USD per hour. This is about 15% more expensive than the competition, but the

Starling 9000 takes double the passengers and flies 30% further compared to that same competition.

Canard, winglet and control surfaces design

The Starling 9000 was designed for a favourable balance between the aircraft's inherent stability and controllability. The canard area and the wing position were decided simultaneously considering both longitudinal stability and the controllability. It resulted in a canard area of 13% of the main wing area and a distance of 20.4 m between leading edge of the wing root and the nose of the aircraft. The stability derivative $C_{m\alpha}$ was calculated for this positioning and has a value of -0.41 per radian, which indicates the Starling 9000 is longitudinal stable.

The two winglets on the tip of each wing were designed to be 4 m tall and by locating them after the centre of gravity, they are able to generate stabilising yawing moment, contributing to the directional stability. Furthermore, there are elevators integrated into the trailing edge of the canard for the longitudinal trim and control. The chord length of the elevators is about 40% of the canard chord length with a maximum downward deflection of 25 degrees and maximum upward deflection of -7.7 degrees. Also, employing rudders on the winglets, the Starling 9000 can handle cross-wind landings of 25 knots and is able to trim the asymmetric thrust. The rudders are 80% of the height of winglets and the maximum deflection is 30 degrees.

Aerodynamic optimisation

Because of the limited computational resources that were available, the optimisation was limited to the aerofoil but could be extended to the entire wing in the 3D case. The baseline aerofoil, RAE-2822, was optimised to reduce drag in two ways. The first of which is immediate drag reduction by minimising transonic wave drag. Secondly implicit drag reduction is achieved by thickening the aerofoil. The thicker the aerofoil is, the higher the moment of inertia of the cross section is and the lighter the wing can be. This allows for a smaller wing and thus less drag, with a snowballing effect. The SU2 RANS flow and adjoint solver was used to optimise the aerofoil with 38 design variables.

After four days of iterating an aerofoil was designed with increased thickness and no shockwaves at the cruise condition. In the validation phase it was confirmed that the aerofoil is efficient enough for the Breguet range of 8,500 nm and the internal wing volume is sufficient to comfortable accommodate the fuel tanks and other internal systems.

Structural analysis

A detailed finite element analysis was performed on the wing box structure and the straight fuselage section using Abaqus. This allowed for an understanding of load paths between the wings and the fuselage, as well as critical stress areas in the main part of the airframe. It was also used to prove the feasibility of the large windows that were incorporated into the design, as well as other cut-outs in the fuselage. For this analysis, three load cases were considered: a high-g manoeuvre during cruise, an impact on one main gear strut during landing, and the fatigue life of the jet. With initial design inputs for shell dimensions, stiffness and thickness, a design satisfying all these requirements was found, weighing 14,400 kg for the fuselage and wing airframe combination, which is well below the Class II weight estimate of 18,800 kg.

Sustainable development

While the business aviation is only a small fraction of the total aviation industry, the environmental impact can be substantial. The whole life cycle of Starling 9000 takes this into account and implements various measures in the design phase until the decommissioning of the aircraft to minimise its footprint. These considerations range from engine choice and aerodynamic efficiency to lean manufacturing and end of life solutions. Starling 9000 uses open rotor engines which are known for their efficiency but come with the downside of being noisy. This was looked at early in the design phase by applying noise abatement measures into the rotor design.

Aft clipping has the largest contribution to the reduction of noise which is achieved by clipping the aft rotor length by 5%. This avoids the vortex interaction between the two rows of rotors. Besides, the airframe noise of the Starling 9000 is lower during approach compared to the competitors since the aircraft does not use slats and utilises smaller landing gears. Analysis showed Starling 9000's propulsion is 10-12 dB louder than conventional turbofan but still is 10-13 dB below regulations.

Furthermore, eco-efficient techniques are also a part of the production to improve the overall environmental performance. It will incorporate energy saving, water saving, waste reduction and emissions minimisation. Also, recycling of the business jet forms a part of the design process. It is achieved through either direct use of a part or component, e.g. engine, or recycling many parts of the airframe through thermal pyrolysis which helps to recover carbon fibre from decommissioned aircraft. Finally, resource-efficient and cost-effective factory buildings will be used for the production of Starling 9000. These building will be LEED (Leadership in Energy and Environmental Design) certified, will use less water and energy, therefore reducing cost and greenhouse gas emissions.

16.5 Conclusion and recommendations

The Starling 9000 is a feasible design that can meet the demand for private long distance travel. While meeting the requirements of hauling 18 passengers over 8500 nm, the spacious interior was designed to impress the passengers and enable them to enjoy the superior and sustainable comfort of organic fibre sound insulation, fully reclining seats, a spacious galley, a shower, and panoramic window four times the size of a standard aircraft window.

A fully carbon composite structure reduces the weight of the Starling 9000 and its open rotor engines give it up to 26% better fuel efficiency as compared to turbofan powered competitors. Aerodynamic improvements such as aerofoil optimization are employed to reduce the cruise drag, and the structure of the wing. Its unconventional canard configuration is designed to maximize the aerodynamic efficiency. Elimination of the tail leads to significant reduction of drag.

Pitch control is enabled by movable canard surfaces while the rudders integrated in the big winglets can provide yaw control. Moreover, the design of the Starling 9000 leads sustainable business aviation by designing for low noise, optimum life-cycle cost, end-of-life solutions and LEED certified lean manufacturing. These improvements directly translate in customer value, with an overall decrease in operational cost of as much as 18% compared to the competitor. The starling 9000 fulfils the requirements and offers tomorrow's corporate travel superior global access.

For the further development of the Starling 9000, recommendations are proposed for future investigation. From the engine aspect, additionally modelling and experiments should be performed regarding the noise analysis. Also the jet core and the open rotors needs to be analysed together to determine the combined noise generation. As for aerodynamics, it is recommended to perform further 3D RANS analysis of the wing.

Future design efforts should focus on optimising both the geometric twist distribution of the wing as well as the aerodynamic twist resulting from the need to reduce the thickness of the wing at the tips in order to decrease transonic shocks. SU2 is known to also be an appropriate tool for this 3D optimisation. Moreover, wind tunnel tests are suggested to be performed to estimate accurately the interactions e.g. upwash effect between main wing and the canard, sidewash between fuselage and winglets for further analysis of stability. What's more, yaw damper could be applied to augment lateral-directional dynamic stability.

17. ORRERY

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17.1 Introduction

Since 1704 a clockwork mechanism has been used to simulate the positions, motions, and phases of the planets in the solar system. A system which serves this purpose is called an orrery. With the recent advance in aerospace technology Unmanned Aerial Vehicles (UAVs) have become increasingly popular part of our everyday lives. With the introduction of UAVs, a vision for the redesign of the age old orrery concept was born. This vision led to Project HORUS (Heliocentric Orrery UAV System). The mission statement of the project was formulated as follows:

"Design an educational and interactive orrery system which utilises spherical UAVs flying in formation to represent the solar system's planets and their planetary motions which will be operational by 2017, cost less than €1,000,000, and be designed by nine students in eleven weeks."

HORUS aimed to represent the planetary motions by designing spherical UAVs flying in formation. This system needed to function as an educational system, such as one would see at a science centre, as well as facilitate human movement between the units, and provided the ability for observers to interact with the system. No system like this currently exists, which was why the group was tasked to design this system.

17.2 Design requirements and constraints

As stated, HORUS was designed to serve as both an educational and interactive system. Additionally the system had to meet specified safety regulations. The driving requirements as given to the design team by the customer were as follows:

Mission requirements:
- The system shall consist of UAVs representing the planets within the solar system.
- The system shall represent the orbital motion of the planets within the solar system.
- The orbital motions shall occur in a non-linear coordinate system.
- The system shall function as an educational tool.
- The system shall have interactive capabilities.
- The system shall facilitate human movement between planets.
- Each UAV shall mimic a specific planet in an identifiable manner.

Design requirements:
- The system prototype shall have a maximum cost of €1,000,000.
- The system shall function 8 hours per day for 7 days per week for a minimum of 1 year.
- The UAV shall be scalable.
- The UAV shall have a crash-resistant structure.
- The UAV shall be able to take off after crash landing.
- The UAV shall have a spherical shape.
- The UAV shall recharge autonomously.
- The system shall function within a flight envelope of 10x10x5 m.
- The UAV shall not have a mass exceeding 2 kg.

Safety requirements:

- The UAV shall not reach a speed higher than 4 km/h.
- The UAV exterior shall comply with playground regulations.
- The UAV shall land without causing injury when loss of power occurs.
- The UAV shall land without causing injury when loss of communications occurs.
- The UAV shall have a collision avoidance system.

17.3 HORUS concept trade-offs

The conceptual design phase consisted of a number of consecutive trade-off rounds, from which a final concept was selected for the preliminary design. During the initial trade-off the non-concepts, the not yet sufficiently developed concepts, and the infeasible concepts were eliminated from the Design Option Trees and strawman concepts were created for each of the different system sections. In the second trade-off each of the remaining concepts were analysed to decide what type of concept was going to be used for each segment. The third and final trade-off led to the determination of the overall system layout, material, and specific versions of the system types decided in the previous trade-off.

First trade-off

As stated, the purpose of the initial trade-off was to eliminate any infeasible concepts, not yet sufficiently developed concepts, or non-concepts. Doing this early in the project eliminates the allocation of time on analysing these concepts. Non-concepts and infeasible concepts are concepts for which no method of implementation can be thought of. Not yet sufficiently developed concepts are concepts for which a method of implementation is conceived, but the technology as it is currently or well be the foreseeable future has not reached the required level of advancement yet. A list of the most important eliminated concepts is shown below.

Eliminated infeasible or non-concepts:
- Jet engines for propulsion
- Hot air balloon for propulsion
- Acoustic levitation for propulsion
- Parachutes for landing
- Turn off and drop for landing
- Projection from the Sun for visualisation
- Name tags for visualisation
- Animal power for ground station power
- Non-rechargeable batteries for ground station power
- Nuclear power for ground station power
- Fossil fuel power for ground station power
- Wired connection for system communications.

Eliminated "not sufficiently developed" concepts:
- Virtual Reality for visualisation

Second trade-off

In the second round of trade-offs all remaining concepts were evaluated. The main points of attention during this trade-off were the complexity, mass, power usage, reliability, sustainability, and cost of the specific concept. For the propulsion system of the UAV's co-axial propellers had been chosen. For the visualisation a combination of flexible screens and coating was proposed, but the usage of flexible screens was later removed due to mass constraints. For the power supply of the UAVs rechargeable batteries had been chosen to provide the power to the system. In order to facilitate recharging two sets of eight UAVs were used.

For the ground station power a regular mains connection had been chosen. For the communication and control system of HORUS it was decided to have the control system decentralised, spread across all UAVs and the server, and communicating with each other via Bluetooth. For the design of the Sun it was decided at first to make it a UAV, however, this was later changed to a sphere suspended by cables. Lastly, the concept for charging was the usage of a cabinet in which the UAVs could autonomously enter and start charging.

Third trade-off
In the final trade-off the remaining concepts from the second trade-off were worked out in more detail. For the propulsion system it was decided to utilise two propellers on one axis, counter rotating to minimise the net torque. Each of these propellers will be powered by a brushless electrical motor. The power was going to be supplied to the UAVs by using Lithium-Polymer (Li-Po) batteries, and the units would recharge via conductive charging. For the control method of the UAVs, UAV based positioning, path planning, and collision avoidance will be utilised. To facilitate this a micro-computer was chosen.

For the positioning, five Bluetooth beacons were going to be used, for the path planning custom code had been used, and the collision avoidance would be facilitated by the use of optical infrared sensors in combination with LEDs. For the controlling of the path of the UAV, variable pitch was used to change the attitude and position of the UAV. For the structure of the UAV, the shell consisted of two separable parts, which could be separated along the horizontal plane. Internally the structure would be present on the bottom half of this shell. Lastly, for the visualisation of the Sun, a sticker will be used in combination with high power light bulbs installed inside the shell.

17.4 Final HORUS design

Once a final concept was selected it moved on to the next phase of the design process. All subsystems of the proposed system underwent a detailed design. For the UAVs these subsystems were the structures and materials, the power supply, the control system, and the propulsion system. Additionally the Sun and the ground station segment were designed in more detail.

Structure and materials
The outer shell consisted of two polyurethane foam hemispheres. The attachment points of these hemispheres were made out of polypropylene plastic. The internal structure which provides the units with structural strength consisted of a certain number of carbon fibre

beams. The number of beams is 8, 14, 10, and 8 for the smallest to largest UAV size, respectively. In order to improve the efficiency and performance of the propulsion unit a shaft was put around the airflow with the two co-axial propellers inside of it. This shaft was made out of polypropylene plastic. Furthermore, the top and bottom of the structure were closed from access to observers by an aramid fibre honeycomb mesh.

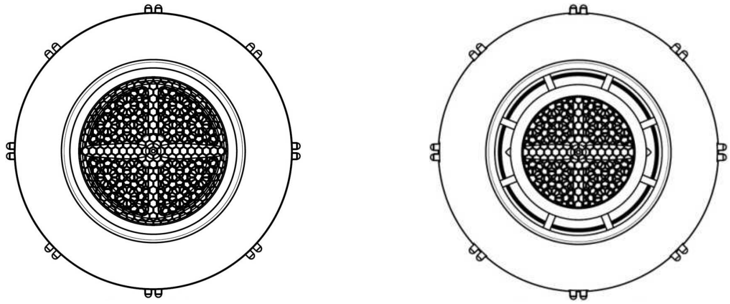

Figure 17.1: Top view and bottom view of the UAV

Power supply

All but the largest sized UAV featured two custom shaped Li-Po batteries. The largest sized unit featured one additional battery of a smaller size to provide enough power for the full desired flight time. The batteries would be recharged using conductive charging. The obtained flight time for all UAVs was 15 minutes, and the possible recharge time was roughly 7 minutes.

Control system

For the control system, a Raspberry Pi 3 Model B micro-computer was chosen, using an internally produced program. For the positioning five Bluetooth beacons were used together with a least-squares algorithm to determine the position of the UAVs. The collision avoidance system functioned using a certain number of sensors which feature a pairing of an infrared sensor and LED. The number of these sensors is 8, 14, 10, and 16 for the smallest to largest UAV size, respectively. For the precision landing the UAVs were equipped with photoresistors, which would be used to line up the UAV with their docking station.

Propulsion

The propulsion for the UAVs was provided by two propellers, counter rotating and on the same axis. Each of the propellers was powered by a brushless electrical motor, which facilitates the usage of a variable pitch method for the position control. Furthermore, to decrease the influence of the first rotor on the second, a honeycomb mesh was positioned in between them to straighten the airflow.

Visualisation

For the method of visualising each planet in a distinct manner it was opted to apply a coating on the polyurethane shell. This allows intricate images to be displayed on each of the unit, identifying which of the plants of the solar system it is supposed to represent, and only adding a negligible amount of mass to the system.

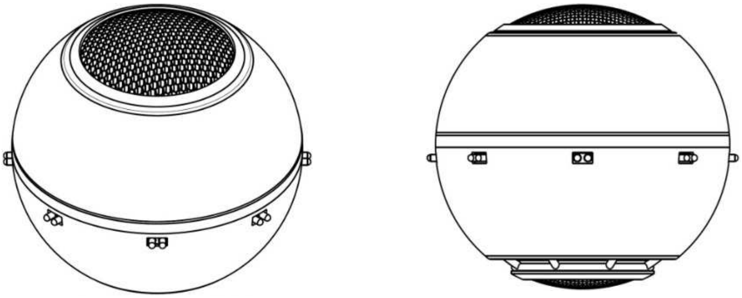

Figure 17.2: Isometric view and side view of the UAV

Sun design

The Sun consisted of a transparent perspex shell with a sticker over it which served as the visualisation. Inside of the Sun four lamps would be fitted to provide full light coverage of the area around it. The Sun was suspended by three cables from the ceiling of the room. These cables were attached to motors via a pulley system which allowed changing the altitude of the Sun, and thus the entire system.

Ground station

The ground station consisted of three main parts: the server, user interface, and the charge station. The charge station was a large case with sixteen pedestals in it. Each pedestal was the charging position of one of the UAVs. In order to prevent observers from interacting with the units while they are charging, a transparent divider was placed between them and the charging units. The user interface was a large touch screen which allows all users to interact with the system. Lastly, the server was a regular laptop, as this provided the system with all necessary functionality.

Orbital model

The system supported a number of unique modes which all had different purposes. Some of these modes, such as presentation mode, bolster the educational value. Others, such as interaction mode facilitate interaction between observers and the units. Outside of these modes, the system operated in default mode. In the default mode the orbits featured their actual inclination and order in the configuration. Only the two most eccentric orbits had their eccentricities and the system orbited at an altitude of 2.5 m.

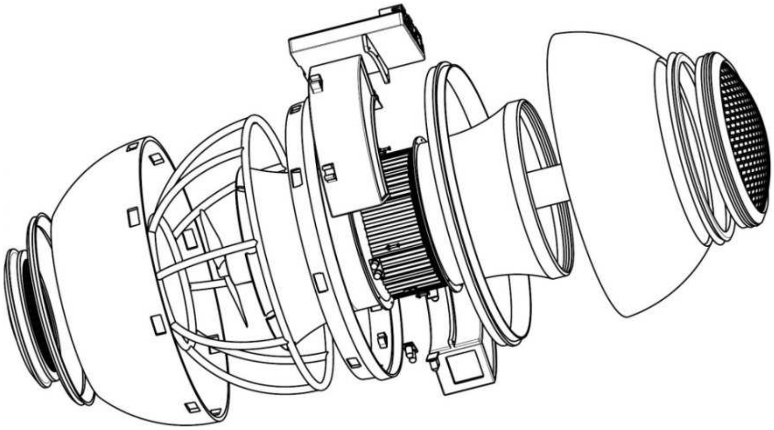

Figure 17.3: Exploded view of a UAV

17.5 Conclusion and recommendations

Conclusion

As the mission statement stated, the goal of the project was to “Design an educational and interactive orrery system which utilises spherical UAVs flying in formation to represent the solar system’s planets and their planetary motions which will be operational by 2017, cost less than €1,000,000, and be designed by nine students in eleven weeks.".

By the incorporation of certain modes that bolstered the educational value of the system and added to the interactive capabilities made it so that both an educational and interactive system had been designed. The UAV structure consisted of a shell, an internal structure to provide strength, and a shaft which increased the performance of the UAV. The UAV was propelled by two propellers which were rotated by two brushless electrical motors which were in turn powered by Li-Po batteries. The Li-Po batteries also provided power to all components within the UAVs. These additional components were the micro-computer, the IR sensors and the photoresistors. Outside of the UAV there was a charge station to allow the autonomous charging of the UAV, a server which handled all the communication between itself, the interfaces, and the UAVs. Lastly there were five Bluetooth beacons present in the room to provide a way to determine the position of the UAVs. It was determined that the system would cost a total of €150,000, which was much lower than the €1,000,000 as was set by the requirements. With all this the system adhered to all of the imposed requirements.

Recommendations

Given the time constraint posed during the DSE, some aspects of the design have not been designed into the desired level of detail. Additionally, some improvements in the design can still be identified. The recommendations aim to further develop the design of the orrery and to improve the functionality and feasibility of the system.

The control system can be optimised to function better on a Raspberry Pi, and a new trade-off can be performed to see if a micro-controller such as the Arduino can be used instead. Because this is a lighter, less-power consuming system, it might be a better choice. The collision avoidance and positioning subsystems might be improved by experimental hardware testing, to see if they cannot be used more optimally than the data-sheets for the various hardware components dictate. The orbit model feedback might be improved as well when the true properties of the components are known.

The internal structure can be optimised in several ways. First the upper ring can be optimised to withstand the forces experienced during a crash on the side of the UAV. This can be done via the same iterative method as performed. This however would most likely increase the overall mass of the UAVs as the dimensions would increase due to the lower number of beams on which the force is distributed. It is not expected to have the UAV crash on its side however. Regarding the optimisation of the current design, local weak points can be improved by, for instance, introducing taper. Furthermore, the manufacturing methods can be perfected by trial and error or new production methods. In case connectors are necessary, these can be custom made to minimise the mass and maximise the structural properties.

All in all, the batteries were sized to fit the required power and capacity of each sub-system without over designing. Improved batteries may be on the market in the future, which could enhance the battery mass, capacity and charge time. It might also be possible to discard the second set of UAVs if the battery performance in increased.

The visualisation system of HORUS can be improved in many ways. For instance e-paper can be used. This adds an extra dimension to the orrery with a dynamic visualisation and also allows for some degree of interaction with the users. With a decreased coverage of the e-paper on the shell the visualisation is feasible with the thrust generated.

For the propulsion system the design could be improved by getting a better understanding on the effects of all the parts on the thrust being generated. This can first be done by setting up a proper Computational Flow Dynamics (CFD) model for analysis and later on during production with wind tunnel testing.

18. PICS (PROMOTION & INSPECTION CUBIC SATELLITE): DESIGNING A FEMTO-SATELLITE FOR INSPECTION AND PROMOTION PURPOSES

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18.1 Background

Since the launch of the first satellite in 1957 the mass of satellites has increased tremendously, leading up to the 17,000 kg Compton Gamma Ray Observatory. Due to innovations in the consumer electronics industry, the size and mass of satellites have decreased significantly over the last few years. Recently, Cubesats have established their presence within the space industry, e.g. universities and other institutions are currently making use of Cubesats to motivate and support the education of their students for a price that is within their budget. The miniaturisation still continues; the next step in this trend is for satellites to reach the sub-100 g mass. Satellites that fall into this next class of satellites are called femto-satellites. This brings us to the objective of this DSE:

"Design a femto-satellite for technology demonstration."

These satellites allow for potential new markets to be opened by allowing for cheaper and faster launches of satellites. This high level of miniaturisation includes many challenges but also opens up many new opportunities. The challenge is to exploit these opportunities and to make clear to the space community why femto-satellites are such a good alternative to larger satellites. That is why a mission needs to be performed that can show this potential and explain why femto-satellites have a place in the satellite market.

18.2 Mission and requirements

Mission selection

The objective of the DSE gave the opportunity to define a mission. This mission had to be in line with the project objective. Based on a literature study, a mission trade-off was performed in order to obtain an achievable mission which can demonstrate the potential of femto-satellites. From the trade-off, four possible concepts were selected:

- Inspection Femto-Satellite
- Scientific Experiment
- Earth Observation Femto-Satellite
- Space Advertisement Femto-Satellite Constellation

Building on these concepts, detailed research was done to make a definitive selection. By analysing the technical feasibility, need, cost, risk, demonstrability and sustainability of each concept, the Inspection Femto-Satellite concept was chosen as the final mission.

Mission Statement

Starting from the selected mission concept, it was decided that in addition to inspection of the host spacecraft the femto-satellite will also be used to take promotional pictures. Hence, the final mission statement was defined as:

"DSE group S13 at the Faculty of Aerospace Engineering at TU Delft will design a femto-satellite to demonstrate its inspection and promotion potential."

Based on this mission statement the Promotion & Inspection Cubic Satellite (PICS) was introduced.

Host spacecraft

In 2016 and 2017 the Google/Alphabet owned company TerraBella will launch a large number of Earth observation satellites in the SkySat C series. These satellites are used for high resolution panchromatic and multispectral Earth observation at an altitude of 450 km. The SkySat C satellites are attractive as host spacecraft for several reasons. First, the satellites are all relatively small (60×60×95 cm) with no large extending components. Furthermore, due to the fact that a large series of them is planned (13 guaranteed) they benefit greatly from inspection since defects found on the earlier models can be fixed on the later models. Additionally, due to its relatively low altitude it is possible to quickly de-orbit the femto-satellite and thus no contribution to space debris will be made. Lastly, Google has spent a large amount of money on new technology start-ups and ideas so it is not unimaginable that they will fund a project like PICS.

Objectives

After having defined the mission statement above, it was decided through a literature study that the inspection potential of femto-satellites is demonstrated best by inspection of damage on solar panels caused by space debris and micro-meteoroid impacts. Currently the only way to assess this impact is to return solar panels back to Earth. This has been done for the European Retrievable Carrier (EuReCa) in 1993 and again for the Hubble Space Telescope in 2002 using the Space Shuttle. Since the Space Shuttle has retired, there exists a need to develop new means to investigate impact damage by space debris and micro-meteoroids in order to update the obsolete existing models. PICS will be launched as a test mission to serve this purpose. If the PICS mission is successful it can be expanded to many more femto-satellites which together can provide a valuable dataset for modelling small space debris and micro-meteoroids in a certain space environment.

Regarding promotion, the space industry is currently unable to provide the general public with real, in-orbit images of their spacecraft. As a substitute 3D renders are often used, but these renders will always be underlined with the words 'artist impression', which takes away the ability of the general public to grasp the work of space organisations. This is where PICS provides a solution: imaging spacecraft, allowing space organisations to share real, in-operation images of their space operations which makes the general public more acquainted with space travel.

Apart from the objectives mentioned above, another objective was defined in the course of the design process. This objective directly relates to a major concern that is widely shared within the space industry with regard to femto-satellites: space debris. The amount of space debris in Low Earth Orbit (LEO) has increased rapidly over the last twenty years, becoming an increasing threat to future space missions. Because of their low price, it is likely that once the potential of femto-satellites has been demonstrated by PICS, large numbers will be launched into orbit. Also, their small size implies that femto-satellites are difficult to track from Earth. Hence, a fear exists that femto-satellites will contribute to the formation of space debris. To resolve this concern, it was decided that PICS will not contribute to space debris by the instalment of an active de-orbit device on the femto-satellite. On command this device will deploy an inflatable balloon which increases the drag experienced by PICS and thus de-orbit the femto-satellite in due course.

Requirements and constraints

In order to achieve an adequate and fluent design process for PICS, requirements need to be well-defined. The mission requirements are defined by complying with the constraints set by the stakeholder requirements and fulfilling the objectives listed in the previous section. Below the most important requirements for the PICS mission are listed, this includes the top-level, driving and key requirements:

Top level requirement

- The spacecraft shall comply with the existing regulation to mitigate space debris, which requires the femto-satellite to de-orbit within 25 years.

Driving requirements

- The femto-satellite shall be able to capture visual images with a smallest feature size of 0.1 mm.
- The femto-satellite shall be made detectable from the Earth after its operational lifetime.
- The spacecraft launch mass shall not exceed 100 g.
- The spacecraft volume shall not exceed 125 cm^3 .

Key requirements

- The cost of each individual spacecraft shall not exceed €10,000.
- The femto-satellite shall be able to identify space debris impact on the solar arrays of the host spacecraft.

18.3 Concept selection

After mission and requirements were established, the design concept used to demonstrate the mission needed to be set. Considering all the requirements and constraints, three concepts were proposed.

Orbiting
This concept consist of a femto-satellite which will be attached to the host spacecraft during launch. At a to be determined time the femto-satellite will be released from the host spacecraft. The femto-satellite will be deployed in such a way that it can inspect the host spacecraft. The concept will be self-sustained in space.

Docking
This concept also consists of a femto-satellite which will be attached to the host spacecraft during launch. After launch the femto-satellite can be released, perform a mission for half an orbit and re-dock with the host. During docking the femto-satellite can refuel, recharge and send the data to the host.

Pick and Place

This concept is specifically designed for the ISS. Here a femto-satellite is deployed from the ISS to inspect the ISS on damage. Also the femto-satellite can assist with human space operations. The femto-satellite can be deployed and re-dock with the robotic arm on board of the ISS. During docking the femto-satellite can be recharged.

In order to select the best concept, a trade-off was made to indicate the strengths and weaknesses of each concept. Five different criteria were used to evaluate each concept: mass, power, sustainability, technical performance, risk and cost. Different weights were given to different criteria. Based on the trade-off, the orbiting design was selected to be the final concept as it scored the highest among all three concepts. After having performed a sensitivity analysis, it is safe to say that orbiting is the most suitable concept to fulfil the mission.

18.4 Design details

After the concept has been selected the next step was to perform a detailed analysis into the mission and to design the required subsystems. Multiple iterations were needed to arrive at the final design as presented in this section. The configuration of PICS is shown at the end.

Astrodynamics

PICS will be released from SkySat by a deployment system. The velocity increment delivered by the deployment system will be applied in a direction opposite to the velocity vector of the host spacecraft. This causes an increase in eccentricity and orbital period. In this way PICS drifts away from SkySat due to the difference in atmospheric drag caused by the mutually different ballistic coefficients. After one orbit PICS will be behind the host spacecraft after which PICS catches up on the host spacecraft due to a higher velocity. Meanwhile the drag causes the femto-satellite to decay faster than SkySat, bringing the two closer together again. Finally PICS decelerates further, leaving SkySat behind. The distance between SkySat and PICS as a function of time is plotted in figure 18.1

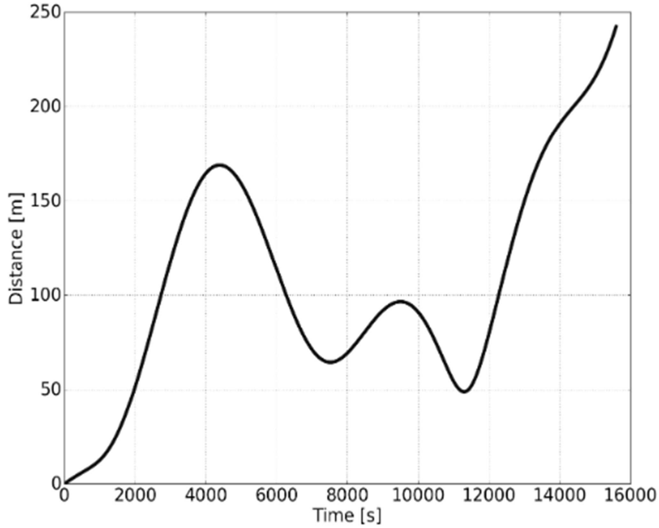

Figure 18.1: Distance between SkySat and PICS as a function of time

Payload

The first part of the trajectory will be used to take pictures of the host spacecraft. PICS uses the 16 MP camera module of the Samsung S6 mobile phone which only weighs 1.9 g. The imaging starts at 0.39 m. At that distance the smallest feature size on the photo is 0.1 mm which is needed to observe the impact craters on the host spacecraft. After 700 inspection pictures are taken in 300 s, PICS will be at a distance of 4.04 m. Subsequently 178 promotion pictures will be taken in 142 s. The imaging process will stop at 5.61 m away from the host spacecraft. The inspection pictures will be post-processed on the ground with multi-frame super resolution to restore the image quality.

Communication

The PICS' mission will be performed in close proximity of the SkySat. This gives the opportunity to use short range communication devices. Then the SkySat can send the data to the ground for further analysis. This is needed because communication devices to communicate to the ground are too large and too heavy for a femto-satellite. A WiFi module with PCB trace antenna was selected. This provides a data rate of 54 Mb/s with a maximum range of 242 m. The mass of the WiFi module is 5 g. The WiFi module will be placed on the SkySat and PICS. The total communication time to send all the pictures is 6250 s. During the whole mission, until PICS is out of communication range, telemetry will be sent to analyse the envelope of the mission.

ADCS

The attitude determination and control subsystem is designed to keep the payload pointed towards the SkySat. To know its orientation relative to the SkySat, the embedded gyroscope of an Inertial Measurement Unit (IMU) is used. Edge-detection algorithm will be performed periodically on images taken with the main camera to counter any inherent drift of the IMU. This will give sufficient pointing accuracy for the main camera to operate.

A three-axis stabilised system using momentum wheels has been designed to control the attitude of PICS. It will be active during the time the payload is taking images of the host-spacecraft and switch off afterwards. There is no need for momentum dumping due to the low operational lifetime. It is sufficient to scale them properly for the duration they are active. The ADC subsystem has a total mass of 5.15 g.

De-orbit

PICS is a femto-sized satellite, which can be easily associated with space debris. Therefore the designers went for an active de-orbit device to guarantee a short after-mission time and so reduce space debris. This resulted in a spherical inflatable de-orbit device (SIDD), which is a Kapton balloon that will inflate to decrease the mass-over-surface-area ratio. The balloon with a radius of 10 centimetres will be inflated with nitrogen gas from a cool gas generator. The mission will be performed on an altitude of 450 kilometres, which results in a natural de-orbit time of 120 days for a PICS sized satellite. Yet PICS, with an SIDD, can reduce this to a de-orbit time of 10.5 days. The total mass of the SIDD is 12.33 g.

Command and data handling

The command and data handling subsystem regulates the data transfer between the multiple subsystems. A processor is used to regulate the signals between the ADCS, payload, SIDD, power and communication. Also a direct connection between the payload, mass storage and communication is set-up to prevent an overload in the main processor.

The information produced by the payload can not be sent directly by the communication subsystem. So the payload will write the information from the camera to a mass storage device. This will then be read from the mass storage device to the WiFi module to send it to the SkySat. This subsystem has a total mass of 1.45 g.

Power

Due to the relatively low operational lifetime of the PICS mission, it is possible to use unconventional ways of providing power to the satellite. This is done using a non-rechargeable battery which is able to provide up to 4.5 Wh during the 2 hours of operations. A physical switch is present to connect the main battery to the main satellite bus before operations to minimise energy losses. This switch is activated before deployment by an electric rod-style actuator (ERSA) on the deployment system. Two auxiliary batteries are added to deliver high power for a short duration to activate the de-orbit device which will be on a separate circuit to increase safety. The power subsystem has a total mass of 24 g.

Thermal system

The components are kept within their temperature ranges by a passive system. A 25.4 μm thick kapton foil of 0.5 g covering the majority of PICS was chosen for that purpose. It has just the right absorptivity and emissivity to keep femto-satellite's components within a temperature range of 5 and 40 °C while images are taken and between -10 and 50 °C during the rest of the lifetime. This was verified with a 1-node thermodynamic model which was validated with telemetry from the FUNcube satellite.

Deployment mechanism

PICS is deployed from SkySat with a deployment system using four springs. Before deployment PICS is restricted from any movement by a closed deployment door. The ERSA is included in the deployment system which activates the femto-satellite before deployment. Subsequently the deployment door will open. A nichrome wire attached to the deployment interface burns the loop connecting the Vectran cables that pull back the springs and the attached supporting plate. In this way the springs are released, the supporting plate pushes forward and PICS is deployed from SkySat with a velocity increment of 1.3 cm/s.

Vibrations and structure

The structure consists of 3 printed circuit boards (PCB) which transfer the loads to four pins which in turn transfer the loads to two frames. On the outside of the frames six plates are mounted that support the kapton foil. This gives a total mass of 21.46 g for the structure. The structure was sized so that it and all its components have a natural frequency higher than 35 Hz and can withstand a launch acceleration of 10.8 g. These requirements originate from the launcher. It turned out that the requirement on natural frequency was leading and that the structure was over-designed with regard to static accelerations.

Configuration

After research, a 5×5×5 cm cube was selected to be the shape of PICS. All the subsystems were designed to be installed on the three PCBs. These PCBs were placed in 3 layers. As can be seen in the final layout pictures (Figure 18.2), the placement of different subsystems was considered based on their functions and limitations.

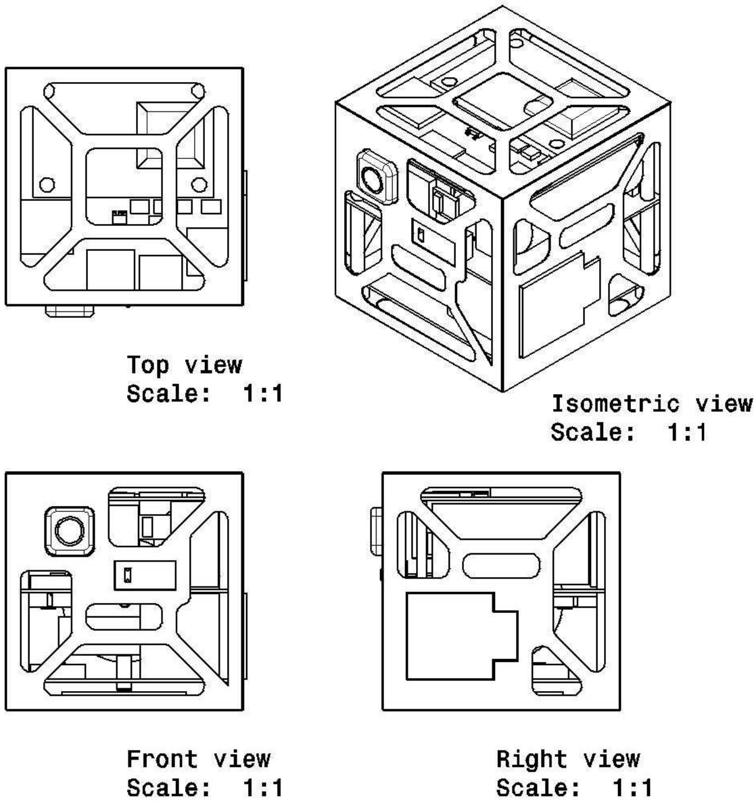

Figure 18.2: 3-view drawing of PICS

As the final design was made, the total weight and bus cost of the femto-satellite could also be calculated. By summing up all the component weights, the final weight for PICS was found to be 71.99 gram. Similarly the final cost of PICS was calculated as 7649 euro.

18.5 Vision on femto-satellites

After the DSE, group S13 believes there exists a high potential in the use of femto-satellites. When people talk about femto-satellites, they often discuss swarming. However, the problem with swarms of femto-satellites is that (depending on the application) they require a very high amount of inter-satellite communication, high accuracy position determination and/or formation flying. Up to this day no spacecraft except the International Space Station has ever carried a secondary satellite on board to be deployed for inspection and promotion purposes. Hence, there exists a niche in this functionality which can be occupied by femto-satellites. For these purposes femto-satellites have an advantage over Cubesats since they pose minimal impact on the host spacecraft's mass budget. Also their low price will make them extremely attractive to spacecraft operators who have an interest in inspection and promotion of their spacecraft.

PICS is a test mission to demonstrate the data acquisition of the space debris and micro-meteoroid environments. Group S13 acknowledges that a single femto-satellite cannot gather sufficient data to provide a significant contribution to validating and updating current space debris and micro-meteoroid models. However, if each future SkySat and potential host spacecraft would carry a femto-satellite, a much better model of the space debris and micro-meteoroids environment can be formed.

DSE group S13 thinks that the application of taking promotion pictures together with a proper marketing plan can drive the popularity of the space industry for the general public given the significant usage of social media and advertisement these days. Together with the scientific application lying in the contribution to the mapping of the space debris and micro-meteoroid environments, the usage of femto-satellites is promising.

18.6 Conclusion and recommendations

The PICS mission will show the potential of femto-satellites by taking inspection and promotion pictures of the SkySat and will only weigh 72 g. PICS will cost 7,649 euro. The inspection pictures can be used to analyse impact craters from space debris and micro-meteoroids. When multiple PICS missions are deployed existing space debris models can be updated and validated. The promotion pictures will help space agencies to promote their work and to increase the popularity of space projects. This will result in a more effective lobby towards governments, which will be more likely to supply funding. Hence, by demonstrating the potential of femto-satellites the first step is taken to establish a foothold in future spaceflight.

In conclusion, a preliminary design of PICS that can be effectively employed for inspection and promotion has been presented in this study. Further analysis on subsystems such as structural and thermal subsystems is needed to derive more precise budgets. The design of the deployment mechanism and SIDD need further investigation to optimize their performance.

19. Project Matryoshka: Finding Venusian Volcanoes

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19.1 Introduction

Venus, the second planet in the Solar System is often called Earth's sister planet, because of its similar size, composition and position relative to the Sun. While for a long time, people expected a green paradise below the ubiquitous cloud decks, past missions such as the Pioneer and the various Venera that visited Venus, have shown the opposite to be true. Below the thick clouds lies a barren wasteland with a scorching surface temperature of over 450^o Celsius and a surface pressure of 90 times that on Earth. The atmosphere mainly consists of carbon dioxide while the clouds are composed of droplets of sulphuric acid solution, which eats through most materials.

The clouds and the harsh environment below them have prevented past missions from gathering much data on the Venus' surface, as landers typically last less than three hours before succumbing to the heat. NASA's 1980's Magellan spacecraft used radar to penetrate the clouds and generate an altitude map of the surface, which showed many signs of possible volcanic activity such as mountain ranges with calderas and small domes. Another strong indication of volcanic activity was data gathered by ESA’s Venus Express spacecraft. A comparison with decades old Pioneer data showed temporal fluctuations in the sulphur concentrations in the clouds, meaning that sinks and sources have to be present. One such source could be volcanoes, that, like those on Earth, expel large amounts of sulphur. Despite these observations, no solid claim could yet be made on active Venusian volcanoes.

The confirmation that active volcanoes exist on Venus would help significantly in understanding its atmosphere, climate, internal properties and geological characteristics. Project Matryoshka aims to deliver this confirmation and through it we hope to answer a fundamental planetary science mystery and re-invigorate the drive to explore Venus in the near future.

19.2 Mission objectives and requirements

Project Matryoshka’s primary mission objective is:

“to find active Venusian volcanoes and characterise their eruption products.”

Additionally, the mission will complete a number of secondary objectives, such as gathering atmospheric data and possibly detecting Venusian lightning. Another important secondary objective is to gain a deeper understanding of Venus’ atmosphere and its runaway greenhouse effect, as that would help to improve the climate prediction models for Earth, and, in addition, raise awareness about the effects of climate change among the public.

The most important requirements to which the design has to adhere are:

- The observations shall be able to unambiguously detect volcanic activity, if present.
- Once an active volcano has been identified, the eruption products shall be analysed.
- The spatial resolution of the eruption product measurements shall be high enough to identify sources and sinks of eruption products.
- The launch date shall be no later than 2026.
- The use of radioactive materials shall be avoided, unless there are strong and compelling reasons to do so.
- The mission cost shall not exceed €1.0 billion (FY2016), excluding launch and operations.
- The launcher selection shall be based on existing launchers.

19.3 Concepts and trade-offs

In the concept study it was decided that any concept would include a spacecraft to get to Venus, and a lander to perform surface measurements. Therefore, these were excluded from the concept analysis.

To fulfil the mission requirements, we considered four different vehicles for use in the Venusian atmosphere (see figure 19.1); an aircraft, a blimp, a balloon, and a blimp-craft. Each of these can roam the Venusian skies in search of volcanic activity. When an active volcano is located, a lander is sent for in-situ measurements, either from the spacecraft or from the airborne vehicle.

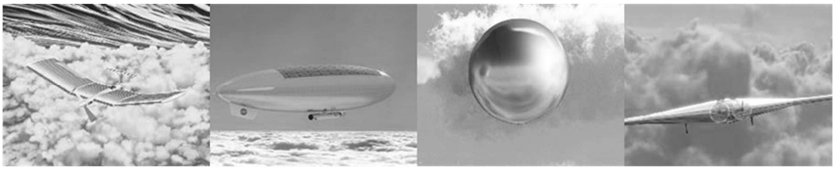

Figure 19.1: Four concepts considered (from left to right): aircraft, blimp, balloon and blimp-craft

Aircraft

The aircraft requires multiple propellers, powered by solar panels, for propulsion. Because of the slow rotational velocity of Venus, the aircraft can remain on the planet's dayside. The aircraft can fly below the clouds to do accurate measurements after which it rises above the clouds to recharge its batteries. Since the aircraft can carry the lander, it can accurately drop the lander when the location of an active volcano is known.

Balloon

Balloons can passively circumnavigate the planet using the strong winds, floating around Venus while scanning the surface. The zonal winds blow the balloons around Venus while the meridional winds slowly push them towards the poles. As balloons are cheap and light, multiple of them can be sent, allowing for larger coverage of the planet.

Blimp

Compared to the balloon, the blimp has the advantage of being able to steer in a certain direction. Using propellers, the blimp can counteract the meridional winds, increasing its lifetime around the equator. Furthermore, it can take several measurement instruments and power them with solar cells that fit on top.

Blimp-craft

The blimp-craft is a combination of an aircraft and a blimp. When residing in sunlight, it can actively fly wherever it wants as an aircraft, while it will passively float like a blimp when it reaches the dark side of Venus. This concept essentially combines the advantages of both concepts.

Final selection

The aircraft concept was chosen as a result of a thorough trade-off. The ability to actively steer and fly was considered important, so the balloon and blimp concepts were discarded. The blimp-craft has as disadvantages that it should be enormous in order to produce enough floating capability, which results in less mobility and considerably less accuracy with dropping the lander. Furthermore, the cost and tech readiness led us to prefer the aircraft.

19.4 Mission profile

The mission consists of five vehicles that operate as a unit to complete the mission objectives. These vehicles are a spacecraft, aeroshell, aircraft and two landers, one of which is stored inside the aircraft and the other beneath the aircraft in the aeroshell that is attached to the spacecraft (see figure 19.2).

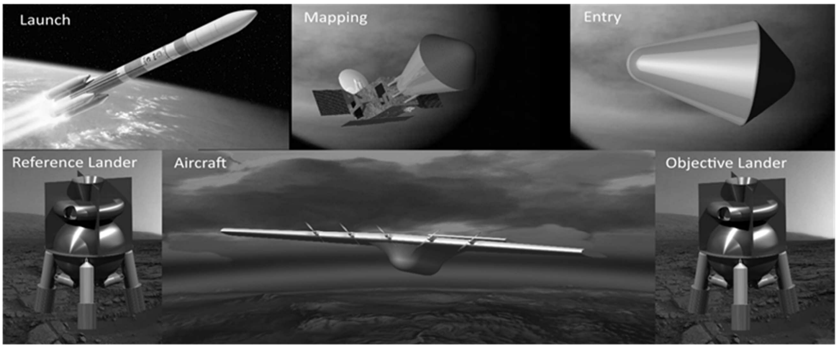

Figure 19.2: Project Matryoshka Concept of Operations clockwise from the top left

After launch in May 2023, the spacecraft and aeroshell will be transferred to Venus in 146 days using a Hohmann transfer that requires a ΔV of 3.51 km/s. Its orbit will have an apoapsis of 37,105 km and periapsis of 385 km, at an inclination of 85 degrees. In that orbit, the spacecraft uses radar to map the Venusian surface for eight months to determine the region most likely to have active volcanoes. The aeroshell will then be deployed such that it arrives at a good location above the Venusian surface while protecting its contents against the heat generated during entry. After a parachute is deployed, the heatshield will be dropped. Then, the reference lander is dropped followed by the aircraft. The aircraft will deploy itself during its descend and level out.

The reference lander will descend to the surface to take reference measurements for the objective lander still on board of the aircraft. The aircraft's flight tracks consist of dives below the clouds to take measurements, followed by climbs up to above the clouds to charge batteries and cool down. When a promising potential active volcano is found, the aircraft will deploy the objective lander that will take in-situ measurements at the surface.

Meanwhile, the spacecraft will continue to take measurements and act as relay station between the aircraft/landers and Earth, in the same orbit, but now with an inclination of 55 degrees. After mission end, the aircraft will land and remain with the landers on the Venusian surface. The spacecraft will eventually burn up in the Venusian atmosphere after it is moved into a decaying orbit.

19.5 Spacecraft design

In order to perform the mission objectives, the spacecraft carries four scientific instruments:

- Synthetic Aperture Radar: for mapping the Venusian surface
- Doppler Wind Lidar: for measuring the wind speeds in the upper cloud layers
- Camera: for taking photographs over a range of frequency bands
- Venus Emissivity Mapper: for detecting local hotspots

The spacecraft also takes the aeroshell containing the aircraft and landers to Venus. At Venus, the spacecraft will insert itself in an elliptical orbit around Venus and start the mission's mapping phase. In this phase the spacecraft will use its Synthetic Aperture Radar, which is 6 m^2 in size, to map the topography of Venus and identify points of interest for volcanism. The mapping data will be compared to the Magellan data: any topographical changes can be proof of recent volcanic activity.

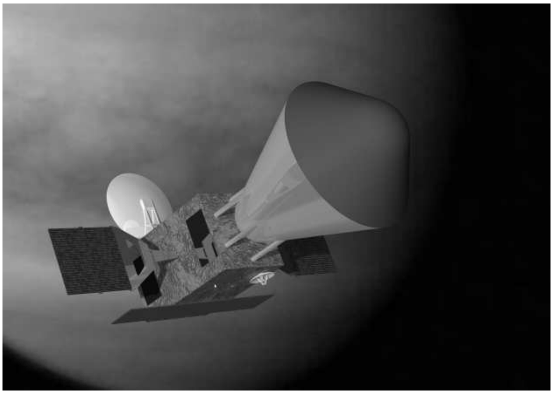

Figure 19.3: Spacecraft design with aeroshell

After the mapping phase, the aeroshell containing the aircraft and landers will be released (see figure 19.3). After this release, the spacecraft will act as the communication relay station for the aircraft and landers to Earth, but also continue taking measurements.

To act as a relay station, the spacecraft will use a 0.8 m diameter X-band dish antenna for communications to the aircraft and lander. To communicate with Earth, the spacecraft has a 2.7 m diameter dish antenna.

The instruments, communications and all other subsystems need power. This power will be provided by two GaAs triple junction solar arrays of 2.94 m^2 each. The mass of the spacecraft is 5,533 kg with propellants and 2,732 kg without.

19.6 Entry and aeroshell design

It is necessary to analyse the aeroshell's entry and decent to ensure that the aircraft will arrive safely in the high altitudes of Venus. The nose radius and cone angle heavily influence the heating and aerodynamic properties of the aeroshell. A larger nose radius is beneficial to the heating properties, while the cone is advantageous for aerodynamic performances. The dimensioning of the aeroshell can be seen in figure 19.4.

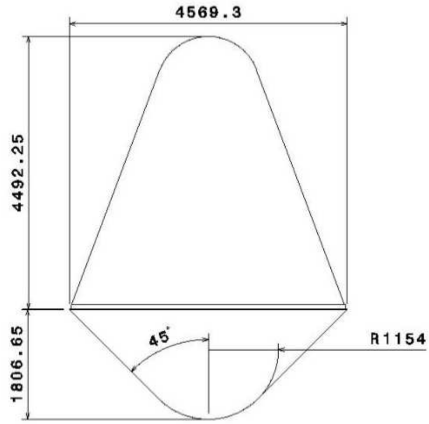

Figure 19.4: Entry vehicle dimensions

For structural reasons, the entry loads should be minimised, therefore an entry flightpath angle of -10° with an initial velocity of 7.2 km/s at an altitude of 260 km is chosen. The resulting maximum acceleration of 26.1 Earth g occurs at an altitude of 172 km. A parachute of 50 m² is deployed at an altitude of 145 km.

Once the aeroshell has reached 50 m/s at an altitude of 93 km, the heatshield is ejected. Once the heatshield is cleared, the reference lander is deployed followed by the aircraft. The aircraft needs to perform a roll manoeuvre to turn right side up.

19.7 Aircraft design

In order to perform the mission objectives, the aircraft carries four scientific instruments:

- Synthetic Aperture Radar: for ground speed determination
- Venus Emissivity Mapper: to map the surface and detect hotspots
- Camera: for taking aerial photographs
- Spectropolarimeter for Planetary Explanation: to measure aerosol particles in the atmosphere.

The desire to fly at lower altitudes to perform more accurate measurements coupled with the extreme Venusian atmosphere imposes challenges on the aircraft that can only be resolved when limiting the speeds, altitudes and cruising times. The aircraft will therefore cycle between an upper altitude of 66.7 km at 2.7 m/s, and a lower altitude of 42.0 km at 23.1 m/s.

The wing planform is constrained by the requirements to have enough available area for solar panels, yet not be too large to limit the viscous drag, while fitting inside the aeroshell. As a result, most of the available surface area on the top and underside of the aircraft is covered with solar cells.

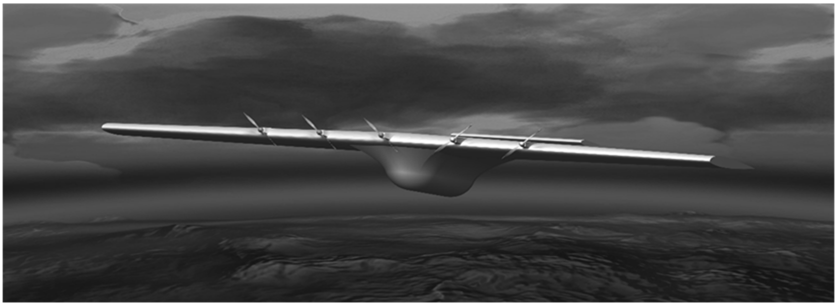

Figure 19.5: Aircraft operating below Venusian clouds

The propulsion of the aircraft will be delivered by five 2-blade electric propellers. Because these require no fuel, the possible mission duration can increase. There is one large propeller at the centre of the wing planform, and four slightly smaller ones located along the wings. Three will be used at the lower altitude track, while only two will be used on the higher track.

Given the power requirements, the aircraft structure must be as lightweight as possible while also being mechanically stable at temperatures up to 2,000 °C , resistant to acid corrosion, and survive potential lightning strikes. In order to keep the interior instruments within 0-800 °C, Pyrogel insulation will be used. The next layer will be the structural composite, PETI-330. An Ultraconductive polymer coating will be used for lightning protection. Finally, the outer Teflon PTFE coating will protect against sulfuric acid.

The aircraft structure was sized by the following load cases: launch, cruise, ascent, descent, entry and turning. Stiffeners and frames were used in the fuselage to prevent buckling. The lander hatch represents a large filled cut-out which imposed a higher safety factor. The wing box is separated into three sections from the root to the chord, that have each been optimised for stringers, spars, ribs and skin thickness.

A stability and control analysis of the aircraft showed that a pure tail configuration was the most efficient considering mass and folding. The horizontal tail will sit on top of two vertical tails attached to booms extending from the main wing. The horizontal tail is thus better configured for the downwash induced by the main wing.

19.8 Lander design

The mission includes two identical landers. One will be deployed directly from the aeroshell that also contains the aircraft and will be used to gather reference data from a non-volcanic Venusian surface region. The second lander will be deployed by the aircraft near or on a suspected active volcano to confirm of the presence of active volcanism.

The Venusian surface is one of the most hostile environments known to mankind. It has a pressure in excess of 90 bar, and temperatures around 500 degrees Celsius. This makes designing a vehicle capable of operating there a serious challenge.

Figure 19.6: Lander design

The lander (see figure 19.6) consists of a main pressure vessel containing the batteries and most of the electronics. It consists of two concentric shells with gas in between. The gas is pressurised to 150 bars, such that the outer shell can be designed for internal pressure, rather than external pressure, while the internal shell can be designed purely for compression, without having to account for the high temperatures and other environmental challenges.

The lander carries two main instruments: a camera capable of imaging in the visual and infrared spectrum and a Laser-Induced breakdown spectrometer. The former gathers visual information from the surface and locates hot-spots, while the latter determines the composition of rocks in a radius of 7 meters around the lander. The optics of the instruments and the laser assembly are located in a spate wheel-shaped housing on the top of the lander that can rotate 360 degrees.

19.9 Conclusion and recommendations

Project Matryoshka aims to unambiguously detect active volcanoes and to characterize the eruption products on Venus, such that the sources and sinks can be identified. After a concept trade-off, it was decided to use a spacecraft, an aircraft and two landers, which as a unit will be able to find active volcanoes if they indeed exist, such that the primary objective can be completed. The mission is planned for launch in May 2023. The overall vehicle characteristics are shown in table 19.1

Table 19.1: Vehicle cost and weight

Vehicle	Spacecraft	Aircraft	Lander (2x)
Cost [M€]	288.5	158.3	93.2
Weight [kg]	5533 (including propellants, aeroshell)	814.9	45.7

Apart from the costs of the vehicles, they also have to be built and tested. For this purpose, a budget of €461.4 million is reserved bringing total mission costs to €999.9 million, meaning that the cost requirement is met. Other requirements such as the use of current launchers, not using radioactive materials and launching before 2026 have also been met. The design therefore meets all of the main requirements.

One lander will be stored inside the aircraft, and the other is stored together with the aircraft in an aeroshell. This aeroshell will be brought to Venus on top of a spacecraft. The spacecraft will carry a camera to map the Venusian surfaces to determine the most likely region to find active volcanoes. In this region, the aircraft will oscillate between altitudes to have a better look at the Venusian surface. A first lander will be dropped to do reference measurements. When a possible active volcano is found, the aircraft will drop the second lander. This lander will use a Visual Near-Infrared camera and a laser induced breakdown spectroscope to unambiguously detect volcanic activity.

Limitations to the current design are handling high temperatures and high risks. Advanced simulation techniques and testing should be used to ascertain the vehicles performance. Furthermore, breakthroughs in power generation, power storage, high temperature electronics, new high temperature and corrosion resistant coatings could improve the design.

20. TubeSat: Suborbital Satellite To Space

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20.1 Introduction

The majority of atmospheric measurements are conducted using sounding balloons, which reach altitudes up to 50 km, and Low-Earth Orbit (LEO) satellites, which can fly as low as 150 km before atmospheric drag makes sustained orbit impossible. This leaves the region between 50 and 150 km altitude relatively uncharted, due to which it has gained the nickname "Ignorosphere".

An attractive solution to this problem is the use of sounding rockets. However, sounding rockets are often relatively expensive. T-Minus Engineering B.V. is a space engineering company manufacturing low-cost sounding rockets with a very short set-up time. Their DART research rocket consists of a solid rocket booster with a so-called passive 'dart' mounted on top. When launched, the solid rocket booster burns for approximately 5 seconds, accelerating the dart to Mach 5.2. At 5 km altitude it automatically separates from the dart

due to the difference in drag. The dart then coasts up to 120 km altitude, where it ejects the payload.
The main challenge of the project followed from the dimensions of the dart. In order to reach these high altitudes the dart has to be aerodynamically very efficient. Therefore, the dart is slim and long and constraints the payload to the dimensions of a tube with a length of 250 mm and a diameter of 30mm. This gave rise to the name of the project, and a new type of satellite, the TubeSat.

20.2 Mission statement and requirements

The mission statement of the project is as follows:

"To design a satellite for the T-Minus DART research rocket that will validate the height of apogee, conduct measurements of the kinematic behaviour at launch and provide 30 seconds of high-definition video footage of its deployment and descent. Additionally, a method/system to retrieve the satellite should be developed."

In order to perform the mission properly, the following top-level requirements have been set:

- The TubeSat needs to be able to transmit all the relevant data before touchdown.
- Everything should fit inside a cylinder with a diameter of 30 mm and a length of 250 mm.
- Commercial off-the-shelf (COTS) components should be used when available.
- The TubeSat should be designed to be sustainable.
- The TubeSat should be able to operate with loads up to 100 g.
- The TubeSat should be able to operate in a thermal environment which will range from −60°C to +120°C.

20.3 Design considerations

In order to make sure a proper design would be delivered, challenges that could be encountered during the mission were analysed. Research was done to find options which could fulfil the main functions (apogee validation, recording of the video and measuring the kinematic behaviour).

Apogee determination
Validation of the apogee is one of the top-level requirements for the TubeSat. Taking just the video will not be sufficient proof that the desired altitude has been reached. There are several ways to determine the altitude. Using the Global Navigation Satellite System was the first option considered. What had to be researched were the speed and altitude limitations which are set on the GPS chips. Radar and Doppler shift were also considered, but these systems would require multiple ground stations. Integration of the accelerations was also considered, but integrating scientific data would lead to errors which were not acceptable.

Kinematic behaviour during launch
Finding the characteristics of the DART rocket involves measuring its rotational speed, attitude and accelerations. To do this an Inertial Measurement Unit (IMU) needs to be implemented. However, these units although small would still take up quite a lot of space inside. The alternative would be using gyroscopes, magnetometers and accelerometers. These three components are all PCB mounted chips but they require more effort to integrate into the complete system.

Camera system
Before choosing a camera system, it is important to know what has to be filmed. T-Minus Engineering B.V. wanted the ejection to be part of the video, so the camera would have to start recording right before ejection and it had to face in the downwards direction of the dart. This led to challenges for the integration of the camera, since the camera would need a clear field of view and had to be shielded from the ejection system. Next to what to film, it was important to make sure

the video would be pleasant to watch. However, the quality of the video will depend on the descent. Large vibrations during descent would lead to a shaky video which is undesired. Tumbling and rapid spinning of the TubeSat would also lead to an unwatchable video.

Data handling

The video footage, the apogee validation and the kinematic behaviour during launch make up the relevant data which will be generated by the TubeSat. The data should be stored to acquire it after retrieval of the satellite. However, since there is the possibility that the TubeSat may not be retrieved at all, all the data should be transmitted as well. Most launch stations use the S-Band (2200-2400 MHz), so this was seen as an option. The transmitting of the data led to challenges in the material selection, since metals would interfere with the transmitted signals. It also led to requirements on the stability and the descent time in order to be able to send all the data.

Descent

The descent of the TubeSat was analysed in order to find out whether there were additional constraints. During the simulation, it turned out a lot of heat would be generated during descent. This heat would be able to destroy all the electronics. This led to requirements for isolation of the nose.

Stability

The DART is spin stabilised during ascent. Due to this, the TubeSat would also be spinning at ejection, which would be detrimental for the quality of the video. A method had to be found which would enable de-spinning of the TubeSat. Besides this a system that would prevent tumbling had to be designed in order to point the transmission antenna of the TubeSat down during descent.

Recovery

In order to find the TubeSat after landing several things can be done. The last known GPS coordinates can be used also a beacon can be placed in the satellite which can send a signal every few seconds.

Other ideas were the make the TubeSat glide back to the launch site or the use of a strong smell so that dogs can be used.

Sustainability

As for any good product, sustainability needs to be taken into account. For our design, sustainable strategy chosen was Eco-design. Eco-design focuses on all the life phases of a product, from the extraction of raw materials all the way until the disposal and possible recycling of the product. The satellite is very small and needs to survive high launch loads, different temperatures and environments. Therefore, it is challenging to follow a sustainable approach. The components, materials and techniques chosen have to firstly be able to fulfil the requirements for the mission, and secondly, be sustainable.

20.4 Concepts and trade-off

To fulfil the mission statement and all the requirements, different concepts were developed. Due to the limited space in the satellite and the preference for COTS components a function had often a clear best solution. The function which had a lot of freedom for design was the control of descent. The concept trade-off for this function is shown below.

PaSoS - Parachute Suborbital Satellite

This option uses a sort of parachute to control and slow the descent.

S3 - Samara Suborbital Satellite

In this option, the satellite will unfold a part of itself to obtain the shape of a samara/maple seed leaf. This will slow down the descent, like with its natural counterpart.

DISC - Dart Initiated Shuttlecock

In this option, the TubeSat claps out feathers of steel after ejection, to take a shuttlecock shape. This naturally point keep the satellite in a stable position during the descent and will slightly increase the fall time.

Figure 20.1: Visualisation of initial concepts

In the trade-off the criteria focused on volume, producibility, performance and development risk. From this trade-off, it was decided to use PaSoS, because it is a proven system which is simple and least prone to error.

After this first trade off, it was found out opening the parachute at high speeds causes problems. Therefore, a decision was made, to make a mechanically deployed parachute. With this in mind a number of concepts were created to function as a drag device, all of which are suspended behind the TubeSat. It was also decided that we do not need reaction wheels to point the camera.

ShuttleChute

This parachute is shaped like a shuttlecock, and is inside the dart with the blades folded. When ejected, a spring system pulls the blades open after which they are locked in position, like an umbrella system.

Veggie Steamer and Reversed Veggie Steamer

The Veggie Steamer is a concept which is similar to the ShuttleChute, but uses the folding mechanism of a foldable vegetable steamer. The blades are interlocked in closed position and a different spring system pushes the blades open. The difference between the Veggie Steamer & Reversed Veggie Steamer is that it is mounted upside down, like a rigid parachute.

Hyperflo

The Hyperflo concept is based on the Hyperflo parachute, which was developed for supersonic speeds. The Hyperflo parachute is shaped like the cap of a plastic bottle, where the sides have a low porosity and the top has a high porosity. It does not need any deployment and already has the final shape a Hyperflo parachute would have, by using a rigid material. The difference is however that the Hyperflo can at most have the same diameter as the TubeSat.

Figure 20.2: Visualisation of parachute designs

In this trade off the focus was on volume, producibility, performance and complexity. Using these criteria the winner was Hyperflo mainly due to its simplicity. Also the fact that no deployment system is needed, the Hyperflo will be put on top of the satellite, once ejected the difference in drag will make the Hyperflo deploy automatically.

20.5 Detailed design process

After the Hyperflo concept was chosen a detailed design of the TubeSat started.

Electronics

The implementation of the electronics was one of the most challenging things during the project. All the electronics were controlled by a micro-controller. Moreover, all the components were put on various printed circuit boards (PCB’s). In total, five PCB’s are used for the implementation of all the electronics. The microcontroller was implemented on the main PCB, which is the biggest of all.

Attitude determination

For the determination of the apogee, it was decided to use a GPS module. It is possible to turn off the limitations of GPS when it will be used for scientific purposes. The GPS module was implemented on a single PCB, in order to increase the freedom to place the GPS module. Since it is important to be able to make contact with a GPS-satellite, the antenna of the module should be outside of the TubeSat. It was decided to make a hole in the side of the TubeSat in order to give the antenna the possibility to receive the signal. Because the complete GPS module is on a single PCB, it was possible to place freely

Video footage

Taking a 30-second HD video is one of the top level requirements of the mission. Initially, complete cameras such as the GoPro were considered, but these cameras would take up too much space, which would leave little room for the other electronics. Therefore, it was decided to use single components in order to take the video. A camera module was used which would be implemented on a single PCB. However, the video that the camera would take would be uncompressed. This would lead to difficulties for the transmission system, since the video would be too large. A video compression chip, which was mounted on the main PCB, is used to solve this issue. The bit rate of the video would then be about 5 Mbps for the video.

Storage

Storing the relevant data is beneficial, for extra data can be generated which cannot be transmitted. When the TubeSat is retrieved, this additional data can be extracted. A micro-SD card is used, which has a connector with the main PCB.

Communications

The data would be transmitted over the S-Band (2200-2400 MHz), since this is available at most ground station. Moreover, the largest bandwidth would be provided and a high transmission power was allowed when using this band. However, there were no full modules that would transmit the data. As a consequence, the transmission system would be made using individual components. A digital-to-analogue converter and a I/Q-modulator were used to generate the signal and put it on the correct carrier frequency. Both these components would be mounted on the main PCB. A transmitter and antenna combination, which are normally used in the tracking of missiles, were used in order to amplify and transmit the signal. These two were mounted on a separate PCB in order to get more freedom in the placement, for no metallic parts should be in front of the antenna. This would lead to interference with the signal.

Power system

A 3.3 V battery was used in order to power all the electronics. Off course, not all components are able to work at this supply voltage. In order to cope with this, several DC-DC converters were used to get the right voltage for the various electrical components. The DC-DC converters were mounted on the main PCB.

Structure

The TubeSat consists of two elements: the main body and the Hyperflo. The main body is used to house the PCB's, the yo-yo de-spin mechanism and the attachment system for the Hyperflo. The Hyperflo is used for the pointing of the TubeSat and is discussed in more detail in the next section. The main body is built up from two halves and a nose cone. The two halves are made of steel and are bolted together. A ceramic called Zirconia is used for the nose cone instead of steel in order to prevent interference for the transmitted signal, while also being able to cope with the large amount of heat which is generated during descent. To ensure that the camera has a clear field of view, two quartz lenses are used. These two lenses will also create an air barrier, which will be sufficient for the insulation.

Attitude control

The yo-yo's are used to de-spin the TubeSat after ejection. These are stored in the back of the main body. Moreover, an attachment system for the Hyperflo is stored behind the yo-yo's. A shark-line is used to attach the Hyperflo with the main body of the TubeSat.

Aerodynamics

There a couple of aspects which influence the aerodynamic characteristics of the TubeSat. These include: shape of the nose cone, shape of the rear and the Hyperflo. The shape of the nose was taken to be a hemisphere, since this would be advantageous for the heat transfer. Designing the shape of the rear was influenced by the generation of wakes behind it. These wakes would influence the performance and the positioning of the Hyperflo. Optimising the shape of the rear in order to minimise the wake was beyond the scope of the project. Instead, it was assumed that the negative influence of the wake would be overcome by placing the Hyperflo sufficiently well behind the rear.

Recovery system

The recovery system consists of a small additional transmitter acting as beacon. This transmitter is placed in the side of the TubeSat. Together with the compatible receiver and the last known GPS location the recovery time should be reduced greatly. The beacon signal will be visible from about 500 m to several km depending on the vegetation.

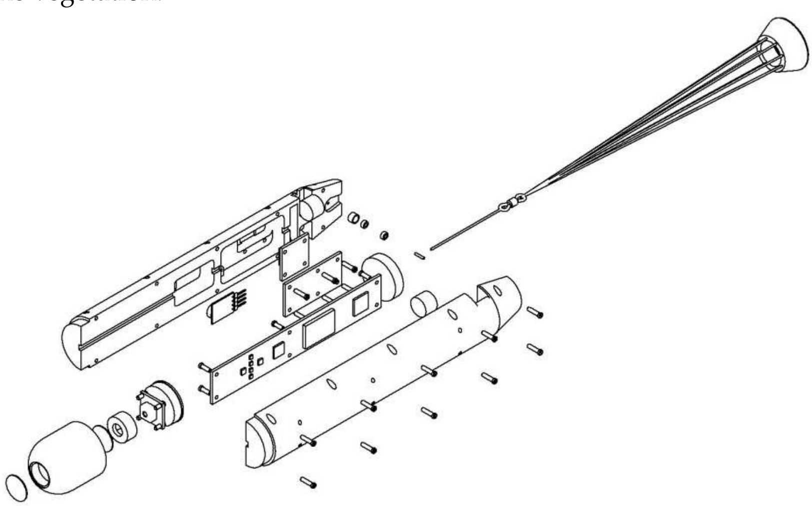

Figure 20.3: The final lay out with all the components

Not all the following things are discussed as this summary is page limited, but in table 20.1 the final lay-out of all the electronics can be seen.

Table 20.1: All PCB's with their designated components

PCB nr.	Comprises of:
PCB 1(main)	Microcontroller, Image Processor, MicroSD connector, Modulator, Digital-to-Analogue Converter, DC-DC converters
PCB 2	Accelerometers, Gyroscopes, Magnetometers
PCB 3	GPS antenna module
PCB 4	5.0 MP camera module
PCB 5	S-Band transmitter, S-Band antenna
PCB 6	Beacon module

20.6 Conclusion and recommendations

The TubeSat should validate the DART rocket of T-Minus. It does this is several ways, firstly it will record the apogee height using the GPS combined with the acceleration data. Secondly a 30 second HD-video will be made of the ejection, earth and space. Thirdly it will also record the behaviour of the DART and thus validate its behaviour during launch. All this data will be send to the ground using a patch antenna.

To quality of the video is ensured by de-spinning the satellite right after ejection. Than during descent the stability of the satellite is provided by the Hyperflo. The Hyperflo will automatically separate from the TubeSat due to the difference in drag. The Zirconia and quartz glass nose protects the electronics inside against the high temperatures encountered during descent.

Of course still work has to be done, therefore the following recommendations have been made:

- A simulation of the descent was created, however in simplified form. It is recommended to extend this and update with more detailed parameters of the satellite, which may result in the need for small alterations in the design. In particular the stability behaviour of the satellite and Hyperflo system should be modelled in more detail. CFD is an option for the aerodynamic characteristics.

- The electronics system should be designed in more detail, which has not been done yet, as the team does not have expertise in this area.
- A vibrational test should be performed, using a vibration bench.
- A more detailed thermal analysis should be performed, to check that the heat transfer to the components does not increase the temperature to an unacceptable value.

21. LYNX SPACECRAFT - A DEPLOYABLE UHF TRANSPONDER PAYLOAD

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21.1 Introduction

In a few years, XCOR Aerospace will provide the space market with suborbital spaceflights. Their spacecraft, the Lynx Mark I, takes off and lands horizontally using its reusable rocket propulsion system. Next to the pilot and a passenger, the Lynx Mark I has the ability to bring along several payloads. The use of one of the payload bays is offered to the Royal Netherlands Airforce (RNLAF) as part of the Dutch Ministry of Defence. This outer payload bay can carry a deployable 2U-unit (10x10x22 cm) and can be used for any kind of scientific purposes.

Until today, the Dutch Ministry of Defence has not owned space assets that are used for its military and civilian operations. However, it does have a space section at the RNLAF, the space security centre, which investigates possible innovative solutions for intelligence gathering. Due to new technologies and cheaper possibilities, it will be possible for the RNLAF to develop their own satellite within a few years. The opportunity offered by XCOR will be used to explore the possibilities for the Air Force in a space environment. For this first space asset, the RNLAF wants to build a deployable payload to perform a technical demonstration mission of a long-range Ultra High Frequency (UHF) transponder. This article will elaborate on the applied design process to arrive at the final design of this transponder payload, called the ICARUS+.

ICARUS+ stands for Innovative Compact Autonomous long-Range UHF-transponder System. The name is derived from the myth of Icarus, who flew too close to the sun and fell down from the sky. ICARUS+ will prove that with the current technology the limit for flying 'too high' does not exist anymore.

21.2 Mission outline

The purpose of this mission is to demonstrate the feasibility of sending a payload with the Lynx Mark I up to an altitude of 60 kilometres and deploy it in the surrounding environment. At this altitude the density is very small and therefore almost vacuum conditions are present. However, the gravitational forces are still present. Furthermore, the payload shall weigh no more than two kilogrammes and fit into the 2U external payload bay C of the Lynx Mark I. It shall be controllable and fly as far as possible. Thus, maximising its range by initiating its optimum gliding path at a maximum altitude is an essential function of the design.

Taking these conditions into account, several requirements are established for the design. Below, a list of the driving requirements for this design is stated. The design together with the box is referred to as the payload. However, when one of the requirements solely applies to the design carried by the box, it will be called the system.

- The payload shall have a size of 10x10x22 cm.
- Each payload shall have a total cost lower than 100,000 euros.

- The payload shall be able to operate in a -70 to +30 °C atmospheric temperature range.
- The payload shall be able to operate in an absolute atmospheric pressure of 0.002196 N/m^2.
- The system shall be able to deploy a flying mechanism.
- The system shall be able to control its attitude.
- The system shall be able to communicate with the ground station.

Once deployed by the Lynx Mark I, the design will immediately deploy from the 2U box and will start its flight path. The initial conditions for the flight path of the design result from several aspects. Due to the low density at 60 km altitude it will not be possible to start a gliding flight immediately. Furthermore, the Lynx has a certain horizontal velocity at the moment of detachment. That velocity can be approximated by analysing the related flight path of the Lynx Mark I and will have an impact on the flight path as it influences the initial conditions. Another important aspect is the altitude at which the payload will be stable and ready for flight. At this altitude, the ICARUS+ starts flying by performing a pull-out manoeuvre. After this pull-out, the design will be able to gain enough lift to perform a gliding flight at a nearly constant flight path angle.

21.3 Concept selection

After the requirements were set the concept selection process started. In this section, the five concepts that were elaborated on are given first, followed by the trade-off on these concepts.

With the use of design option trees a lot of possible concepts were made. These were narrowed down to five promising concepts which were taken to the next design phase. The concepts can be seen in figure 21.1. The Mini Space Shuttle resembles the Space Shuttle in appearance. Its wings can fold into the 2U box whilst still having the capability to carry a payload in the fuselage. The Parachute concept first deploys a drogue parachute to reduce speed. This parachute will be released once an optimum velocity is reached. Following, a ram-air parachute will be deployed in order to produce lift. The third concept resembles an owl when seen from above, therefore it is referred to as the Owl. Since the Owl has wings with triple the length of the 2U box, a large aspect ratio is achieved. This is done by rotating the wings and then unfolding them in both directions. The Hang Glider, seen in the bottom left corner of the figure, is a delta wing glider which is folded into the 2U box. When a specified velocity (low enough for the glider to operate) is reached the Hang Glider is released from the box and deploys with a mechanism similar to a hand fan. The last concept can be seen in the middle of the figure and is called the Flying Wing. The Flying Wing can be fit in the 2U box by using the two hinge lines in the middle of the design.

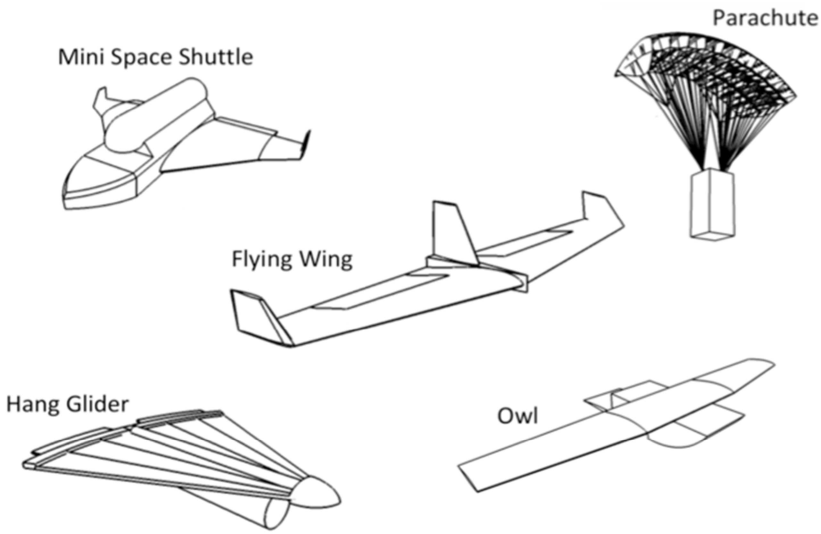

Figure 21.1: Design concepts for a deployable UHF transponder payload

These five concepts were elaborated on for several weeks. During a trade-off process each concept was graded on the following subjects: range, controllability, reliability, stability, structures, configuration, thermal control, cost, sensitivity of the design and sustainability. Since each of the five concepts scored the same on the latter three categories they were left out of the trade-off. When the trade-off was completed, one concept clearly stood out: the Mini Space Shuttle. This concept became thus the precursor of the ICARUS+.

21.4 Final design

The performance analysis of the final design process was split up into several engineering fields. Multiple iterations were done until the final design was reached, which can be seen in figure 21.2. In the figure the design can be seen completely deployed and how it fits in the 2U box. In this section the performance analysis for the final design is given and all the design features seen in the figure are explained. First of all aerodynamics is elaborated on, followed by stability and control and structures. Afterwards, a thermal analysis is done and the electronics engineering results are given. As last the flight path of the ICARUS+ is given with its final range and endurance.

Figure 21.2: Final design ICARUS+, folded and deployed

Aerodynamics

The main decision that needed to be made in this field is the type of aerofoil for the wing and canard. These aerofoils were both chosen for optimal flight. During the airborne time of the ICARUS+ both supersonic and subsonic flight regimes will be experienced. The differences between super- and subsonic wings are very significant. A supersonic wing is thin, has a large sweep angle, small aspect ratio and no winglets. Contrary, a subsonic wing is thicker with a small sweep angle, high aspect ratio and winglets. A trade-off was made between these features and finally a wing with a dual sweep angle, no winglets, large aspect ratio and subsonic aerofoil was chosen.

The chosen aerofoil was based on the fact that most of the flight path (95%) will be in a subsonic flight regime rather than supersonic. The aerofoil chosen for the wing is the NACA-43012A. This aerofoil has a good gliding performance and performs well at low Reynolds numbers. For the canard a symmetric aerofoil was chosen, the Biconvex 8% t/c. This aerofoil has a good supersonic performance. Based on this, the lift and drag coefficients of the ICARUS+ over the entire flight path could be determined. Lastly, it was determined that the complete ICARUS+ has a negative Cm-αcurve, which is desired in order to obtain natural stability.

Stability and control

Regarding stability and control, several main categories had to be taken into account. These are the planform, subsonic and supersonic flight as well as the free fall stability. First of all, the planform of the wings needed to be decided on. Due to stability problems in previous design phases, it was decided there will be a fixed planform given from a stability point of view. The decision was made to have a compound delta wing, or in other words, a double delta wing. This means that there are two different angles of sweep. Consequently, the stall angle of attack increases and the vortex lift will be more controlled. Furthermore, the aircraft becomes more trimmable.

As mentioned before, the ICARUS+ was mainly designed to fly at subsonic speeds. However, supersonic control is still needed since the pull-out occurs at supersonic speeds. Therefore, a movable canard was added to the design as stated before. Next to the benefits in supersonic flight, the canard also adds to stability during the entire flight path. In order to obtain controllability at subsonic speeds as well, elevons were added to the wing. Due to the negative Cm-αcurve it was already confirmed that the ICARUS+ is stable. Lateral stability is provided by a vertical tail surface while lateral control can be reached by moving the elevons separately.

When the payload is going to be released it will fall down and start rotating depending on its initial orientation and rotational velocities. The ICARUS+ has to be able to stabilise itself in order to start flying. This can be done by using its control surfaces in an optimised manner with the help of a flight control computer. It was decided that the

ICARUS+ should be stable at 40 km at lowest in order to be able to fly as far as possible. This process was simulated and up to a certain rotational velocity given by the Lynx during the release process, the ICARUS+ will be able to stabilise itself.

Structures

The first loads that the design must survive, occur during the launch. Besides the 6g force that arises during launch, also some vibrational accelerations occur and have to be considered in the design process. However, in order to analyse those vibrations a coupled loads analysis must be performed, taking the natural frequencies of both the launch vehicle and the design into account. The loads transferred from the Lynx Mark I could not be specified up to this point. To account for this uncertainty, an optional isolation system can be implemented between the payload and the Lynx. This system can change the Power Spectral Density (PSD) that will be passed on by the Lynx and will therefore change the effect of the vibrations.

After launch the design will be deployed. As stated before, it is fitted within a 2U box. In the first stage of deployment, this box has to be ejected from the Lynx. A mechanism for this stage is built in the Lynx itself. Once the 2U box has been deployed from the Lynx, the design will be released from the box. This is done by means of a special release mechanism that makes use of a spring system and the deployable titanium UHF antenna. Once deployed from both the Lynx and the 2U box, the design will unfold into its final configuration. This is done by means of a special hinge mechanism which is loaded with a spring while in the box. This mechanism is shown in figure 21.3. The design is mainly made out of titanium. This is widely available for aerospace structures and will be able to handle all the forces arising during both launch and flight.

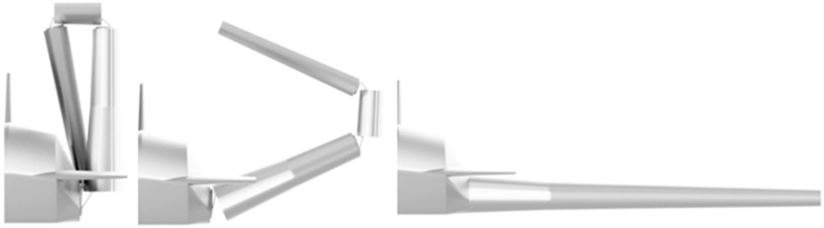

Figure 21.3: Unfolding mechanism of the ICARUS+

Thermal control

The ICARUS+ is subjected to a wide range of temperatures. The payload bay is exposed to the outside atmosphere during the launch, but the rocket engine of the Lynx is also located right next to this payload bay. After deployment the ICARUS+ is only in contact with the outside air. This temperature envelope depends on two factors, the altitude and the heating due to friction and compression of the surrounding air, especially during supersonic flight. The first one is determined by the flight path and the second one is estimated with the flight speed.

It can be concluded that the design will need an isolation layer within the fuselage in order to protect the electronics on the inside. The minimum temperature only depends on the insulation thickness and not on the initial temperature. Taking the limited internal space into account an insulation thickness as small as possible is desirable.

Electronics

Of course, the ICARUS+ is also equipped with an electrical system, focusing on communication, data handling, actuation, navigation and power. During launch, the electrical power is coming from the LYNX itself via a power connector, indicating to the computer that it is not deployed yet. As long as the Lynx provides power, the 3 V battery that serves as a power source during flight will be charged. After the ICARUS+ has been deployed from the Lynx, it will provide a power supply to the other components for two hours.

As the flight path only lasts 70 minutes, there will still be enough power in order to track and locate the location of the ICARUS+ after touchdown on the ground. In order to be able to track the design, a GPS and IMU sensor is incorporated into the fuselage. As the shell is made of titanium a small composite patch on the topside has been designed, allowing for an aerodynamic placing of a patch antenna for the sensor system. The second antenna which is used to send and receive signals at a frequency of 395 MHz is extendable through the back of the ICARUS+.

This antenna is also assisting deployment from the box by means of a spring, as mentioned before. This deployment is done by using a burn resistor that will heat up and burn through the wire compressing the spring of the antenna, which will then spring out and simultaneously disconnect the design from the 2U box. The antenna is also connected to the main payload of the design, the UHF transponder. Furthermore, in order to control the attitude of the aircraft, four servos are providing movability to the control surfaces, based on the input of the board computer.

Flight mechanics

All the previous calculations stated resulted in a final flight path. The flight path was calculated in four different stages. First of all, there is the free fall phase, as stated in the previous section about stability and control. Once the ICARUS+ is stable, which has to be above 40 km, the ICARUS+ can start a pull-out. This pull-out will be followed by a transitions phase which will finally result in the optimal glide phase. Taking those phases into account, together with all the relevant properties from the ICARUS+, this results in a range of 230 km with an endurance of 70 minutes. The flight path can be seen in figure 21.4.

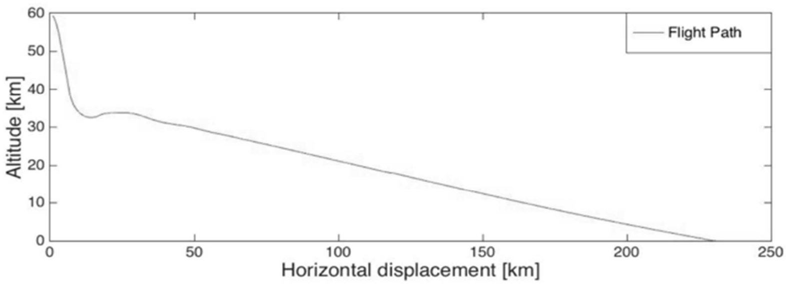

Figure 20.4: Flight path of the ICARUS+

21.5 Conclusion and recommendations

In collaboration with RNLAF and XCOR the ICARUS+ was designed by the DSE project group S16 from the Delft University of Technology. The ICARUS+ is a deployable long range UHF-transponder which will be dropped by the Lynx Mark I from XCOR at an altitude of 60 km. After a design process of 10 weeks the final lay-out was reached, which is a double delta wing-canard configuration.

The main limiting requirement of the ICARUS+ is the size limit of 2U. The ICARUS+ was designed in such a way that a maximum wing span can be reached by using special designed hinges in optimum locations. As follows, when the payload is fully folded, it fits into a 2U box. However, a wing span of 360 mm is reached in deploy configuration. The total mass of the ICARUS+, including its box, resulted to be 0.980 kg. The aerofoil used for the wing is the NACA-43012A, which has a good gliding performance at subsonic flight conditions.

A double delta wing was chosen due to the preferable conditions at high velocity flight regimes and due to stall delays. Furthermore, a movable canard was added to increase stability. The canard was made movable and elevons were added to increase controllability. The material chosen is titanium since it is very strong and has a good resistance against high temperatures. In the ICARUS+, a UHF-transponder with antenna is integrated. The ICARUS+ has the ability to receive and send data. However, it has been found that the customer will need to make a trade-off between quality and the budget, as the estimated cost is exceeding the set budget of €100,000. Finally, it was found that the final range of the ICARUS+ is 230 km and the corresponding flight time is 70 minutes.

Recommendations for the future are mainly to develop more detailed design performance models, that is, models without assumptions which are thus closer to reality. Furthermore, more testing is needed on the design in order to reduce the risks and to validate the entire design. More specific recommendations are that supersonic flight needs to be looked at in more detail and the box from which the ICARUS+ will be deployed needs further designing, mainly on its maintenance procedures.

With everything analysed it can be concluded that the ICARUS+ will fulfil its purpose and it will show that the limit of flying 'too high' is indeed off the table.

22. Floating Wind Turbines: The Future of wind energy?

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Project tutor: dr.ir. A.C. Viré

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22.1 Introduction

As the global demand for energy continues to increase, the environmental footprint of fossil fuel power plants rises. The world's fossil fuel reserves gradually deplete and the search for renewable energy sources goes on. Wind power has proven to be a reliable renewable energy source and, as time progresses, is expected to become more important.

As the prevailing winds on sea are stronger and steadier than onshore, the next step in wind power lies in ocean-based wind turbines. This means more power can be generated while removing the aesthetic problems oppressing their land-based counterparts. However, the majority of interesting locations for offshore wind power plants are positioned far from the coastline. When the depth of the ocean is over 50 m, the traditional bottom-founded wind turbine is no longer economically viable. Hence, the need arises for cheaper and more innovative solutions in the design of offshore wind turbines. The solution is a cost-effective floating wind turbine design, eliminating the need for rigid foundations.

22.2 Project objective and requirements

The objective of this project is:

"To design an offshore wind energy system placed on a floating support for harnessing wind energy in deep sea. The design will cover all aspects of the system. Including wind energy system, support structure, wind/wave climate considerations, and connection to the grid. Innovation in the design of the wind turbine and floater are encouraged ."

From the project objective the following mission need statement was identified:

"To prove the technical feasibility and economic viability of floating wind turbines that produce at least 6 MW while in water depths of at least 100 m."

Main requirements:

- Rated power of at least 6 MW
- Levelised cost of energy below €130 per MWh
- Suitable for water depths of at least 100 m
- Positioned 20 km offshore
- Operational life of at least 25 years
- Shall survive a 50-year storm

22.3 Concept trade-off

Different floater designs and wind energy systems were initially assessed separately. The concept generation and trade-off were done for the three main subsystems, namely power generation (how the kinetic energy of the wind is transformed to mechanical energy), power conversion (how the mechanical energy is transformed to electrical energy) and stability (how the system is stabilised). These concepts were not only assessed on manufacturability, reliability, efficiency and maintainability in a RAMS analysis, but also ease of assembly, cost and innovativeness are taken into account.

Using general engineering sense 21 system concepts were generated. These consisted of different combinations of power generation type, power conversion type and floater type. After further analysis and a second trade-off, three concepts remained. These concepts are shown in figure 22.1.

A single rotor Horizontal Axis Wind Turbine (HAWT) mounted on a barge with ballast, a multirotor HAWT using hydraulic transmission mounted on a spar buoy and finally two H-rotor Vertical Axis Wind Turbines (VAWT) mounted on a barge.

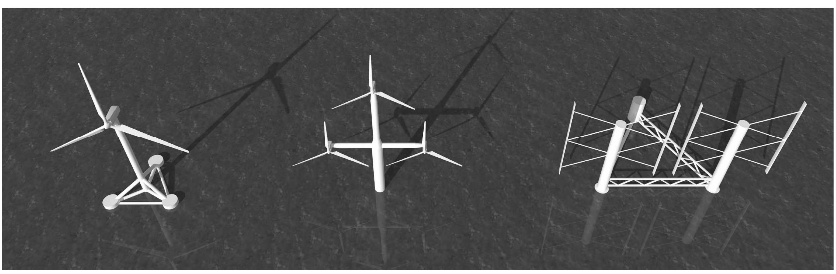

Figure 22.1: 3D renders of the three concepts

A final trade-off resulted in the selection of a concept that slightly differs from the ones selected in the previous trade-off. The recommendation was made that multirotor configurations would be beneficial when upscaling the design. The final concept was therefore chosen to be a twin rotor HAWT using hydraulic transmission mounted on a barge with ballast underneath each outside pontoon. The system had a combined rated power output of 10 MW; two rotors of each 5 MW. The main aspects of the final design are elaborated in the next section.

22.4 Final design

After the trade-off process, detailed design began of the twin rotor wind turbine generating 10 MW. In the next sections different aspects of the detailed design are presented.

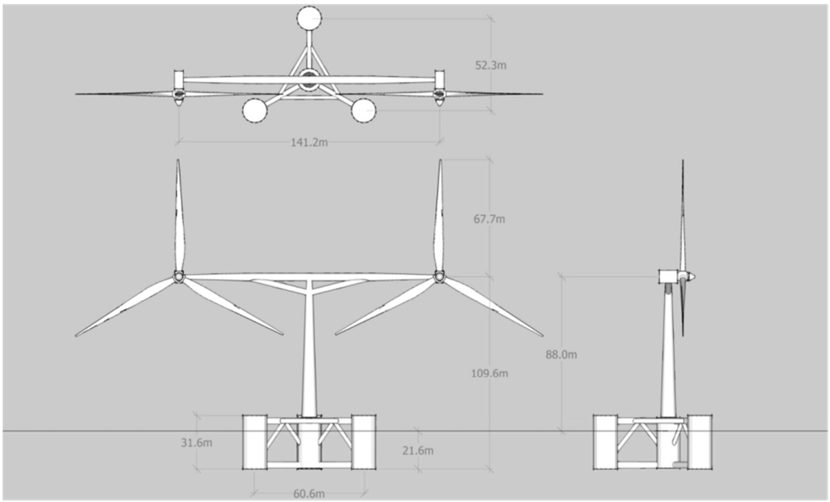

Figure 22.2: Floating wind turbine Aeolus

Blades

One of the most important aspects of the system is the blades. Their main function is to convert kinetic (wind) energy to mechanical energy which can subsequently be converted to electrical energy by the generator.

In HAWT applications, the blade speed increases throughout the radius. This, in combination with the constant incoming wind speed, results in a decreasing angle of attack over the radius. Therefore, it is preferable to install a different aerofoil at every radial location of the blade to maximise efficiency. Unfortunately, this is not possible due to manufacturing constraints and therefore a limited number of sections is used. The aerofoil type for each section was selected based on their performance, which resulted in a blade geometry as shown in figure 22.3.

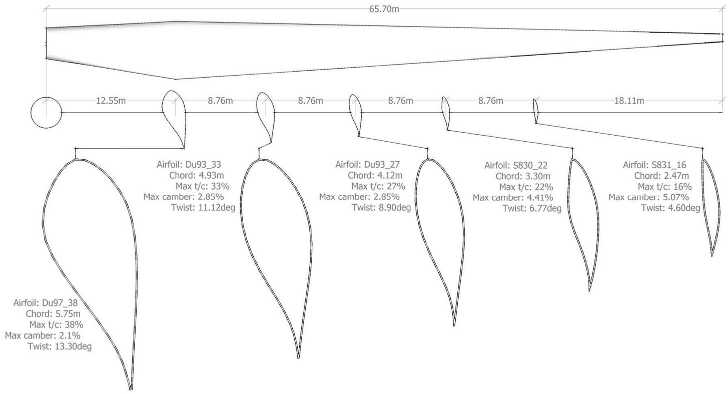

Figure 22.3: Blade geometry

The blades were optimised using software called Qblade. After multiple iterations, this process led to a rotor with an efficiency of about 0.5 (0.593 is the theoretical limit according to Betz's law). The rated wind speed is 11 m/s with a corresponding annual energy output of 27,491 MWh for one rotor.

After having designed the blade for aerodynamic performance, the internal structural design took place. When analysing the blade it was important to know where the highest stresses occur. This meant the type and magnitude of loading was determined over the whole radius. In the case of the Floating Wind Turbine (FWT), bending loads causes the highest normal stress. Also tip deflection had to be taken into account. This means that the tip is only allowed to deflect 2 meter before striking the tower and causing failure. With this in mind an optimal design had to be found which could resist the stresses due to bending while keeping the tip deflection as small as possible.

A wing box approach was used to design the internal structure. Discretising the blade into 'n' sections and analysing each section for normal and shear stress. The thicknesses of the spar and webs were increased until each section could withstand the loads. Subsequently the maximum tip deflection was checked, apparently it seemed that the tip deflection was the driving parameter. Despite the fact that the blade could withstand the loads, the thicknesses had to be increased to avoid tower collision. Finally, this resulted in a blade mass of 24 tonnes which only differs 4.8% from reference blades.

Tower

The tower was designed for three different load cases: nominal operation, extreme storm and asymmetric loading. The first load case which is evaluated is the working when full rotor thrust is available. Secondly, the load case of an extreme storm is assessed in which the rotor drag is leading. Finally, the asymmetric case was considered, where only one rotor is operating, causing a large torque on the tower.

Several structural concepts were compared on weight, complexity and flexibility. Complexity is based on the amount of elements and the chosen assembly process. Flexibility is measured in terms of how easily the yawing system could be integrated in the tower. For instance a yawing system at the bottom of the tower causes difficulties as the bearings must withstand the bending moment. The possibility of placing the system higher, where the moment is smaller, simplifies the problem. Finally a T-structure with struts was chosen.

The structure is made of steel S355. An optimisation was done to get the lowest weight by varying the location of the struts, the radii of the arms, the tower and the sheet thicknesses. Taper was also included for both the tower and the arms. The optimisation was done by using Python and took into account the Von Mises stresses caused by normal forces, shear forces and torsion. Also column buckling and skin buckling were considered. A safety factor of 2.0 was included and the structure was verified with Matrixframe afterwards.
When looking at the tower one might doubt whether the struts are placed at the optimum location. A strut angle of approximately 45 degrees was expected prior to the optimisation. However it has been verified that this was the lightest design within the boundaries. It can be explained by the fact that more vertically placed struts take a higher percentage of the loads (by relieving the arms) and also have a higher length. Column buckling is then critical.

Floater and stability

A floater is designed to generate the required buoyancy and stability. Six concepts were generated and assessed on two structural aspects: rigidity and weight. A preliminary assessment was made using Finite Element Method software Matrixframe. The choice was made for a very light concept, though requiring improvements in rigidity. This concept consists of three pontoons placed in a triangular configuration and one pontoon in the centre to install the turbine. These four pontoons generate the required buoyancy.

Static and dynamic stability analyses lead to an optimised floater size. For the static stability, the restoring moment generated by the system when disturbed was tweaked using the pontoon radius and floater spacing, while optimising with respect to cost. Dynamic stability of the system was assessed based on the natural periods (and natural frequencies) of the system with respect to its degrees of freedom. Degrees of freedom of interest are the heave, roll and pitch directions. Based on environmental conditions, the natural periods for all directions should be outside the range of 11.0 to 19.2 seconds. The natural periods of the system were influenced by adding ballast to the pontoons and changing the floater spacing. This process finally resulted in a pontoon radius of 6.8 metres, floater spacing of 60.6 metres and ballast mass of 9,000 tonnes.

With the above presented concept and stability constraints, it was possible to move on to the detailed design phase. Here, the concept was assessed for two load cases. These were the nominal load case, in which thrust was considered at the rated wind speed of 11.7 m/s, and the asymmetric load case with one operating rotor. Furthermore, the truss forces were investigated for various angles of yaw of the wind turbine. Finally, the truss structure was optimised for buckling and yielding of which the latter was the predominant failure mode.

Power conversion

Common wind turbines are equipped with a gearbox which mechanically transports energy at the optimal ratio, to achieve highest energy output. These gearboxes are heavy and fail often. The remedy to this problem is Digital Displacement Hydraulic Transmission (DDHT). The working principle is simple: rotational energy from the rotor moves a radial piston positive displacement pump and pressurized hydraulic fluid. The high pressure (300 bar) hydraulic fluid flows to the hydraulic motor and is converted back into mechanical energy. A low pressure fluid line connects the motor back to the pump. The last step is the conversion of the rotation into electricity by the generator. Figure 22.4 shows the general layout.

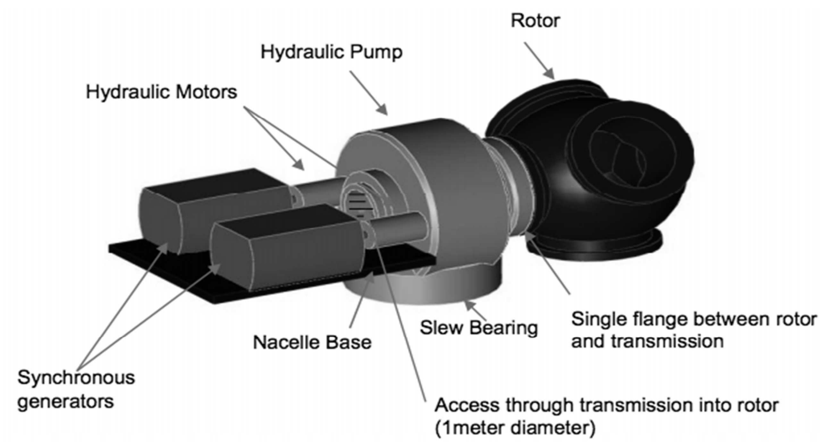

Figure 22.4: General layout of the DDHT

Digital Displacement technology uses a hydraulic piston pump/motor with actively controls high- and low pressure poppet valves to control the output of each cylinder. The electro-magnetic latch controls the opening and closing of these valves per cylinder stroke. Through this, the Digital Displacement Pump (DDP) is able to reduce pressure losses by enabling the cylinders to meet the load on demand.
The configuration presented here uses two motors as this allows for high efficiencies throughout the full displacement range, even at low wind speeds. Only one motor is active at wind speeds below 8.6 m/s, above that the second motor kicks in. Each motor is connected to a separate 11 kV, 2.5 MW generator.

A DDHT is able to keep the motor at a constant speed. This allows the use of synchronous generators to produce synchronised power and thus eliminate the need for a frequency converter; a component that fails frequently.

22.5 Wind farm

The wind farm layout was designed such that aerodynamic wake losses were minimised, while also minimising the amount of electrical cabling and minimising the amount of anchor points needed. A simplistic layout of the wind farm in the form of a parallelogram was chosen. In this layout the losses due to aerodynamic wake effects were predicted to be 10.5% with eight rotor diameters turbine spacing in both wind- and crosswind-direction. This is equivalent to a turbine spacing of 1,577 metres. This spacing is required as in the near wake region of a turbine, the airflow still has a low energy which results in a lower power output of the next turbine. Furthermore, due to the turbulent properties of the wake, the wake will potentially reduce the rotor its lifetime.

To transport the energy to the onshore grid a electrical transport system is needed. The wind turbines are connected in arrays of five turbines to two electrical offshore substations with a power capacity of each 500 MW. Therefore, the 10x10 configuration was chosen in the wind farm layout. The inter-turbine cabling was done using 66 kV cables. These cables have a higher power rating than conventional cables, which means more turbines can be connected on one set of cables, reducing the cable length. Furthermore, electrical transport over a higher voltage means less losses. At the offshore electrical substation the power is transformed to a high voltage of 245 kV. Then the substations are connected to the grid using 245 kV high voltage alternating current (HVAC) cables.

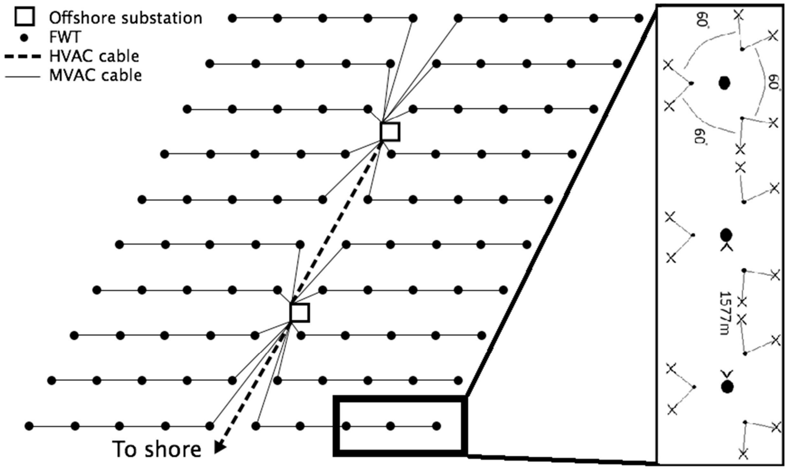

Figure 22.5: Wind farm lay-out

22.6 Cost

To make offshore wind turbines more competitive in the current energy market, huge cost reductions are required. That is why costs are a very important parameter during the design choices. To present the cost to potential investors, the entire lifetime expenditures are calculated with the help of a Life Cycle Cost Analysis. The latter includes the cost of every stage in the lifetime of the floating wind turbine farm.

The first stage is the initial investments for the turbine, floater, mooring lines, electrical grid, and electrical substation. Total initial production costs are calculated to be €2,746 million for a 100 turbine wind farm. Second phase consists of the costs during the operational life. Expenditures needed are operational and maintenance cost for planned- and unplanned repairs. An annual based cost of more than €130 million is estimated. The third phase is at the end of the lifetime. For decommissioning, after the lifetime of 25 years, €160 million is set aside.

With all the phase costs known, the levelised cost of energy could be calculated. This is an economic assessment of the total average cost to build, operate, repair and decommission the wind turbine project divided by the total energy generation. The total project is estimated to cost €3,939 million over a lifetime of 25 years while the sum of energy produced in the operational life is 39.65 million MWh. Dividing the total cost by the total generated energy gives a levelised cost of energy of €99.36 per MWh.

High cost reductions were possible due to both external factors and engineering factors. External factors being the low steel price and the high average wind speed at the location of the wind farm. Engineering factors such as high efficiency blades, low mass of the floater and tower and the use of smaller and thus cheaper blades drove the cost even further down to €30 per MWh below the requirement.

22.7 Conclusion and recommendations

The purpose of this report was to prove the technical feasibility of a Floating Wind Turbine while keeping the costs low to compete with non-renewable energy sources. The final design is a 10 MW twin rotor Floating Wind Turbine which is a good possible solution for the global problem of climate change and energy supply. Furthermore, sustainability is a major factor and has been taken into account in each and every design phase.

This resulted in a renewable energy system that can be recycled almost completely and thus which minimises the impact on the environment. Additionally a hydraulic transmission was implemented which drives the costs down with respect to maintenance and levelised cost of energy. This new system also increases the reliability by not needing gearboxes which is one of the main causes for failure during operation.

As the location of the wind farm is essential for power production, the wind farm was placed North-West of Scotland where the average wind speed is 11.7 m/s, hereby maximising energy output. Due to the optimised blade (with a radius of 65.7 m and mass of 24 tonnes) and a good location, the total annual yield is 54,982 MWh which powers more than 750,000 households in Europe. Moreover, the unique floater which keeps the wind turbine afloat and stable has inter pontoon spacing of 60.6 m which is about the same size as the pitch width at Wembly. The total capital expenditure is about €2.7 billion, while the annual operations and maintenance cost are €130 million. Resulting in a levelised cost of energy of €99.36 per MWh. Which is considerably lower than other offshore competitors and approaches the LCOE of non-renewable energy, making it a good replacement.

Last but not least, the sustainable development strategy led to a cumulative carbon dioxide-equivalent emission of 10.4 gCO_2/kWh. About 85 times less than for their traditional fossil-fuel based counterparts. This fact in combination with the low LCOE shows there are other solutions in the market which limit their footprint while being inexpensive. Seeing as worldwide more and more countries are searching for such a cheap and renewable energy source.

As this project was fairly short, not all aspects of the design were investigated in much detail. With that in mind the following recommendations have been made for future projects relating to this subject. First of all, for stability purposes, heave plates can be included as on the Dutch Tri-Floater, which improves heave damping and overall stability. Furthermore when investigating the structure, fatigue was not taken into account whilst this is evident for offshore structures. This should be looked at in more detail to see how this influences a design and ultimately the weight. Another important aspect is the power conversion, seeing as the hydraulic transmission is a fairly new technology, the efficiencies are not guaranteed. This should thus be investigated. Lastly, assumptions were made for the costs of operation and maintenance. As Aeolus is a new design some of the assumptions may not be as accurate as estimated, this should also be investigated in more detail.

23. . MAGNUS AEOLUS: REDISCOVERING THE MAGNUS EFFECT IN AIRCRAFT

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Project tutor: dr. S. Garcia Espallargas

Coaches: dr. B. Santos, ir. T. Pestana

23.1 Introduction

Innovation has always been a driving characteristic of the aerospace industry. Over a century of development has left us with a great number of different aerial solutions. During the evolution of aircraft from the first versions to modern aircraft, numerous concepts have been discarded, deemed unfeasible or simply forgotten. The concept of using the Magnus effect, an effect that relies on the generation of an aerodynamic force perpendicular to the free stream velocity and the axis of rotation of rotating cylinders or spheres, is one of those forgotten concepts. Even though it was considered almost 80 years ago, no actual flight was ever recorded. Nowadays the Magnus effect finds its application mainly on water, for example on the E-ship 1.

The Magnus Aeolus project was a feasibility study of using the Magnus effect as the main lift providing force for an aircraft. In the following sections the mission, the concepts considered and the detailed design of the aircraft are discussed.

23.2 Mission statement and mission requirements

The mission statement of the Magnus Aeolus project was as follows:

"Transport one passenger from The Hague-Rotterdam airport to London-Heathrow in an aerial vehicle, demonstrating the viability of the Magnus effect as main lift providing force."

The customer, i.e. the tutor of the project, defined a set of requirements. These requirements served as both constraints and guidelines for the design of the Magnus Aeolus.

- The range shall be at least 350 km.
- The mission shall be accomplished in less than 10 hours.
- The Magnus effect shall be the main lift provider.
- There shall exist an end-of-life solution for the product.
- The size of the product shall be comparable to a 2-seater conventional aircraft.
- The product shall be able to carry a minimum payload of 100 kg.

23.3 Conceptual designs and trade-off

Following the mission statement and the requirements a conceptual brainstorm was conducted, where several design concepts were analysed. The process resulted in three feasible concepts, shown in figure 23.1.

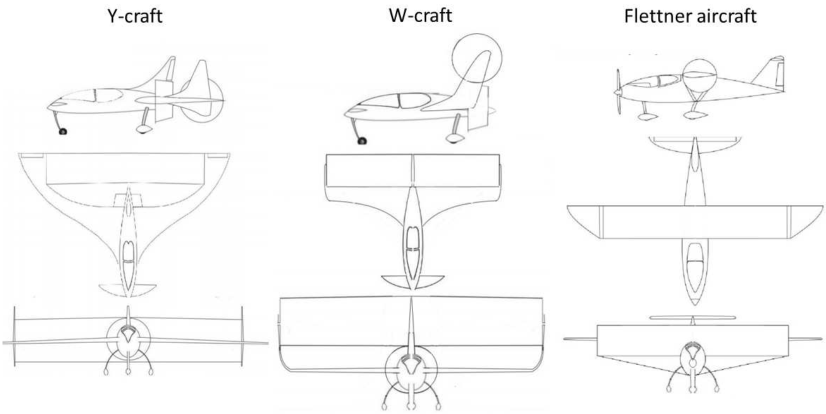

Figure 23.1: Three-view of the concepts considered during conceptual design

The best of the three concepts to accomplish the mission was selected based on a trade-off. The first step towards a trade-off was selecting the criteria that will be assessed and their respective weights. The five considered trade-off criteria were feasibility, weight, drag, originality and stability.

The W-craft came out as the most desired concept. With respect to the criteria, the feasibility was higher because the cylinder does not interact with other parts of the aircraft, which aids in the analysis of the cylinder behaviour. Stability was increased because the moment caused by the drag of the cylinder can be used to decrease the size of the canard, also reducing weight. Originality was similar to the Y-craft, but better than for the Flettner aircraft.

Finally drag was a disadvantage for all of the concepts, but as explained it was useful for this concept. All in all, the W-craft proved to be the best concept for the preliminary design of the Magnus Aeolus.

23.4 Detailed design

Rotor

A CFD simulation of the flow pattern around the rotating cylinder was made using ANSYS Fluent and is shown in figure 23.2. Using this figure the physics phenomena around the cylinder can be effectively explained: the clockwise rotating cylinder is pushing down the low velocity. Additionally, there is a region of high velocity on top of the cylinder. This corresponds to a region of low pressure, sucking the cylinder up into the air.

Compared to a normal aerofoil, the amount of lift that can be generated by the Magnus effect is a lot higher per unit surface area. This allowed for aircraft span to become smaller. However, the main problem corresponding to these high lift coefficients, as can be seen from the size of the wake, was the drag value. This corresponds to a non-efficient way of lift provision, meaning a low lift-to-drag ratio of around 2.5. This also caused the fuel consumption of the aircraft to be very high. However, by looking into literature and using the contact with ENERCON, it was decided to add two endplates to the sides of the cylinder. This way, the maximum lift to drag ratio increased to 5.5, heavily decreasing fuel consumption and resulting in a more environmentally friendly solution.

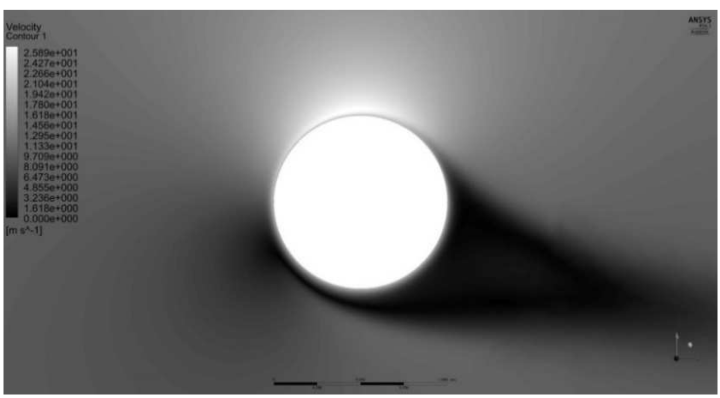

Figure 23.2: CFD simulation of the flow pattern (indicating velocity) around the rotating cylinder

Rotor support

The cylinder provides a distributed lift throughout its span while rotating at high velocity. The combination of a large diameter, high revolutions per minutes (rpm) and bending load, made the support of the rotor an engineering challenge. In addition, by lifting the cylinder up for stability reasons, extra aerodynamic moments are introduced into the structure. The rotor support should not only withstand all the forces. It should also include stability and controllability functions to make efficient use of space and ensure a stable aircraft by having a centre of gravity close to its lift component.

Having the cylinder rotating and loaded will cause misalignment with the bearings. Every bearing allows for a certain amount of misalignment. However, loading the cylinder in bending, means that for regular bearings the misalignment is too large and the bearing will not spin smoothly. In order to make sure the cylinder spins efficiently, the cylinder is supported by self-aligning bearings, which allow the cylinder to spin at its maximum rpm.

The cylinder is held up by two components: the vertical rotor support and horizontal rotor support. The vertical support is an aluminium truss structure and the horizontal rotor support is an aluminium wing box. Both structures have an incorporated symmetric aerofoil. This allows the vertical rotor support to be used as a vertical tail, including a rudder. The horizontal support includes ailerons and fuel tanks for both the propulsion engine and the rotor engine.

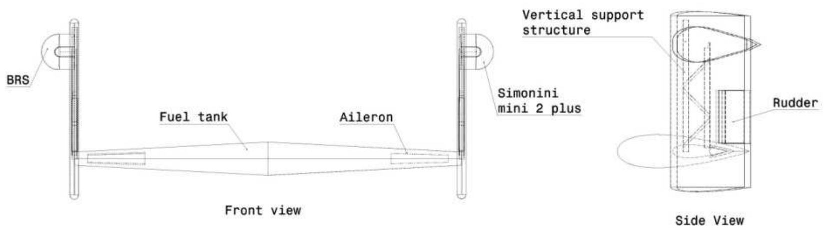

Figure 23.3: Technical drawing rotor support

Propulsion and fuel system

Special consideration was given to the engine selection to overcome the large drag induced by the cylinder. An engine was selected off-the-shelf to increase the reliability. The Mistral G200 engine was selected based on its high power to weight ratio, low fuel consumption when operated at constant high rotational speed, compact size and high reliability. An additional engine is required to rotate the cylinder.

The two-stroke Simonini mini 2 plus engine was chosen because of its high power to weight ratio, high power to volume ratio, low fuel consumption and the low power requirement for the cylinder. A ducted fan is used to shield the fans in order to increase the efficiency of the propeller and decrease the emitted noise.

The high lift coefficients of the cylinder are accompanied by large amounts of downwash. To have the cylinder not interfering with other elements of the aircraft, it was decided to place the cylinder in the back. This location of the cylinders alone however, was not sufficient to completely solve the downwash problem. The flow around the cylinder still interfered with the ducts, pushing down on the top surfaces. Using a CFD simulation, the position of the cylinder with respect to the duct was optimized to align the downwash with the top surface of the duct. Noticing that this was not possible using a single large fan, the duct was split into two. This resulted in having smaller ducts, less interference and a more optimized design.

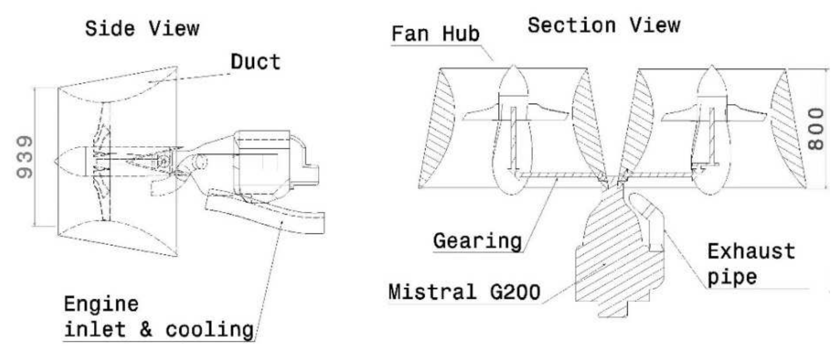

Figure 23.4: Technical drawing ducted fans and engine

Stability and control

The lift production of a Magnus cylinder is not affected by the angle of attack of the aircraft. Therefore most common flight manoeuvres can be executed at 0° angle of attack to minimize the drag. To provide longitudinal stability the chosen concept requires a canard. Canard aircraft are dynamically unstable and therefore require an automated control system for the canard.

The rotating cylinder has a gyroscopic effect on the flight dynamics. Research was conducted on the result and severity of this effect. It was found that the gyroscopic effect acts as a stabilizing force on the short term and induce oscillations on the long term. This behaviour is illustrated in figure 23.5. This represents the aircraft roll response to an aileron input. It can be seen that for an increasing angular momentum of the rotating cylinder the aircraft resists the disturbance and the behaviour becomes oscillatory. The angular momentum of the Magnus Aeolus is 456 kgm²/s. As one can see in the figure the gyroscopic effect for this low angular momentum is small, which makes that conventional control surfaces, like ailerons and rudders can be used to control the aircraft. These control surfaces will be operated by an automated control system to ease flight control for the pilot.

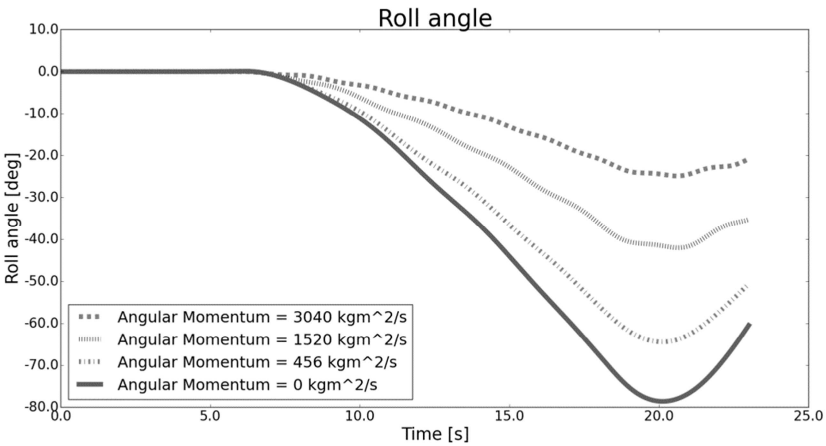

Figure 23.5: Gyroscopic effect on the flight dynamics of the aircraft

Design summary

The most important design characteristics of Magnus Aeolus are presented in table 23.1, and the main characteristics for the rotor in table 23.2. This is followed up by a 3-view technical drawing of the design in figure 23.6.

Table 23.1: General characteristics of Magnus Aeolus

General Characteristics		
Passengers	2	
Length	6.2	m
Height	3.1	m
Width	5.2	m
OEW	646	kg
MTOW	1030	kg
Fuel capacity	246	L
Cargo capacity	20	kg
Average cruise speed	135	km/h
Range	970	km/h
Endurance	7.2	h
Cruise altitude	2134	m

Table 23.2: General characteristics of the rotor

Rotor Characteristics		
Lift-to-drag	5.60	
Span	4.20	m
Radius	0.27	m
Endplate radius	0.68	m
Engine type	Simonini Mini 2 plus	-
Engine power	26	bhp
Cruise RPM	2700	rpm

Design performance

The mission for the Magnus Aeolus was the following:

"Transport one passenger from The Hague-Rotterdam airport to London-Heathrow in an aerial vehicle, demonstrating the viability of the Magnus effect as main lift providing force."

The design proposed, surpasses this mission significantly. The Magnus Aeolus is able to take-off at Rotterdam carrying two passengers to London Heathrow, deliver 20 kg of payload and fly back to Rotterdam without the need to refuel in London. The range of the aircraft is 970 km, almost three times the required range.

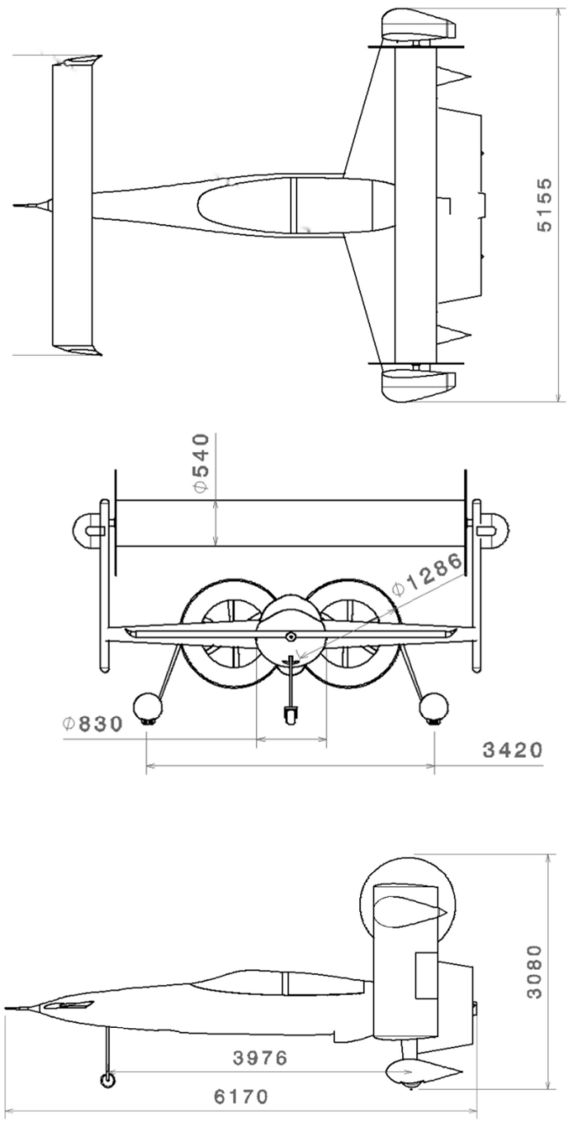

Figure 23.6: 3-view of Magnus Aeolus

Magnus Aeolus is able to adhere to the airport regulations, descending and landing at an angle of 3 degrees at London Heathrow to mitigate noise, resulting in a total landing distance of 405 meters. The take-off distance of the Magnus Aeolus is 504 meters at MTOW.

In case of failure of the main engine, the aircraft can do an unpowered descent and landing at MTOW, improving the safety of the aircraft. In case only the rotor engine fails, the aircraft can descent and land using full power and the horizontal support as lift providers. For the worst-case scenario, where both engines fail, the Magnus Aeolus features an all saver parachute system to protect both the pilots and the aircraft.

Remote controlled Magnus Aeolus model

To gain more insight into the Magnus effect and validate the performance, a remote controlled version of the preliminary design was made. The model had a cylinder span of approximately 1 meter and a mass of 1.2 kilogram. Figure 23.7 shows the model, note that the endplates described in the design are not present in the model.

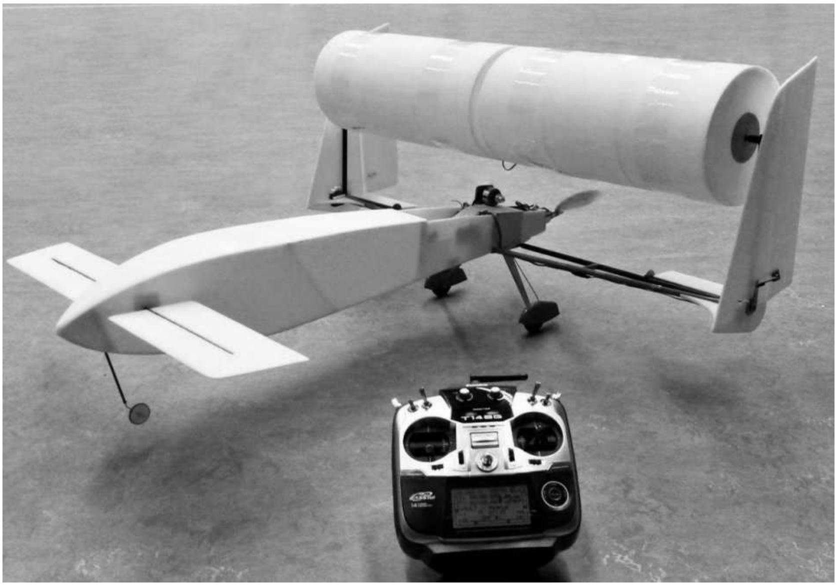

Figure 23.7: Remote controlled Magnus Aeolus Model

A number of test flights were conducted with this model, but it hardly lifted off the ground. The endplates present in the design of the Magnus Aeolus were then also added to the remote controlled model. This allowed the model to take off and thereby demonstrate the functionality of the Magnus effect and the addition of endplates. The model was not stable, which caused it to flip backwards and land on its nose. It did however, confirm many of the design choices made after the preliminary design, like the addition of endplates, the importance of the control system and the chosen bearings.

23.5 Conclusions and recommendations

The Magnus Aeolus project served as a feasibility study of using the Magnus effect as the main lift providing force for an aircraft. The primary conclusion of this study is that this concept is indeed feasible. This raised the question, what place will the Magnus effect have in the future of aviation?

From a lift providing perspective, the Magnus effect has proven to be very effective. The maximum lift coefficient is 5 to 6 times higher than that of a conventional wing, and does not rely on the conventional angle of attack. Unfortunately, the high lift coefficient comes at the price of a high drag coefficient, reducing the lift over drag ratio to 2 or 3 times lower than for a conventional fixed-wing aircraft. Further investigation on the Magnus effect should focus to increase this lift over drag efficiency. Possible modifications which can be developed are a cylinder embedded in a wing or the addition of mid plates on the cylinder.

To illustrate the feasibility of the Magnus effect, a design was made by implementing the Magnus effect to its full potential. This design exceeded the imposed requirements by doubling the payload and traveling 970 km with a speed around three times higher than required. Once again, the aviation industry found its way to innovation.

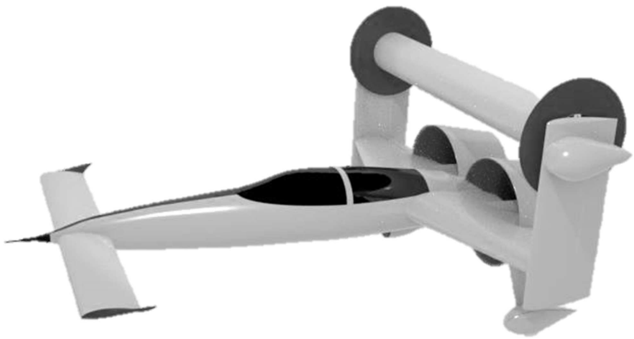

Figure 23.8: Full render of the Magnus Aeolus