Airplane Design and Analysis

AIRPLANE DESIGN AND ANALYSIS

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PREFACE

Textbooks on the subject of Airplane Design are quite well established and known around the world. Classic text books by Egbert Torenbeek, Jan Roskam and Leland Nicolai have been on the desks of airplane designers for many decades. In recent years, modern text books such the one by Daniel Raymer and Leland Nicolai have been used by many college students in their senior year of studies. And there are many more authors that have each written comprehensive texts on the subject of Airplane/Aircraft Design. So why is there a need for yet another text book?

This textbook is written for students who have not yet advanced far into their aeronautical engineering studies. Typically, freshman or sophomore students who have a good grasp of high-school mathematics and physics, but have limited knowledge of the fundamental disciplines that underpin an airplane design: aerodynamics, structures, propulsion, stability & control. In this textbook it is assumed that the reader is familiar with the basic concept of flight of a powered fixed-wing aircraft: the concepts of lift, drag, weight, and thrust. Furthermore, it is assumed that the reader has a basic understanding of the flight mechanics of such a vehicle. All other concepts required to successfully design an airplane are explained within the text and are used directly within the design process.

This book has been written to allow students in the very beginning of their studies to enjoy the process of designing an airplane and learn about the integration challenges that arise when doing so. While the book hardly introduces any novel design methods, it does familiarize the student with a process that can be built upon in more advanced airplane design courses. It allows the reader to understand how the individual aeronautical disciplines come together when a vehicle is designed.

As designing is a process, this process is best internalized by practicing it. Therefore, this textbook has been written including multiple assignments to allow the student to design their own airplane in a step-by-step manner. A particular focus is being placed on the iterative nature of the design process: when or where are assumptions to be replaced by analysis? What "fidelity" of analysis is required at what stage of the design process? And, when do you stop the iterative process and consider a design to have converged? These are some of the questions that are answered in the current text book.

As the amount of different flying vehicles is vast and ever expanding, in this text we only treat the design of fixed-wing aircraft (i.e. airplanes) to limit the scope of the textbook. The aim of this book is to give the reader an introduction into the airplane-design process, rather than covering the design of as many flying objects as possible. On the other hand, we do distinguish between two different energy sources on board the aircraft: electric energy stored in batteries or chemical energy stored in liquid fuel. While electric aircraft are, at the time of writing, still relatively new, it allows for flights without emissions and could therefore contribute to climate-neutral aviation in the future.

A final unique aspect of this text book is that we also explicitly show how a change in design objective affects the outcome of the design process. For example, an airplane designed for minimal operating cost is different from an airplane designed for minimal climate impact, even when either airplane has been designed for the same set of requirements. While it is acknowledged that the design of a fuel-burning engine has a substantial impact on the emissions of an airplane, this has been left out of the scope of this book.

With this book we provide anyone who likes to design an airplane a set of basic methods to do so. Once familiarized with the design process presented herein, they can explore more intricate analysis methods, include more design variables, or alter the order of the design steps during a more advanced Airplane Design course. This book therefore acts as a stepping stone to explore more aircraft design. As the title suggests, this book is intended to really give the student an *Introduction to Airplane Design*.

Roelof Vos
Lawrence, KS, August 2022

1

INTRODUCTION

1.1. ENGINEERING DESIGN

The design of any artifact is typically a creative process. Whether you are designing a coffee mug or an airplane, designing inherently comes down to making decisions. For the coffee mug, these could be the decisions about the size of the mug, the material it is made of, and its color. While the mug needs to be manufactured, very few engineering calculations are required during the design process. For the design of an airplane, decisions also need to be made: what type of propulsion system is used, how is the cabin arranged, or how is the wing attached to the fuselage? The difference with designing a coffee mug is that during the design of an airplane, many engineering calculations need to be made on which many of the design decisions are based. Airplane design, as used in the context of this book, is therefore a version of *engineering design*.

ABET, the Accreditation Board for Engineering and Technology, defines engineering design as follows:[1] *Engineering design is a process of devising a system, component, or process to meet desired needs and specifications within constraints. It is an iterative, creative, decision-making process in which the basic sciences, mathematics, and engineering sciences are applied to convert resources into solutions.* In our case, we consider the design of a product: an airplane. What is evident from this definition of engineering design is that it is not a fundamental science but a process in which engineering science is being *applied*. Therefore, in this textbook, mathematical formulas are often presented without derivation. Where possible, we offer a reference to where the derivation of the formula can be found.

A structured engineering design process is split into several phases in time. Each of these phases consists of similar steps but with an increasing level of detail in the design and increased fidelity in the applied analysis methods. Now, let us consider the typical steps that are to be found in engineering design in each phase of the design process:

1. Define the problem
2. Establish requirements and objective

[1] From: www.abet.org, retrieved 18 May, 2022.

3. Set up options and define design variables
4. Analyze options
5. Verify compliance and, if necessary, **I**terate
6. Compare options
7. Make a choice
8. Evaluate outcome and, if necessary, **I**terate

In the following paragraphs, we briefly elaborate on each of these steps.

Each design starts with a good definition of the *design problem* that needs to be solved. A design problem is a problem that you face when creating a product. A clear problem statement enables you to formulate what you need to find a solution to. It requires you to analyze the problem and possibly decompose the design problem into separate design problems. For example, the design of an aerial reconnaissance system could be decomposed into the design of an aircraft, the design of the payload, and the design of a dedicated ground station. While these three system components have a clear connection, each of them requires its own design solution. An example of a design problem statement is as follows: “More and more people will be flying between city pairs that are relatively close to each other. As each flight contributes to global warming, the greenhouse-gas emissions of regional air travel need to be reduced.” As you can see, this problem statement has an apparent problem (“contribution to global warming”) but also hints towards the solution (i.e., airplanes with reduced greenhouse-gas emissions). This problem statement would be a good starting point for a design process.

Requirements on the design can come from various sources: customers who will be using the product, regulatory bodies that dictate safety standards, or subsystem suppliers whose systems need to be installed in the product. Typically, not all requirements are used in all of the phases of the design process. It is up to you to state all relevant requirements for that particular design phase. A design objective must also be stated to enable the comparison of various design options. This usually means one would like to minimize or maximize a specific performance aspect of the product. The word “performance” should be interpreted here in the broadest sense, as it can relate to any design aspect. A thorough analysis of the requirements and a statement of the design objective is important before progressing to the next step in the design process. In Chapter 3, we will tell you more about how to use requirements and objectives in the design process.

Setting up the design options is the creative part of the design process. In this step, you might look at design solutions that have been proposed in the past. Why were some designs successful and why did other designs fail? Are there some lessons learned that can be applied in the current design? In this step, it is good to think in terms of functions: what should your product be able to do? Should you use a variation of an existing design solution, or are conceiving a completely original design solution? It is good to explore multiple ideas, but you should also keep in mind that the solutions you propose in this step must be engineered in the following steps. To describe each design option, we use the concept of *design variables*. Design variables are entities that can change the shape or properties of the design.

Your design options can be analyzed using engineering methods of various fidelity and accuracy. In this step, it is key to select analysis methods that are suitable to the phase of the design that you are in. The result of the analysis methods should allow you to evaluate how the performance of the design compares to the requirements as well as the design objective. In this textbook, we are at the beginning of the design process of an airplane. Therefore, we select relatively simple analysis methods: empirical or analytical methods. However, in later phases of the design process, the methods can be more complex and might require dedicated software, hardware, and/or experts to perform the analysis. To select the *right* fidelity of the analysis method for each design stage, ensure that the analysis method can distinguish between various design options such that a comparison between the design options can be performed in the next step.

To make sure the various design options can be compared, it must be verified that each of the design options meets the requirements. If a design option does not fulfill the requirements, there is the possibility of making changes to the design variables. These changes result in a different performance. For example, if your analysis tells you that the airplane you are designing is unable to take off within the required take-off distance, you could select an engine with more thrust to comply with this requirement. In many instances, the verification of requirement compliance is closely tied to the analysis of the performance aspects of the design. When we use requirements directly to determine the "right" dimensions of (a component of) the design, we call this *sizing*.

To compare the performance of different design options, a list of performance criteria should be made. Note that the performance criteria are complementary to the requirements. If a design solution cannot meet the design requirements, it is disqualified from the trade-off process that is being performed in this step. Usually, the performance criteria are closely aligned with the design objective, i.e., the design's performance aspect(s) that you try to minimize or maximize. For example, the design objective for an airplane could be to minimize the fuel required for a given mission while minimizing its noise footprint around airports. In that case, the performance criteria could be fuel burn and landing and take-off (LTO) noise. To enable the evaluation of these criteria, the *analyze* step should, therefore, include analysis methods that can quantify these metrics for each of the design options.

The next step is to make a choice. Given the previous step's output, this might seem like a trivial task. However, in practice, this choice can be relatively difficult for several reasons. First of all, you might realize that the comparison performed in the previous step is based on analysis methods that are inherently flawed. Analysis methods capture a simplified version of reality in a mathematical form. Assumptions underpin these methods, which causes an inherent uncertainty in the output of the analysis methods. Particularly when the performance metrics of two design solutions are relatively close, this modeling uncertainty should be taken into account when making a decision. Secondly, the design is still in the initial phase of the product cycle, meaning that in the next phases, there could still be changes that would enable a change in the performance metrics and, therefore, the comparison. Finally, there might be qualitative performance metrics that are difficult to quantify but might have an important impact on the value of the design solution. The aesthetics of a product are an example of such a performance metric. In short: making decisions can be difficult but is necessary to advance the design. A structured trade-off process can help make this decision.

In the final step, the design solution is evaluated. Is the design solution the best product possible? Are there still margins for improvement? Is the product better than competing products? These are the types of questions that are answered at this stage. It is a critical reflection on the design solution. If you see room for improvement, the design can be further iterated, i.e., the design variables can be changed to increase the performance of the design in a certain aspect. If you cannot further improve the design by changing the design variables, you have finished this part of the design process.

ASSIGNMENT 1.1

In this assignment, you are going to define a design problem. As this is a creative assignment, there is no right or wrong answer. To help you along in setting up the problem statement, we have prepared the following questions. If you have trouble answering these questions, you could also use an existing airplane instead.

a. What air transportation needs do you foresee in the near future?
b. What transportation need would you like to focus on during the course of this book?
c. If there are existing design solutions for these transportation needs, how do you want your design solution to distinguish itself from what is already existing?
d. Having answered these questions, write down your design problem statement.

1.2. LEARNING OBJECTIVES

The engineering design process, as described above, is the formulation of a plan to help an engineer build a product with a specified performance goal. In this textbook, we apply this process in a structured manner to the design of fixed-wing aircraft, i.e., airplanes. We thereby focus on steps 1 through 5 of the engineering design process shown on page 1. If you follow the assignments in the subsequent chapters of this book, you will follow a design process and produce your own unique airplane design. The objective is for you to get experience with this design process. Because this is an introductory book, the focus is on traditional aircraft configurations: a fuselage with wings, engines, and tail surfaces. Unconventional airplanes, such as the blended-wing-body or the flying wing, are outside the scope of this book. However, the design process applies to both conventional and unconventional airplanes.

After having read this textbook and having performed the end-of-chapter assignments, you should be able to:

- Formulate an airplane design problem
- Perform a structured airplane design process
- Perform sizing for components: propulsion system, wing, fuselage, and empennage
- Integrate components and make a three-view drawing
- Iterate a design at various stages of the design process
- Know when and how to replace assumptions with analysis results
- Present your design results in a report

1.3. OUTLINE OF THE TEXTBOOK

To achieve the learning objectives, this book will guide you through all the necessary steps. The contents of this book can be divided into three parts. In the first part, we present the design process (2) and how to set up the requirements and objectives (3). Then, in Chapters 7 through 12, we perform the necessary design steps to come to a three-view of the designed airplane accompanied by the performance characteristics. Finally, in Chapter 13, we see how this process could be formalized such that it could be implemented in a software environment.

To help you through this book, we use three fictitious design problems throughout the book. With these examples, we show how each step of the design process could be performed. To practice with the procedures that are presented, each chapter has an end-of-chapter assignment that should be performed for a given design problem. You are free to choose the design problem you wish to use for the assignments in this book. However, the scope of this book is limited to the design of fixed-wing passenger aircraft that have a cruise speed lower than the speed of sound. While some of the methods that are presented in the upcoming chapters can be used for other aircraft types as well, there are also parts that only apply to this category of aircraft. It is therefore recommended to select a design problem that fits within the scope of this book.

2

DESIGN PROCESS

In this chapter, the airplane design process that we follow in this textbook is presented. By completing the assignments that are presented in the subsequent chapters, you will follow this design process and produce your own unique airplane design. Before you get started, this chapter shows how the airplane design process is organized in the industry (Section 2.1. Furthermore, we explain the process you are going to follow when designing your own airplane(Section 2.2). We also introduce the concept of *sizing* and how it differs from designing (Section 2.3).

2.1. AIRPLANE DESIGN PROCESS

The airplane development process in the industry is organized in different phases. Figure 2.1 shows schematically which distinct phases can be recognized and which milestones are present. It is important to distinguish between the different design phases of an airplane as very different activities take place, different tools are used, different numbers of people are involved, and very different amounts of costs are associated with each of them. We will briefly discuss these aspects below.

In the top row of Figure 2.1, two continuous processes occur: research and development (R&D) and market investigation. The R&D department of a company looks into new technologies that could be employed in future airplane projects. Advancement of these new technologies is done through a process of designing, testing, and evaluating these technologies, which we typically refer to as technology maturation. Sometimes, the R&D department of a company is therefore referred to as the R&T (Research and Technology) department. Technologies that have been matured sufficiently can be embedded in the next airplane project. For example, the development of lightweight batteries and electric motors could be matured such that they can be used in a future commuter airplane.

The second continuous process is the Market Investigation. This process is performed by a department that monitors the developments in the market as well as society at large to predict the demand for airplanes in the coming decades. Based on the market

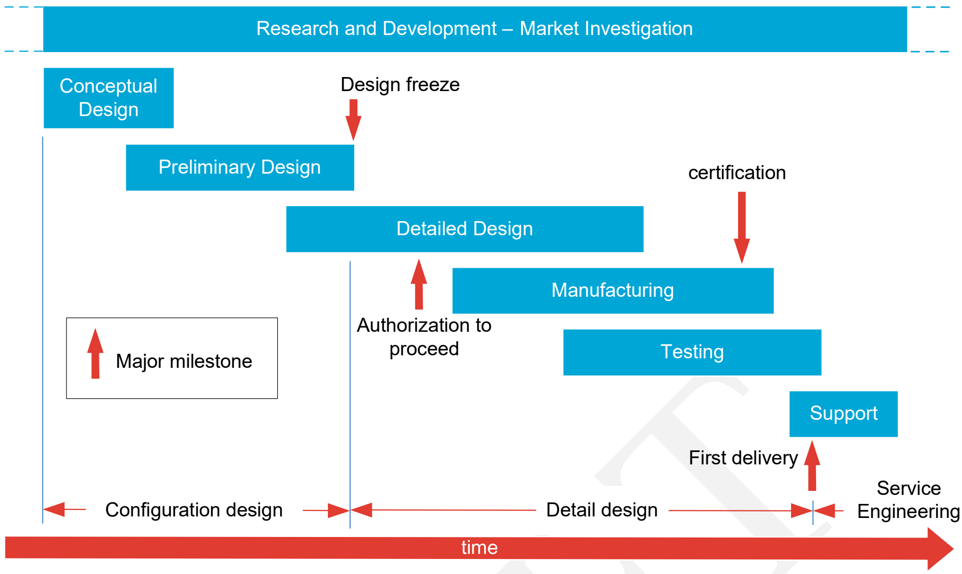

Figure 2.1: Schematic of design and development process in industry (adapted from [20]). Note that the boxes are not drawn to scale.

investigation, requirements for future products can be derived. For example, the economic growth in parts of the world might lead to more demand for air travel between two city pairs. A future airplane should be able to fly between those city pairs and, therefore, be able to cover the range, i.e., the distance between those two cities. The combination of market requirements and selected technologies are often prerequisites to commence the design process of a new airplane.

The design process itself is divided into three non-equal phases: Conceptual Design, Preliminary Design, and Detailed Design. In conceptual design, an airplane is being designed taking into account the so-called top-level aircraft requirements (TLARs) . Using a combination of sizing methods, creative thinking, and critical evaluation, an airplane is conceived that meets these requirements and optimizes a well-defined design objective. The output of this process is typically a report that comprises a three-view drawing of the airplane, including its major components (engines, landing gear, flaps, etc.), a list of airplane specifications (i.e. components masses, center-of-gravity location, stability and control properties, etc.), and a set of assumptions that have been made in order to perform the calculations during the design process. The conceptual design of an airplane is typically performed within a matter of days to weeks by a relatively small team of people. Simple analytical or empirical methods are used in the analysis and sizing methods, which are often embedded in software tools such as AAA.[1]

In the preliminary design phase, the conceptual design is further analyzed and detailed by a variety of disciplinary teams within the company. Within each discipline, parametric studies are performed, and the baseline design concept is refined further.

[1] AAA is a comprehensive airplane design program developed over 30+ years by DARcorporation

The preliminary design phase typically takes months to a few years and is performed by much larger teams compared to the Conceptual Design team. The dimensions, specifications and assumptions from the Conceptual Design phase become part of the requirement set for the preliminary design phase. If you think about the aerodynamic design of the wing, for example, the planform dimensions, thickness distribution, and structural weight constrain the preliminary design of the wing. Furthermore, its performance in terms of lifting capability or the drag it produces also constrains the design of the wing, i.e., the wing needs to possess these capabilities in order to satisfy the performance specifications resulting from the conceptual design process. This also explains why there is an overlap between the conceptual design process and the preliminary design process: specifications from the conceptual design process might be too optimistic or too conservative, leading to an unfeasible or noncompetitive design, respectively.

At the end of the Preliminary Design phase, the design is frozen, meaning that the top-level design variables, along with the performance specifications, are kept constant from that moment onward. At this point, it needs to be decided whether the design is progressing to the next phase of detailed design. This is an important decision as the amount of engineering that is required to perform the subsequent phases is relatively high compared to the previous two phases. For many designs, this is the terminal station of the design process and the design is shelved, only to be revisited in the future if circumstances change. However, if the go-ahead is given to proceed with the design, the outcome of the preliminary design process becomes the specifications for the Detailed Design phase.

In the Detailed Design phase , the manufacturing drawings are being produced for every component of the airplane. Here, a strong interaction with the manufacturing and testing of newly designed components exists because the airplane needs to demonstrate compliance with the specifications stemming from the previous design phases as well as additional specifications coming from the airworthiness regulations (see Chapter 3). This phase typically takes multiple years to complete. Each airplane component is broken down into parts, and an engineering design process is being performed for each part. In this process, suppliers are often included as the expertise for any of the subsystems, which does not necessarily exist at the Original Equipment Manufacturer (OEM). This makes the detailed design process a multi-team, multi-disciplinary process that requires good project management to ensure all the results of all efforts can be properly integrated into a successful product. Systems Engineering tools and methods have proved themselves very useful to enhance this process.[2] These methods are effective in organizing and controlling the product development process.

2.2. CONCEPTUAL DESIGN PROCESS

Within this textbook, the conceptual design process is presented. In this section, we present the process that we propose to be used for this first design phase. We shortly introduce metrics and concepts that are further elaborated in the subsequent chapters. Figure 2.2 on page 11 shows a schematic of the design process. This schematic is called a

[2] Systems Engineering is a ‘language’ (words and diagrams) that is shared by many designers all over the world and helps you communicate about problems, progress, and results of the product development process. See [23] for more information on Systems Engineering.

Design Structure Matrix (DSM[3]). On the diagonal of this matrix the activities are shown: sizing, analyzing, evaluation. The feed-forward connections are shown above and to the right of the diagonal. Meaning that the output of one activity is the input of a subsequent activity. Below and to the left of the diagonal are the feedback connections. The feedback connections show that the process is iterative. If you look carefully, you can distinguish the 3 through 5 of the engineering design process in the design structure matrix. In other words, this process needs to be repeated for every design option before the options can be compared.

The design process starts at the top left of the figure with the definition of the concept and the selection of the design variables. It is assumed that the TLARs have already been defined. The first activities that are proposed here are the estimation of the mass of the airplane as well as the sizing of the wing and propulsion system. These activities are often referred to as Preliminary Sizing and are the subject of Chapter 7. In the second block, the fuselage and wing are being designed. Then, the wing is being positioned on the fuselage based on the predicted location of the component centers of gravity. Also, the propulsion system is further designed and integrated with the vehicle. Finally, the landing gear and empennage are being integrated. Then, a three-view of the full airplane can be constructed.

Once the dimensions of the airplane have been established, a rudimentary aerodynamic analysis can be performed to estimate the drag polar of the airplane. This can be done in various configurations, i.e., with or without the gear extended and with or without the flaps extended. Also, a component-based mass estimation can be performed to estimate the mass of each of the components, i.e., the fuselage, the wing, or the propulsion system. Then, the aircraft's center of gravity excursion during loading and unloading can be computed. The center of gravity is a common term for what is formally known as the center of mass of an object, i.e., the spatial location where the resultant weight vector of the system acts. Furthermore, the maximum lift capability of the airplane can be analyzed for various configurations, as well as its stability and controlability properties.

The output of all these analysis modules can be combined with the output from the previous two activity modules to give us all the information we need to evaluate whether all the constraints are satisfied and what the value is of the design objective. However, we can also use the output of these analysis methods to replace assumptions that have been made earlier in the design process to perform, for example, the mass estimation of the airplane. If this feedback loop is utilized and the process is repeated, the outcome from the analysis methods can be different from the first iteration. Usually, a small number of iterations (2 - 3) are needed to ensure that the output of the analysis methods has converged. Convergence means that the difference in analysis results from two subsequent iterations is below a predefined threshold. For conceptual design, a threshold of 5% is typically sufficient. It is up to you to decide a priori what convergence threshold you would like to use.

At the end of the design process, the constraints and objective values are evaluated. When you follow the process all the way to the evaluation step, you have already used many of the constraints and TLARs to size various aspects of the aircraft (see also Section

[3]A design structure matrix is an example of a Systems Engineering tool to unambiguously describe a design process.

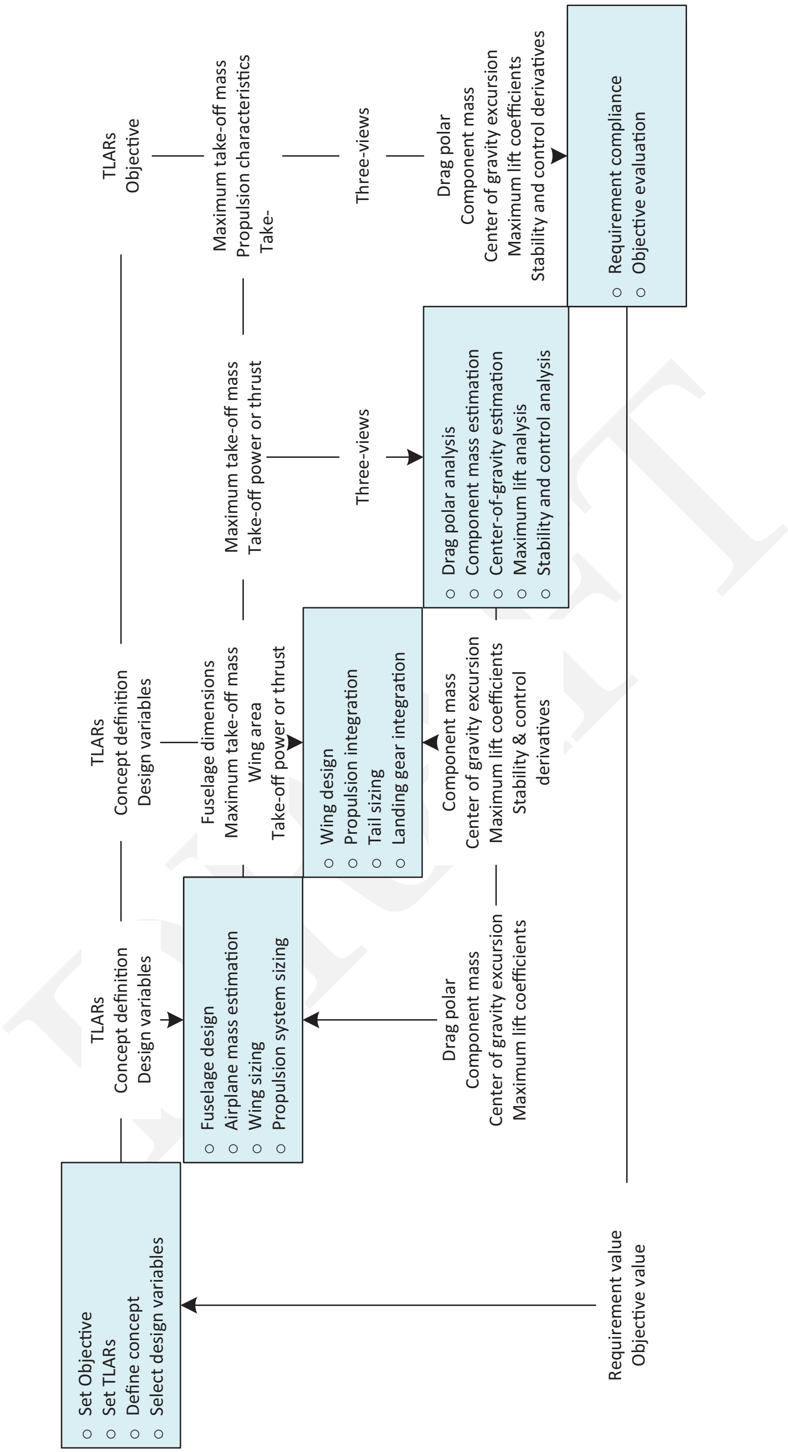

Figure 2.2: Design structure matrix showing the design process proposed in this textbook.

2.3). Demonstrating constraint compliance for these requirements and constraints is, therefore, trivial. However, there might be additional design constraints that need to be evaluated at this stage. Furthermore, the output of the analysis methods should allow you to compute the value of the design objective. If you are satisfied with the outcome of these two evaluation processes, you can stop the process and document your design. However, if either constraints are not satisfied or you think you can further improve the design objective, there is a second feedback loop that brings you back to the beginning of the design process. There, you can change some of the top-level design variables without altering the concept.

In the second rectangular block of Figure 2.2, the words *wing design* and empennage sizing are displayed in italic. We do this to emphasize that we use various *fidelity* methods to design the wing and the empennage surfaces. For the wing design we propose here to first design the wing without the explicit definition of high-lift devices and/or control surfaces. Then, after we have iterated the design once or twice, we refine the wing design by including the wing design variables. For the empennage, we start out by using a very simple method to dimension the size of the tail surfaces. After having done at least one design iteration of the airplane, including its wing movables, we employ a more sophisticated method to size the empennage. We do this to familiarize you with the concept of multi-fidelity analysis. There are two reasons not to start with the higher-fidelity analysis right from the start. First of all, there is not sufficient design information available at the beginning of the design process. Secondly, the higher-fidelity analysis that is required to perform the sizing requires more time and effort to compute. Therefore, you would like to limit the number of design iterations for which you have to perform this sizing process.

2.3. SIZING VERSUS DESIGNING

The design process proposed above includes multiple steps in which the word *sizing* is used. In Chapter 1, we explained that the sizing of a component is a feed-forward process where the dimensions of a component are determined directly based on a requirement. A sizing process is different from a design process because you do not need to make any explicit decisions. The following example tries to make the difference between designing and sizing apparent.

Example 2.1

We are designing a wing for an airplane, which only has one requirement: at sea level and a speed of $V = 100$ m/s, the wing should be able to lift a mass of 50 metric tonnes. The question is: how large should the wing area (S) be in order to fulfill this requirement? It may be assumed that the lift coefficient in this condition is $C_L = 2.0$.

To solve this problem, we employ two approaches, i.e., the design approach and the sizing approach:

1. In the design approach, we would propose a design, analyze it, and compare it to the requirement. Let's propose the wing size is $S = 25\ \text{m}^2$. The lift (L) that this wing

can generate is given by the lift equation (5.7):

$$L = \frac{1}{2}\rho V^2 C_L S = 310\ \text{kN} \sim 31\ \text{t} < 50\ \text{t}$$

Here we have used a sea-level density of $\rho = 1.225\ \text{kg/m}^3$. In other words, the wing that we have proposed is not large enough to carry the mass of the airplane. We can repeat this process multiple times until we find a wing size that is large enough. However, that becomes quite tedious and there is an obvious way in which we can do this more efficiently.

2. In the sizing approach we use the requirement directly. We know that 50 metric tonnes equals 490 kN of weight and with lift equaling weight (W), we rewrite the lift equation as follows:

$$S = \frac{2W}{\rho V^2 C_L} = 40\ \text{m}^2$$

As you can see, we have rearranged the lift equation and combined it with the lift requirement to find a lower bound for the required wing size. Clearly, the sizing approach gives us a precise answer to our design question much quicker than the design approach. Sizing can often be a much faster way to dimension a component than using the design approach. Therefore, in many steps of the airplane design process, one or more design requirements are directly used to size a component.

In the previous example, we sized a wing. However, sizing applies not only to components with physical dimensions that we measure in meters or feet. It can also apply to the thrust of an engine or the pressure in a tire. If a requirement has been used to size a component using a simple algorithm like the re-written lift equation in Example 2.1, the same equation can be used at the end of the design process ("constraint evaluation" in Figure 2.2) to demonstrate that the design complies with that requirement.

3

REQUIREMENTS AND OBJECTIVES

A problem statement is the starting point of any design process. However, if we need to design an airplane, we have to know what requirements the design solution needs to fulfill in order to solve the design problem. Therefore, this chapter is about design requirements and design objectives. The requirements and objectives guide the design process and, together with your design choices, result in your airplane design. If you are designing an airplane for a customer, understanding what their requirements are is an important part of the design project. Apart from these so-called top-level airplane requirements (TLARs), many requirements come from airworthiness regulations. These also need to be taken into account during the design process.

3.1. DESIGN OBJECTIVES

The start of a design process is often triggered by a need. For example, there is a need to reduce the operating costs of an airline or the emissions of aviation in general. In that case, a new airplane design should be better than existing airplanes while performing the same *function*. In other words, your *design objective* would be to design an airplane that is cheaper or less polluting, respectively, than the competition. Alternatively, your design could serve a completely original need. For example, the need to transport a thousand passengers over just 500 km. This would be addressing part of the market for which no dedicated airplanes exist. If there is a desire to address this market segment, then the design objective might be less clearly defined. However, for a new airplane design to be attractive to the market, the cost of the airplane and/or the cost to operate the airplane should be as low as possible. Therefore, minimizing the life-cycle cost of the airplane would be a suitable objective. The following example demonstrates how a design objective can be formulated based on a problem statement.

Example 3.1

PROBLEM STATEMENT:

The CO_2 produced by airplanes contributes to 3% of the total global CO_2 emissions.

Long-haul passenger airplanes are responsible for 45% of the global CO_2 emissions. Therefore, long-haul transport airplanes are an important contributor to global warming.

DESIGN OBJECTIVE:
To design a long-haul passenger airplane that minimizes CO_2 emissions

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Apart from solving a design problem, there might be other triggers to start a design process. For example, the maturation of new technologies that could improve a new airplane significantly. A well-known historical example is the introduction of the jet engine, which increased the speed of passenger airplanes by a factor of two in the early 1950s. Rather than being triggered by a *need*, the design process is then triggered by a *seed*. In the case of military programs, the design process typically starts with a Request for Proposal (RFP). On the base of the actual and foreseen political scenarios and the current capabilities, the problem statement, top-level requirements and design objectives for a new airplane type are included in the RFP and sent to various Original Equipment Manufacturers (OEMs). Each of the OEMs is asked to respond to the RFP with a design proposal.

In the statement of the design objective, it is important that the objective is *specific* and *measurable*. In Example 3.1, the objective is to minimize CO_2 emissions. These emissions can be quantified in the design process as long as analysis methods are included that can compute this. In other words, the objective can be *measured* at the end of the design process. However, it is not very specific. The objective does not state what mission the airplane needs to perform for which the CO_2 emissions should be minimized. An improved design-objective statement would therefore be: "To design a long-haul passenger airplane that minimizes CO_2 emissions per seat-kilometer over a 10,000-km mission." This design objective allows you to quantify the CO_2 emissions in tonnes by analyzing a mission of the airplane. To enable this, a mission-analysis method should therefore be included in the design process as well as a method to quantify how much CO_2 is produced when fuel is combusted within the engine.

In general, the following aspects should be carefully considered before starting a new airplane program:

- Introducing an airplane to the market at the right moment (not too early, not too late), which is "better" than all competitors, if there are any.
- Capability to address the largest market share (customization vs. standardization). Designing a commercial airplane for the special needs of a single customer might only give you access to only a small portion of the market.
- Enter the market with the right technologies. The implemented technologies need to be at the right level of maturity such that they can be manufactured by the OEM, operated by customers, and repaired by a maintenance company.
- Clear understanding of development risks. Everything can be solved when time and money are infinite, but in practice, both resources are limited.

To investigate each of these aspects, continuous communication with potential customers is often key. In the end, they should be the ones purchasing the airplane once it has gone through its design, manufacturing, and testing cycle. Other stakeholders in the process include maintenance companies, passengers, regulatory bodies, and suppliers.

3.2. TOP-LEVEL AIRCRAFT REQUIREMENTS

One of the most important goals when communicating with the stakeholders is to deduce a good set of top-level aircraft requirements (TLARs). These requirements unambiguously state what conditions the airplane design needs to fulfill at the end of its design cycle.

The first and foremost requirement is the payload requirement. In other words: what is it that the airplane needs to transport? And, secondly, how much of that payload needs to be transported in terms of the number of items, payload mass, and payload volume? These requirements are often driving the size of the fuselage, where the payload is typically stored. Furthermore, the mass of the payload is an important driver for the mass of the airplane and, therefore, an important requirement to start the design process. For military airplanes, the payload can consist of armament (bombs, missiles, or guns). Reconnaissance drones typically carry a payload system, for example, a photo camera. Regardless of the type of airplane you are designing, the payload requirement is an important first step.

Example 3.2

For a commercial transport airplane, the payload requirement typically comprises passengers and cargo. Based on the interaction with potential customers, there might be a list of payload requirements. This is an example of such a list:

1. The airplane shall be able to transport 300 passengers and their luggage in a high-density cabin configuration with a 31” seat pitch and 16.5” seat width.
2. The airplane shall be able to transport 250 passengers and their luggage in a typical two-class configuration with 15% business class (55” seat pitch, 20” seat width) and 85% in economy class with a 32” seat pitch and 16.5” seat width.
3. The airplane shall be able to store 20 m^3 of cargo (excluding volume for luggage).
4. The airplane shall have a maximum structural payload of 35 tonnes.

The previous example shows that there are different payload requirements that do not necessarily act at the same time. Often a so-called *design payload* is specified. The design payload is the payload that you decide to be the most flown. In the example above, the design payload would be the second item: 250 passengers plus their luggage. The design payload can be the same as the maximum structural payload, but could also be lower, as is the case in the previous example. In specifying the range requirements of the aircraft, the design payload is coupled to the so-called *design range*, which we discuss below.

You might have noticed from the previous example that the payload requirement list is quite specific. The dimensions of the seat width and seat pitch are important to determine the required volume for the passengers in the fuselage. Also, the required volume for cargo is specified. Many transport airplanes carry cargo and luggage together in the same cargo holds. If that is indeed the case, you have to compute how much volume is required in the cargo hold to store the luggage and the cargo together. Note, that for large transport airplanes, the carry-on luggage is stored in the main passenger cabin. In other words, you need to investigate how much volume is required for luggage in the cabin and how much volume is required for luggage in the cargo hold.

The average mass of a passenger at European airports in 2022 was 76 kg, and their average carry-on luggage weighed 8 kg. The mass of checked luggage is typically bounded by a maximum set by the airline (i.e. 23 kg for long-haul flights). However, the average checked luggage mass in 2022 at European airports was 16 kg. A single average passenger at a European airport therefore has a payload mass of 100 kg. To enable the estimation of the required volumes for storing luggage and cargo, you may assume an average density of luggage of $\rho_{\text{luggage}} = 170\ \text{kg/m}^3$ and an average density of cargo of $\rho_{\text{cargo}} = 160\ \text{kg/m}^3$.

ASSIGNMENT 3.1

Define the payload-related requirements by answering the following questions:

a. Answer the following questions relating to the *maximum-capacity condition* of the airplane:
   i. How many passengers are required to be transported?
   ii. What is the mass of the passengers plus their luggage?
   iii. How are the passengers divided among different classes?
   iv. What seat width and seat pitch do you require?
   v. How many cubic meters of luggage do you need?
   vi. How much volume is required in the cabin for carry-on luggage?
   vii. How much volume is needed in a cargo hold for luggage?
   viii. What is the cargo mass?
   ix. How much cargo volume is required?
   x. What is the maximum structural payload mass?
b. Answer the following questions relating to the *design condition* of the airplane: [a]
   i. How many passengers are required to be transported?
   ii. What is the mass of the passengers plus their luggage?
   iii. How are the passengers divided among different classes?
   iv. What seat width and seat pitch do you require?
   v. How many cubic meters of luggage do you need to transport?
   vi. How much volume is required in the cabin for carry-on luggage?
   vii. How much volume is needed in a cargo hold for luggage?
   viii. How much cargo volume is required?
   ix. What is the cargo mass?
   x. What is the design payload mass?

[a] If the maximum-capacity condition and the design condition are the same, you can skip this step.

To derive part of the flight performance requirements, you should define a *mission profile* for the envisioned design. This can graphically show what it is that the airplane needs to do. The mission profile can be a starting point for deriving the top-level aircraft requirements. The following examples illustrate how one sketches a mission profile and derives the flight performance requirements.

Example 3.3

The mission profile for a transport airplane is shown in Figure 3.1. It shows the two-dimensional profile of the mission, with the horizontal axis representing the distance traveled and the vertical axis representing the altitude of the airplane. As can seen, each phase of the flight is annotated to stipulate what is happening during that phase.

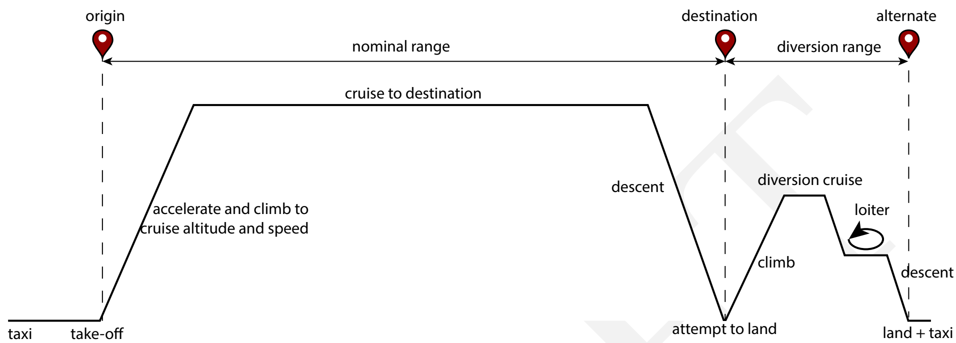

Figure 3.1: Typical mission profile for a transport airplane.

Based on this mission profile, we deduce the following top-level aircraft requirements:

- Taxi: the airplane needs to be able to taxi on a paved taxiway
- Take-off: the airplane needs to be able to take off from a 2.5-km paved runway at sea-level conditions
- Climb: the airplane needs to have a climb rate of 1 m/s at top-of-climb at cruise speed and cruise altitude at 98% of its maximum take-off mass.
- The airplane shall cruise at a Mach number of $M = 0.8$
- The nominal range of the airplane with all passengers on board in a typical seating configuration and without additional cargo is 3,000 km.
- Landing: the airplane shall be able to land on a 2.9-km paved runway in all weather conditions at sea level.
- Diversion: the airplane shall carry sufficient fuel to divert to an alternate airport that is 400 km away, hold (loiter) for 30 minutes, and then perform a landing.

The mission profile of Figure 3.1 shows two distinct 'hops'. The first hop is the *nominal mission*; i.e. the mission that is flown from origin to destination. The distance between these two points is referred to as the *nominal range*. The second hop is the *diversion* part of the mission, which is only used when the airplane is unable to land at the end of its nominal mission. It then needs to fly to an alternate airport, hold (i.e. loiter), and land. To be able to fly to an alternate airport is often required by airworthiness regulations. However, the top-level aircraft requirements typically only specify the nominal range. When designing the airplane, sufficient fuel needs to be taken on board to fly the complete mission profile, including the diversion leg. Specifying the diversion range is therefore required.

If we examine the requirements that we have derived based on the mission profile, you can see that they are *specific* and *measurable*. Of course, these requirements are just an example, but you can see how we use the mission profile to start deriving the mission requirements. It shows that from each phase of the mission, different requirements might result. You need to gather all these requirements to the best of your ability prior to starting the design process. Quantification of the requirements is important such that you can verify compliance with the requirement at the end of the design process. Therefore, you also need to select analysis methods that can quantify the performance aspects that are listed in the requirements. That would allow you to determine whether your design meets all the requirements.

Some of the requirements might seem somewhat ambiguous at first sight and we need to make them more specific in order to be able to check whether they have been met. For example, a 'typical seating configuration' is not specific. Therefore, we need to turn to the payload requirement to understand what is implied by this statement. The other example of a seemingly ambiguous requirement is the statement to be able to land in 'all-weather conditions.' However, this typically implies that you need to consider the worst-case weather scenario and still make sure you comply with this requirement. It is evident that on a dry runway, you can break much harder without slipping than on an icy runway. Therefore, this requirement *implies* that you should be able to land within the specified distance on a dry, wet, and icy runway.

Example 3.4

Figure 3.2 shows the mission profile for a high-altitude reconnaissance airplane. Such an airplane is typically used by the military to gather information. The airplane flies for a long period of time over the same area and relays photographs through a satellite connection to the ground station.

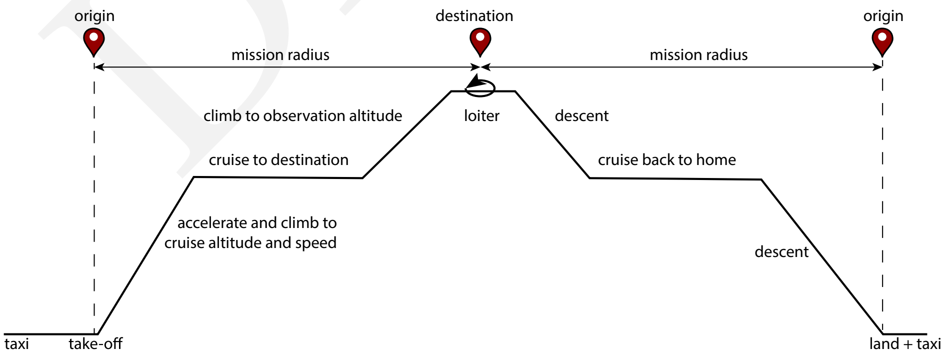

Figure 3.2: Typical mission profile for a reconnaissance airplane.

From this mission profile, we can deduce the following requirements:

- Taxi: the airplane shall be able to taxi on an unpaved, gravel taxiway
- Take-off: the airplane needs to be able to take off from a tarmac runway of 1200 m at sea level.
- Climb (1): the airplane shall have a climb rate of 0.5 m/s at top-of-climb and cruise speed
- Cruise: the airplane shall have a cruise speed of 100 m/s (≈ 200 kts)
- Destination: the mission radius of the airplane shall be at least 500 km
- Observation: the airplane should be able to loiter at an altitude of 14 km
- Climb (2): at the top-of-climb of the observation altitude, the airplane should be able to have a climb rate of 0.5 m/s
- Loiter: the airplane should be able to loiter for 2.5 hours at the observation altitude
- Landing: the airplane shall be able to land on a 1200 m runway in all-weather conditions at sea level.

The mission profile for the reconnaissance airplane is almost symmetric. This is because the airplane typically flies to a particular location, stays there, and flies back. For a given cruise speed and airplane weight, there exists a unique altitude where the airplane cruises most efficiently. That is the altitude one would select to cruise to the destination. When it arrives, the airplane then ascents to the observation altitude to stay there for an extended period of time. Then it flies back home using the same route but then in reverse.

If we look at the requirements that have been deduced in Example 3.4, we recognize some similarities with the ones that have been derived for the transport airplane. However, there are also a few distinct differences. First of all, the mission range is replaced by a mission radius. This is because a military reconnaissance airplane typically does not fly from A to B, but flies from airfield A towards its target and then back to airfield A. When specifying a mission radius, you basically specify a circular domain around the airfield that can be reached by the airplane while performing the mission specified in the mission profile.

Secondly, we see that two climb-rate requirements are set at the end of each climb phase. The reason why the climb rate is typically specified at the top-of-climb is because at that altitude, the air has the lowest density. Air-breathing engines have less available power at high altitudes because of the lower density. Therefore, if you can meet the top-of-climb climb-rate requirement, you can sustain that climb rate also at lower altitudes. In the requirement evaluation, it should be verified that the airplane fulfills both climb rate requirements.

Finally, it might be observed that there is no speed specified to loiter. Therefore, this speed might be freely chosen. If some of the performance aspects are not completely specified, you might turn to the design objective for guidance on how to make this decision. If we use the design objective of Example 3.1, we would try to minimize the CO_2 emissions during the loiter phase. To minimize this, we should try to find a speed at which the loitering flight requires minimum power from the engine while still flying at the observation altitude.

The payload and performance requirements form the top-level aircraft requirements for your design problem. In practice, we often see that the requirements of payload and range are specified as a pair, i.e., a range specified for a given payload (as shown in Example 3.3. Sometimes, multiple payload-range pairs are specified in the requirements, and you need to demonstrate that your design complies with all of them. In this book, we limit ourselves to a single payload-range pair. For this, we use the design payload and the *design range*. The design range is the nominal range that needs to be flown with the design payload.

ASSIGNMENT 3.2

In this assignment, you will derive the mission-related requirements by performing the following tasks:

a. Draw a two-dimensional mission profile that depicts the mission of your airplane.
b. In your drawing, annotate each mission leg appropriately.
c. What is the nominal range required with the design payload?
d. What is the diversion range required with the design payload?
e. What is the loiter time required with the design payload?

In the previous assignment, you derived the mission-related TLARs. This is part of the requirement discovery process. Airports and their infrastructure can also pose important requirements on the airplane. For example, the type of runway surface that an airplane needs to operate from has a great impact on the landing gear design. We subsequently show how to derive these requirements.

During taxi, take-off, and landing, the airplane interacts with the ground surface. Ground surface hardness can range from soft grass to rigid concrete. The type of ground surface greatly impacts the design of the landing gear. Therefore, it is important to specify a requirement to ensure your airplane is able to operate from the airfields that you require it to serve. For example, your can specify that your airplane needs to be operated from a grass runway. When you design the landing gear, you can use this requirement directly to size your wheels and tires. For paved surfaces, the derivation of the requirement is more elaborate. We use the *pavement classification rating* (PCR) to express the load-carrying capacity of a pavement for unrestricted operations. The PCR is defined by the International Civil Aviation Organization (ICAO) and consists of five entries [9]:

$$\mathrm{PCR} = G_1 / G_2 / G_3 / G_4 / G_5 \tag{3.1}$$

where:

- G_1, is a number expressing the pavement bearing strength for a specified standard *subgrade* strength. The subgrade is the soil below the pavement that forms the pavement's foundation. G_1 is numerically defined as twice the Derived Single Wheel Load (DSWL), expressed in hundreds of kilograms. The DSWL is dependent on the layout of the landing gear, i.e., how the wheels are distributed over the pavement.

- G_2 indicates the type of pavement. There are two options:

  R: for rigid pavement
  F: for flexible pavement

- G_3 indicates the subgrade strength category with the following options:

  A: high
  B: medium
  C: low
  D: ultra-low

- G_4 is the maximum allowable tire pressure. This is a numerical entry and can have one of the following values:

  W: No pressure limit
  X: 1.75 MPa
  Y: 1.25 MPa
  Z: 0.5 MPa

- G_5 is the evaluation method that is used. It has one of the following values:

  U: Using aircraft experience
  T: Technical evaluation

The PCR is dependent on the cumulative damage to the pavement stemming from airplane passes. Therefore, it changes over time for a given runway. For design purposes, it is proposed that the PCR be estimated based on the formerly used *pavement classification number* (PCN). The PCN for airport runways is widely published and easy to find on the internet. However, the PCN is no longer used to determine the pavement bearing strength. While PCN uses the same nomenclature as the PCR, the interpretation differs. For example, PCN reports G_1 in thousands of kilograms. Therefore, we propose to derive PCR from the published value of PCN as follows:

$$[G_1 / G_2 / G_3 / G_4 / G_5]_{\mathrm{PCR}} \approx [12 \cdot G_1 / G_2 / G_3 / G_4 / G_5]_{\mathrm{PCN}} \tag{3.2}$$

Figure 3.3: Examples of runway surfaces: grass (left), gravel (center), and asphalt (right). Photos by: AhNrMa, Major Ryan Daughtery, and Paullymac.

Related to PCR is the ACR, or Aircraft Classification Rating. The ACR has the same five entries as the PCR, but is dependent on the instantenous weight of the aircraft. To allow an aircraft to land on a paved runway, the load it exerts on the runway should be below the bearing strength of the runway. In other words, the following requirement must be satisfied:

$$\mathrm{ACR} \leq \mathrm{PCR} \tag{3.3}$$

In summary, to derive the ACR requirement, perform the following steps:

Step 1 Select one or more reference airports from which you wish to operate your airplane.

Step 2 Classify the softest runway from your airport selection in qualitative terms, e.g. soft grass, gravel, hard sand, pavement.

Step 3 For paved runways, find the published PCN values for your reference runways.

Step 4 Select the lowest value for G_1 (runway load) and G_4 (tire pressure) from your reference runways.

Step 5 Using (3.2), compute the G_1 based on the lowest value in your reference runways. Also, select the lowest allowed tire pressure, G_4.

Step 6 State the requirement for the aircraft classification rating, ACR, at maximum take-off mass.

In Chapter 9, we will show how to use the ACR to select the number of wheels and tire dimensions of the landing gear. In the following example, we show how you can derive the ACR requirement for your airplane.

Example 3.5

In this example, we show how to deduce the ACR requirement of your airplane. Assume that we are designing a large passenger aircraft. We require this airplane to be operated from airports near medium-sized cities

Step 1 We select the following three reference airports:

1. Amsterdam Schiphol Airport
2. Kansas City International Airport
3. Manchester Airport

Step 2 All airports have paved runways.

Step 3 For each airport, we choose a single runway from which we require the airplane to be operated, and we list the runway surface in the table below.

Step 4 For each of the three reference runways, list the length and the pavement classification number in the table below

airport name	runway	Surface	length (m)	PCN
Amsterdam Airport	18C/36C	asphalt	3,300	89/F/C/W/T
Kansas City Int. Airport	01R/19L	concrete	2,900	71/R/B/W/T
Manchester Airport	05R/23L	asphalt	3,050	79/F/C/W/T

Step 5 We choose the lowest value for G_1 from the PCN numbers of our reference runways and employ (3.2) to get $G_1|_{\text{PCR}} \approx 850$. G_4 is the same for all runways, meaning there is no limit on the tire pressure, i.e. $G_4 = \text{W}$.

Step 6 We arrive at the following requirement at maximum take-off mass: ACR ≤ PCR = 850/F/C/W/T.

The stepwise process of the previous example can help in determining the ACR requirement for your airplane. Because the process is based on reference airfields, you need to choose them carefully. If you choose only airports with very long runways and high bearing strength, your airplane will not be able to operate from runways with a lower bearing strength. This might limit its operational use. In the following example, you will derive from what runway surface your airplane needs to operate from and, if applicable, the ACR requirement.

ASSIGNMENT 3.3

In this assignment, you will derive what runway type your airplane needs to operate from.

a. Select at least three airports from which your airplane needs to be operated.
b. For each airport, select a reference runway and qualify the surface type.
c. State the softest runway surface that the airplane should be able to land on.

The following questions only pertain to paved runways:

d. Produce a table similar to the one of Example 3.5 and find the PCN for each of your reference runways
e. Based on the PCN values of your reference runways and (3.2), produce the lowest pavement classification rating (PCR).
f. State the requirement for the maximum aircraft classification rating (ACR) and at what mass (i.e. maximum landing mass, maximum take-off mass) it should be evaluated.

In the subsequent assignment, you are going to discover performance-related requirements by referring to the mission profile of Figure 3.1. To make each requirement specific and measurable, we ask what conditions the requirement needs to be fulfilled. The first condition is the altitude, which affects the density, pressure, and temperature. You can require your airplane to fulfill a requirement at a certain altitude above sea level. For example, a take-off must occur from an airfield that is elevated 1.5 kilometers above sea level. The second condition is the difference in local temperature with respect to the International Standard Atmosphere (ISA). For example, a requirement might need to be fulfilled on a warm day when the temperature is 10 degrees Celcius higher than normal. We then write ISA+10°C. For a given pressure, an increase in temperature reduces the air density, which affects the performance of the airplane. The final condition is the instantaneous mass at which the requirement needs to be fulfilled. Examples of the mass are the maximum take-off mass, the maximum landing mass, or a fraction of the maximum take-off mass. More information on the various aircraft mass definitions is provided in Chapter 5. In the following example, we derive the requirement for the landing field length.

Example 3.6

In this example, we derive a fictional landing field length requirement for a large jet airplane. As is shown in Figure 7.5, the landing field length comprises an air distance and a ground roll. We ask ourselves the following questions:

i. What is the minimum required landing field length?
ii. What runway condition is considered, i.e. dry, wet or icy?
iii. What airfield altitude is considered?
iv. What temperature w.r.t. ISA is considered?
v. What airplane mass is considered?

In the following, we present the explicit values for this requirement along with the justification for each number.

i. The shortest runway of our reference airports has a length of 2900 m (see Example **??**). We require that our airplane needs the entire runway to come to a standstill in case of icy conditions. Subsequently, we assume that the ground friction is three times higher in dry conditions compared to icy conditions. In other words, in dry conditions, we have an actual landing distance of no more than 970 m. For safety reasons, we add 2/3 to that distance[1] to arrive at a minimum landing field length, $\mathcal{L}_{\mathrm{LF}} = 1600$ m.
ii. We choose to provide a landing field length requirement for a dry runway. We do this because our design methods for wing sizing are tuned to dry runway requirements (see Chapter 7).
iii. We select the airfield altitude to be at sea level.
iv. We want to be able to operate the airplane also on hot days when the local temperature is 30°C. This is 15 degrees higher than the average temperature at sea level in ISA conditions (288 K or 15°C). The temperature is ISA +15°.
v. Finally, we have to think about the mass of the airplane during landing. Here we envision that the airplane is able to shed at least 15% of its maximum take-off mass by burning fuel. The maximum landing mass, m_{ML}, is therefore set to 85% of the maximum take-off mass.

The previous example shows how you can derive a particular requirement. You can derive these requirements on your own, using straightforward investigative tools. You can also derive these requirements by interacting with other people. The questions that are listed in the previous example can help you to make sure that the requirement formulation becomes specific and measurable. This is important because we need these requirements in the sizing of the wing and powerplant (see Chapter 8).

ASSIGNMENT 3.4

In this assignment, you are going to derive specific flight performance requirements based on the phases of the mission profile of Assignment 3.2. As you are deriving the requirements, there is no single correct answer to each of these questions. Also, for one or more phases of the flight, you might not have a specific performance requirement. These phases you can skip.

a. Take-off

[1] For airplanes that are certified under CS-25 (See Section 3.3), a safety factor of $1\frac{2}{3}$ is required.

   i. What is the minimum required take-off distance?
   ii. What runway surface is considered, i.e. grass, gravel, tarmac, concrete?
   iii. What airfield altitude is considered?
   iv. What temperature w.r.t. ISA is considered?
   v. What airplane mass is considered?

b. Climb
   i. What is the minimum required climb rate?
   ii. What altitude is considered for this climb rate?
   iii. What temperature w.r.t. ISA is considered?
   iv. What airplane mass is considered?

c. Cruise
   i. What is the minimum required cruise speed (or cruise Mach number)?
   ii. What cruise altitude is considered?
   iii. What temperature w.r.t. ISA is considered?
   iv. What airplane mass is considered?

d. Landing
   i. What is the minimum required landing field length?
   ii. What runway condition is considered, i.e. dry, wet, or icy?
   iii. What runway surface is considered, i.e. grass, gravel, tarmac, concrete?
   iv. What airfield altitude is considered?
   v. What temperature w.r.t. ISA is considered?
   vi. What airplane mass is considered?

3.3. AIRWORTHINESS REQUIREMENTS

While part of the top-level requirements can be derived from the mission that the airplane has to perform, additional top-level requirements are dictated by airworthiness authorities. In this section, we discuss where these airworthiness requirements come from, how we distinguish between different categories, and which ones you need for your design.

The aviation industry as a whole is heavily regulated. Not only does the airplane need to comply with airworthiness requirements, also maintenance organizations, airplanes, and airports are heavily regulated. The goal of these regulations is to ensure a lower level limit for the safety of the airplane system. If we consider an airplane system, it can be decomposed in various ways:

- Its subsystems: aircraft, training, support, facilities, personnel
- Its segments, e.g. airframe segment, avionics system, environmental system
- Its life cycle: system planning, research, design, production, evaluation, use, support and disposal

For each of the components above airworthiness regulations exist.

In the United States, the Federal Aviation Administration (FAA) specifies what requirements an airplane needs to fulfill if it is to be certified. It is determined by law what requirements an airplane must comply with. These so-called Federal Aviation Regulations (FARs) are closely aligned to the European Certification Specifications (CS) that are dictated by the European Union Aviation Safety Agency (EASA). While there are many more airworthiness bodies in the world, we will limit our discussion here to the FARs and the CSs. And, while the FARs and CSs comprise thousands of requirements, we only focus on the ones that are relevant to the design of your airplane at this very early stage.

The airworthiness standards in the US are divided into different Parts of Title 14 of the Code of Federal Regulations, CFR. Which standards apply depends on the following aspects: the size (i.e. number of passengers and/ or weight), the function of the airplane (i.e. the kind of maneuvers it makes), and the powerplant technology (either turbine of propeller). In Table 3.1 on page 29 we show the various airworthiness standards. For the smallest airplanes, the US regulations and EU regulations differ slightly where the Very Light Aircraft specifications have a maximum take-off mass limitation of 750 kg and the Light Sport Aircraft specifications have this limit at 600 kg. For either category, a maximum stall speed of 45 kts is specified. The stall speed is the lowest speed an airplane can sustain in level flight. Airplanes that are larger can be certified under CS/FAR-23 when they weigh less than 5670 kg (12,500 lb), have 9 seats or less (excluding the pilots), and fall in the normal, utility or aerobatic category. Commuter aircraft can still be certified under CS/FAR-23 when they weigh less than 8618 kg (19,000 lb) and have 19 or less seats (excluding the pilots). All larger airplanes that do not fall under CS/FAR-23 are certified under CS/FAR-25. Note that CS/FAR 23 has a stall-speed requirement for all single-engine airplanes as well as twin-engine airplanes of less than 2722 kg (6000 lb) that are unable to meet a climb-gradient requirement of 1.5% at an altitude of 5000 ft with one engine inoperative. There is no stall speed requirement for other twin-engine airplanes, commuter airplanes, and large airplanes.

The propulsion system that is used in each category can differ. For small airplanes, propulsion is provided through propellers. Large airplanes either use propellers or jet propulsion. Propeller torque can be created by a reciprocating engine, an electric motor, or a turboprop engine. A turboprop engine is a gas turbine that produces shaft power to spin a propeller. In the VLA/LSA category, airplanes are required to have (a maximum of) one engine or motor. CS/FAR-23 airplanes in the normal, utility, and aerobatic categories are also required to have a single powerplant, which can be either a reciprocating engine or a turboprop. For the commuter category, one or two engines can be present. Examples include so-called *twin props* as well as very-light jets (VLJs). Large airplanes, certified under CS/FAR-25 are turbine-powered, either employing turboprop engines or turbofan engines. In a turbofan engine,, the gas turbine is connected to a ducted fan, and thrust is produced as a reaction to the jets coming from the exhaust of the gas turbine and the ducted fan. The word 'turbine' in Table 3.1 refers to thrust being produced from a turbofan engine. The following examples show the category under which an airplane qualifies.

Table 3.1: Applicability of airworthiness regulations per airplane category.

	CS-VLA	FAR LSA	CS/FAR-23		CS/FAR-25
Category	normal	normal	n/u/a[2]	commuter	large
Seats	≤ 2	≤ 2	≤ 9	≤ 19	
max. take-off mass (kg)	≤ 750	≤ 600	≤ 5670	≤ 8618	
stall speed (kts[3])	≤ 45	≤ 45	≤ 61 [4]		
Propulsion	prop	prop	prop	prop/turbine	turbine

Example 3.7

The Pipistrel Velis Electro (Figure 3.4) is a two-seater airplane. The function of this airplane is to train pilots. It has a single electric motor powering a propeller. Its maximum take-off mass is 600 kg. In terms of maneuvers, it falls in the 'normal' category meaning it can do: *(1) Any maneuver incident to normal flying; (2) Stalls (except whip stalls); (3) Lazy eights, chandelles and steep turns or similar maneuvers, in which the angle of bank is not more than 60°.* [5] Finally, it has a stall speed of 45 knots. The Velis Electro can therefore be certified under CS-VLA as well as in the FAR-LSA.

Example 3.8

The de Havilland Canada DHC-6 Twin Otter is a twin turboprop airplane. It can seat a maximum of 19 passengers and has a maximum take-off mass of 5670kg. It is of the 'utility' category, meaning it can be used for transporting people as well as goods. In terms of maneuvers, it should be able to handle the same maneuvers as in the 'normal' category plus *(1) Spins (if approved for the particular type of aeroplane); and (2) Lazy eights, chandelles, and steep turns, or similar maneuvers in which the angle of bank is more than 60° but not more than 90°.*

Example 3.9

The Cirrus SF50 (Figure 3.6) is a single-engine Very Light Jet. It is controlled by a single pilot and can seat up to six passengers. It has a single turbofan engine and a maximum take-off weight of 2,722kg (6000lb). It is certified under CS-23 in the 'normal' category. It features a ballistic parachute system that can rescue the entire airplane in case of an emergency. Such a parachute system is not required by CS-23, demonstrating that the regulations merely provide a lower limit of safety. An airplane manufacturer can choose to improve the safety of the airplane by adding systems like this parachute.

[2] n =normal, u = utility, a = aerobatic
[3] 1.0 [kts] = 0.51 [m/s]
[4] See main text for applicable conditions
[5] See `https://www.easa.europa.eu/sites/default/files/dfu/decision_ED_2003_18_RM.pdf`, Accessed 2 June, 2022.

Figure 3.4: The Pipistrel Velis Electro is certified under CS-VLA. Photo: Airjuice Photography

Figure 3.5: The DHC-6 is certified under CS/FAR-23 in the 'utility' category. Photo: Timo Breidenstein

Example 3.10

The Embraer E190E2 (Figure 3.7) is a transport airplane seating 96 passengers in a typical two-class configuration. It has two turbofan engines and a maximum take-off mass of 56,400 kg. This airplane clearly qualifies as a Large Airplane and is certified under CS/FAR 25.

Figure 3.6: The Cirrus SF-50 is certified under CS/FAR-23 in the 'normal' category. Photo: Markus Eigenheer.

Figure 3.7: The Embraer E190E2 is certified under CS/FAR-25. Photo: Valentin Hintikka.

When you define the mission of the airplane, it is important to also clearly state for what airworthiness category the airplane should be designed. CS/FAR 25 contains more and stricter requirements than CS/FAR 23. The same holds for CS-VLA compared to CS-23. The latter contains more and stricter requirements than the former. If you design a short-range two-seater airplane, you have to comply with fewer requirements if you include the top-level constraints stemming from CS-VLA. In practice, this implies that you have to do fewer tests to demonstrate that the airplane complies with all certification requirements. This reduces the development cost of the airplane compared to certification under CS-23.

ASSIGNMENT 3.5

What airworthiness regulations apply to your design? Motivate your answer based on the following criteria: weight, number of passengers, and powerplant type. If CS-23 applies, in what category should be selected?

Once you have determined which airworthiness regulations apply to the design of your airplane, it is time to think about which requirements need to be included at the earliest stage of the design process. Depending on the regulations that apply, there can be hundreds to thousands of requirements that the airplane needs to demonstrate compliance with. To demonstrate compliance, each set of airworthiness regulations consists of two books. Book 1 is the Airworthiness Code and contains all the requirements. It is divided into several subparts to distinguish between various aspects of the airplane, including structure, equipment, and power plant. Book 2 holds the Acceptable Means of Compliance (AMC). It has the same subparts as Book 1 and explains how one can demonstrate compliance with the requirements specified in Book 1. In the examples below, we present requirements that are important for the design process that is covered in the present textbook. The curious reader is encouraged to explore the various CS or FAR documents that can be found online.

The flight performance requirements are specified in subpart B of CS/FAR-23 and CS/FAR 25. These requirements are going to be important for sizing the wing and power plant in Chapter 7. We focus here on two sets of requirements: requirements that specify the minimum speed the airplane should be able to fly and requirements that specify the minimum climb gradient the airplane should be able to achieve.

First, we look at the stall speed requirements in the airworthiness regulations. For CS/FAR-23-certified airplanes a minimum stall speed is prescribed as is shown in the following example:

Example 3.11

CS/FAR 23.49 Stalling Speed

(c) ..., VS0 at maximum weight must not exceed 113 km/h (61 knots) for –

i. Single-engined aeroplanes; and
ii. Twin-engined aeroplanes of 2722 kg (6 000 lb) or less maximum weight that cannot meet the minimum rate of climb specified in CS 23.67 (a)(1) with the critical engine inoperative. [...]

The previous example shows that a 61-knot (31 m/s) stall-speed (VS0) requirement applies to all CS-23 airplanes, but also that there are some exceptions. Reference is made to other paragraphs in the regulations that detail these exceptions. If you wish to certify your airplane as a very light aircraft (VLA) or light-sport aircraft (LSA), the minimum speed cannot exceed 45 knots (23 m/s). CS/FAR 25 does not state a minimum speed requirement. However, as large airplanes need to be able to fly a specific traffic pattern in order to land safely, they do need to fly slow enough to do so safely. Therefore, typically a final approach speed should be specified (V_{APP}) in the TLARs. According to CS-25 there needs to be a speed margin of 23% between the minimum final approach speed (VREF) and the stall speed (VSR0) as specified in CS/FAR 25.125:

Example 3.12

CS 25.125 Landing

(a) The horizontal distance necessary to land and to come to a complete stop from a point 15 m (50 ft) above the landing surface must be determined [...]

(b) In determining the distance in (a):

(1) The aeroplane must be in the landing configuration.

(2) A stabilised approach, with a calibrated airspeed of not less than VREF, must be maintained down to the 15 m (50 ft) height.

(i) In non-icing conditions, VREF may not be less than:

(A) 1.23 VSR0;

Here, the 'landing configuration' implies that the flaps are fully extended and the landing gear is deployed. If a final approach speed is specified in the TLARs, the stall speed in the landing configuration can easily be derived by dividing the final approach speed by 1.23. The relation between approach speed (V_{app}) and stall speed (V_{S0}) is defined in CS-25:

$$V_{\text{app}} = 1.23 V_{\text{S0}} \tag{3.4}$$

While the conditions for the stall speed for CS/FAR-23 airplanes are listed in the paragraph describing the stall, the conditions can be chosen for the CS-25 airplane. For example, one could choose to specify a minimum final approach speed at a lower mass than the maximum take-off mass,i.e. at *maximum landing mass*, at a higher altitude, or at an increased temperature. The increase in altitude and temperature both result in a lowering of the density, which affects the generation of lift. This is why we need to be specific when listing our requirements. In the following assignment, you will derive the stall speed requirement or approach speed requirement for your airplane.

ASSIGNMENT 3.6

Complete one of the two tasks depending on the certification basis:

1. For CS/FAR 23 airplanes and CS-VLA airplanes, state the stall speed requirement along with the following conditions:
   i. altitude
   ii. airplane mass w.r.t. maximum take-off mass
   iii. temperature
2. For CS/FAR 25 airplanes *choose* the approach speed and specify the associated conditions:
   i. altitude
   ii. airplane mass w.r.t. maximum take-off mass
   iii. temperature

Now, we will look at the climb gradient requirements that are specified in the airworthiness regulations. The climb gradient tells you how steep you can climb (see Figure 3.8). You can imagine that this requirement relates to being able to clear obstacles after a take-off. Example 3.13 is an example of a climb gradient requirement.

Example 3.13

CS/FAR 23.65(a) Climb: all engines operating
Each normal, utility, and acrobatic category reciprocating engine-powered airplane of 6,000 pounds or less maximum weight must have a steady climb gradient at sea level of at least

(a) 8.3% for landplanes
(b) 6.7% for seaplanes

With: (1) Not more than maximum continuous power on each engine; (2) The landing gear retracted; (3) The wing flaps in the takeoff position(s); and (4) A climb speed not less than the greater of 1.1 V_{MC} and 1.2 V_{S1} for multi-engine airplanes and not less than 1.2 V_{S1} for single-engine airplanes.

You can see that this requirement is very *specific*: four conditions must be met when the requirement is verified. These conditions are related to the available power, the landing gear configuration, the wing flap configuration, and the speed relative to the stall speed in cruise configuration (V_{S1}) and, when applicable, the minimum control speed[6]. Secondly, you can also tell that the requirement is *measurable*, i.e. the climb gradient can be quantified by dividing the climb rate by the forward speed. Both of these quantities can be measured in flight.

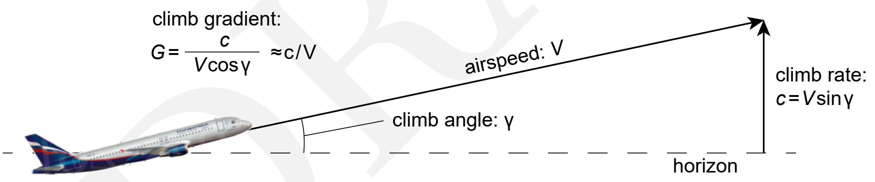

Figure 3.8: Definition of the climb gradient (G), climb speed (c), and climb angle (γ).

The climb gradient requirement of Example 3.13 is just one climb-gradient example stemming from CS/FAR-23. Several climb gradient requirements might be applicable to your design and need to be verified. It is up to you to discover which climb-gradient requirements apply to your design. The following paragraphs in CS/FAR-23 should therefore be consulted:[7]

CS/FAR 23.57 - Take-off path
CS/FAR 23.63 - Climb general

[6] The minimum control speed is the calibrated airspeed below which directional or lateral control of the airplane can no longer be maintained, after the failure of one or more engines.
[7] See EASA online documentation of CS-23

CS/FAR 23.65 - Climb: all engines operating
CS/FAR 23.67 - Climb: one-engine inoperative
CS/FAR 23.77 - Balked landing

These paragraphs cover the climb gradient requirements that must be met in various parts of the mission profile. In paragraph 57, the take-off maneuver and requirements are presented. Then, in paragraph 63 the conditions are specified, which apply in paragraphs 65, 67, and 77. Then, paragraph 65 details the requirement in case of an all-engines-operative (AEO) condition (see 3.13). For multi-engine airplanes, CS/FAR 23.67 specifies the requirements that need to be satisfied in case of a one-engine inoperative (OEI) condition. Finally, paragraph 77 specifies the climb gradient requirements in case of balked landing, i.e. a go-around near or from the ground. The latter needs to be evaluated in the AEO condition.

As CS-23 has four different categories of airplanes, there is a differentiation between the climb gradient requirements for each of these categories. Furthermore, within the aerobatic, normal, and utility category, another distinction is made between airplanes of 2722 kg (6,000 lb) or less and airplanes heavier than 2722 kg. If you are at the beginning of the design process and you do not yet know what the mass of the airplane is going to be, it is recommended to state the set of climb-gradient requirements for both cases. After the initial weight estimation (Chapter 7) it can be determined which of the sets should be used. Similar to CS/FAR 23, CS/FAR-25 also has a set of climb gradient requirements. Here is an example from CS-25.

Example 3.14

CS/FAR 25.119 Landing climb: All-engine-operating.
In the landing configuration, the steady gradient of climb may not be less than 3.2%, with

(a) The engines at the power or thrust that is available eight seconds after initiation of movement of the power or thrust controls from the minimum flight idle to the go-around power or thrust setting; and
(b) A climb speed of not more than 1.3 V_S.

The previous example describes the landing climb, i.e. the climb gradient after the pilot has decided to abort the landing. In that case, the airplane is in *landing configuration*, meaning that the flaps are fully extended and the landing gear is deployed. All of the climb gradient requirements that need to be satisfied can be found in the following CS/FAR paragraphs:[8]

CS/FAR 25.111 - Take-off path
CS/FAR 25.117 - Climb: general
CS/FAR 25.119 - Landing climb: all engines operating
CS/FAR 25.121 - Climb: one-engine inoperative

[8] See: EASA online documentation of CS-25

In short, paragraph 111 describes the take-off path in detail, showing that it consists of various segments of climb and acceleration. Paragraph 117 states that each of the requirements in paragraphs 119 and 121 need to be shown at each weight, altitude, and ambient temperature within the operational limits of the airplane. Paragraph 119 describes the climb gradient requirement in the landing phase in the AEO condition (see Example 3.14). For each of the climb segments of the take-off path, paragraph 121 defines the gradient requirement in case of an OEI condition. Note that in paragraph 121(d) also a climb gradient requirement is provided for the approach segment, where the airplane is landing configuration. This ensures that in case of an OEI condition, after a missed approach the airplane is able to climb. A summary of these requirements is shown in Table 3.2.

Table 3.2: Summary of climb gradient requirements of CS/FAR 25.119 and CS/FAR 25.121. L = landing, TO = take-off, CR = cruise, APP = approach.

						climb gradient		
CS/FAR	condition	Segment	flaps	gear	mass	$N_e = 2$	$N_e = 3$	$N_e = 4$
119	AEO	L	L	↓	m_{MTO}	3.2%	3.2%	3.2%
121 (a)	OEI	1st TO	TO	↓	m_{MTO}	0.0%	0.3%	0.5%
121 (b)	OEI	2nd TO	TO	↑	m_{MTO}	2.4%	2.7%	3.0%
121 (c)	OEI	3rd TO	CR	↑	m_{MTO}	1.2%	1.5%	1.7%
121 (d)	OEI	APP	L	↑	m_{ML}	2.1%	2.4%	2.7%

Note that the requirements are different depending on the number of engines (N_e) that you specify for your design. A two-engined airplane needs to comply with less strict requirements compared to a three-engined or four-engined airplane in case of an OEI condition. For each requirement, particular conditions apply: the flap configuration of the airplane may be different, the landing gear may be up or down, and the mass of the airplane is either the maximum take-off mass (m_{MTO}) or the maximum landing mass m_{ML}.

ASSIGNMENT 3.7

Define the climb gradient requirements stemming from the airworthiness regulations. If the design requirement depends on the number of engines, revisit this assignment after you have completed Assignment 4.3.

a. Which paragraphs and subparagraphs from the airworthiness regulations define the climb gradient requirements?
b. For each applicable paragraph and subparagraph, state the climb gradient requirement along with the following conditions:
   i. Weight
   ii. Thrust available relative to maximum thrust available
   iii. Landing gear configuration (i.e. extended or stowed)
   iv. Wing flap configuration (i.e. cruise, take-off, or landing)

The climb-gradient requirements discussed above are part of the flight performance requirements of the airplane, which also include requirements on the take-off distance, the cruise speed, the climb rate, and the landing field length. However, you will also need other requirements to design your airplane. To design your fuselage, you need to know how many doors are required. Requirements for emergency evacuation can be found in subpart D 'Design and Construction' of CS/FAR-23/25. A top-level requirement for both CS/FAR-23 and CS/FAR-25 airplanes is that passengers should be able to leave the airplane in case of an emergency situation within 90 seconds (see paragraph 803 CS/FAR-23/25). In the conceptual design stage, you cannot easily test this requirement. Luckily there are other requirements that you can use to make sure this 90-second requirement can be met. The example below shows a shortened version of a paragraph in CS/FAR-23 describing the requirements for the placement of emergency exits.

Example 3.15

CS 23.807 Emergency exits

(a) Number and location. Emergency exits must be located to allow escape without crowding in any probable crash attitude. The aeroplane must have at least the following emergency exits:

  (a) For all aeroplanes with a seating capacity of two or more, excluding aeroplanes with canopies, at least one emergency exit on the opposite side of the cabin from the main door specified in CS 23.783. [...]

(b) Type and operation. Emergency exits must be movable windows, panels, canopies, or external doors, openable from both inside and outside the aeroplane, that provide a clear unobstructed opening large enough to admit a 48- by-66 cm (19-by-26 in) ellipse. [...]

(d) Doors and exits. In addition, for commuter category aeroplanes the following requirements apply:

  (1) In addition to the passenger-entry door

    (i) For an aeroplane with a total passenger seating capacity of <u>15 or fewer</u>, an emergency exit, as defined in subparagraph (b) , is required on each side of the cabin; and

    (ii) For an aeroplane with a total passenger seating capacity of <u>16 through 19</u>, three emergency exits, as defined in subparagraph (b), are required with one on the same side as the passenger entry door and two on the side opposite the door. [...]

It may be noted that this requirement specifies the number and required dimensions of the emergency exits for CS-23 airplanes. Note, that sub d gives special attention to commuter category airplanes, which again emphasizes the differences between the various categories. When adhering to these requirements it may be assumed that the 90-second rule of paragraph 803 may be complied with. Similarly, CS-25 also has regulations regarding the number of doors and their placement. In particular, paragraph 807 covers this explicitly. There exist several standardized door types, which are defined in paragraph 807(a) and are further detailed in Chapter 6. The largest standard exit is the Type A exit, measuring 107 cm in width and 183 cm in height. The smallest is a Type IV exit, which is 48 cm wide and 66 cm high, the same size as CS-23 emergency exits. In the following example, the minimum number of emergency exits, along with their type, is specified.

Example 3.16

CS 25.807 Emergency exits

(e) Uniformity. Exits must be distributed as uniformly as practical, taking into account passenger seat distribution.

(f) Location.

(3) If more than one floor-level exit per side is prescribed, and the aeroplane does not have a combination cargo and passenger configuration, at least one floor-level exit must be located on each side near each end of the cabin

(4) For an aeroplane that is required to have more than one passenger emergency exits for each side of the fuselage, no passenger emergency exit shall be more than 18.3 m (60 feet) from any adjacent passenger emergency exit on the same side of the same deck of the fuselage, as measured parallel to the aeroplane's longitudinal axis between the nearest edges.

(g) Type and number required. The maximum number of passenger seats permitted depends on the type and number of exits installed on each side of the fuselage. Except as further restricted in subparagraphs (g)(1) through (g)(9) of this paragraph, the maximum number of passenger seats permitted for each exit of a specific type installed on each side of the fuselage is as follows:

Emergency exits (each side of fuselage)	Maximum number of passenger seats allowed
Type A	110
Type B	75
Type C	55
Type I	45
Type II	40
Type III	35
Type IV	9

(1) For a passenger seating configuration of 1 to 9 seats, there must be at least one Type IV or larger over-wing exit on each side of the fuselage or, if over-wing exits are not provided, at least one exit on each side that meets the minimum dimensions of a Type III exit.

(2) For a passenger seating configuration of more than 9 seats, each exit must be a Type III or larger exit.

(3) For a passenger seating configuration of 10 to 19 seats, there must be at least one Type III or larger exit on each side of the fuselage.

(4) For a passenger seating configuration of 20 to 40 seats, there must be at least two exits, one of which must be a Type II or larger exit, on each side of the fuselage.

(5) For a passenger seating configuration of 41 to 110 seats, there must be at least

two exits, one of which must be a Type I or larger exit, on each side of the fuselage.

(6) For a passenger seating configuration of more than 110 seats, the emergency exits on each side of the fuselage must include at least two Type I or larger exits. [...]

(7) The combined maximum number of passenger seats permitted for all Type III exits is 70, and the combined maximum number of passenger seats permitted for two Type III exits on each side of the fuselage that are separated by fewer than three passenger seat rows is 65.

(8) If a Type A, Type B, or Type C exit is installed, there must be at least two Type C or larger exits on each side of the fuselage

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ASSIGNMENT 3.8

Based on the maximum number of passengers and the applicable airworthiness regulations,

a. Define how many emergency exits are required and;
b. on which side of the fuselage each exit needs to be placed.

The next requirements that stem from the regulations stem from subpart E, 'Powerplant.' In this subpart, you typically find the requirements that have to do with the installation of the powerplant. However, it must be noted that there are separate regulations regarding the powerplant itself (e.g. CS-E or FAR-33 for engines and CS-P or FAR-35 for propellers). Subpart E also specifies the requirements with respect to cooling, fuel systems, and the placement of firewalls. Here is an example of a regulation regarding propeller clearance in CS/FAR-25, which can also be found in CS/FAR 23.

Example 3.17

CS 25.925 Propeller clearance

Unless smaller clearances are substantiated, propeller clearances with the aeroplane at maximum weight, with the most adverse centre of gravity, and with the propeller in the most adverse pitch position, may not be less than the following:

(a) Ground clearance. There must be a clearance of at least 18 cm (7 inches) for each aeroplane with nose wheel landing gear or 23 cm (9 inches) for each aeroplane with tail-wheel landing gear, between each propeller and the ground with the landing gear statically deflected and in the level take-off, or taxiing attitude, whichever is most critical. In addition, there must be positive clearance between the propeller and the ground when in the level take-off attitude with the critical tyre(s) completely deflated and the corresponding landing gear strut bottomed. [...]

These examples from CS-23 and CS-25 show how detailed the requirements are specified. To comply with these requirements, the AMCs of Book 2 can be consulted. However, in addition to the AMCs, the FAA and EASA have issued so-called advisory circulars

(ACs). These documents provide guidance for compliance with the airworthiness regulations. Some of these ACs consist of merely a few pages, but others consist of tens of pages and are quite extensive. While the ACs are not regulations and are not mandatory to comply with, following the guidelines typically leads to compliance with the CS/FAR regulations. Therefore, they could be perceived as defacto airworthiness regulations. Advisory Circulars can be found online at www.faa.gov. The following example describes the design considerations that need to be taken into account when positioning turbine engines and auxiliary power units (APUs). An APU is a gas turbine that provides power to the airplane in case the main engines are off.

Example 3.18

AC 20-128A - Design Considerations for Minimizing Hazards caused by Uncontained Turbine Engine and Auxiliary Power Unit Rotor Failure

(1) Purpose. This advisory circular (AC) sets forth a method of compliance with the requirements of Sections 23.903(b)(1), 25.901(d), and 25.903(d)(1) of the FAR pertaining to design precautions taken to minimize the hazards to an airplane in the event of uncontained engine or auxiliary power unit rotor (compressor and turbine) failure and engine fan blade failures. [...]

(6) Definitions [...]

  (c) Uncontained Failure. For the purpose of airplane evaluations in accordance with this AC, uncontained failure of a turbine engine is any failure which results in the escape of rotor fragments from the engine or APU that could result in a hazard. Rotor failures which are of concern are those where released fragments have sufficient energy to create a hazard to the airplane. [...]

  (f) Fragment Spread Angle. The fragment spread angle is the angle measured, fore and aft from the center of the plane of rotation of an individual rotor stage, initiating at the engine or APU shaft centerline (see Figure 3.9). [...]

(7) Design Considerations. [...] The most effective methods for minimizing the hazards from uncontained rotor fragments include location of critical components outside the fragment impact areas or separation, isolation, redundancy, and shielding of critical airplane components and/or systems. The following design considerations are recommended:

  a. Consider the location of the engine and APU rotors relative to critical components, systems or areas of the airplane such as:

    (i) Any other engine(s) or an APU that provides an essential function;
    (ii) Pressurized sections of the fuselage and other primary structure of the fuselage, wings and empennage;
    (iii) Pilot compartment areas;
    (iv) Fuel system components, piping and tanks;
    (v) Control systems, such as primary and secondary flight controls, electrical power cables, wiring, hydraulic systems, engine control systems, flammable fluid shut-off valves, and the associated actuation wiring or cables; [...]

(10) Safety analysis. [...]

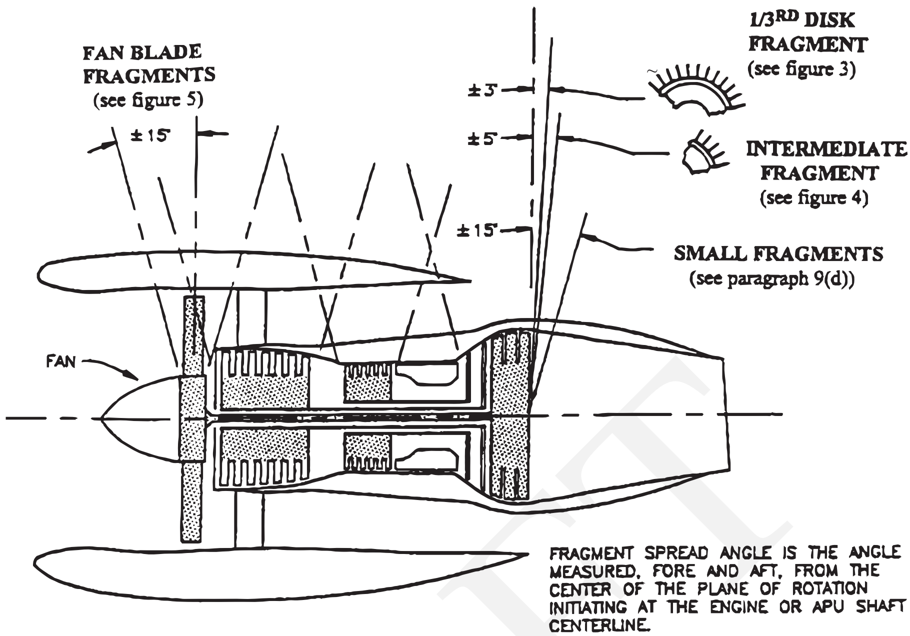


Figure 3.9: Estimated path of fragments (copy from AC 20-128)

c. For part 25 transport and part 23 commuter category airplanes, the following hazard ratio guidelines have been achieved:
   (i) Single One-Third Disc Fragment. There is not more than a 1 in 20 chance of catastrophe resulting from the release of a single one-third disc fragment as defined in Paragraph 9a.
   (ii) Intermediate Fragment. There is not more than a 1 in 40 chance of catastrophe resulting from the release of a piece of debris as defined in Paragraph 9. [...]

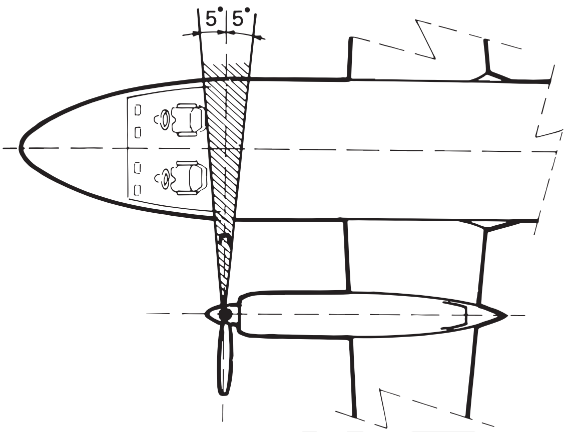
When installing propellers, you must also take into account that a propeller blade can break off during rotation. Similar to the turbine discs in the previous example, these broken propeller blades could cause major damage when they impact with airframe. Therefore, several design requirements in CS/FAR 23 and CS/FAR 25 pertain to the installation of the propeller with respect to critical systems. In CS/FAR 25.771, the propeller location with respect to the pilot compartment is presented. CS/FAR-25.905 specifies what precautions need to be taken in order to minimize the risk of catastrophic failure. In addition, AC 25.905-1 presents further design guidelines for the safe installation of propellers. The following example presents text from these two documents. When we discuss power plant installation in Chapter 4, we will address what implications these requirements have on the feasible engine locations.

Example 3.19

CS 25.771 Pilot compartment

(b) The primary controls [...], excluding cables and control rods, must be located with respect to the propellers so that no member of the minimum flight crew [...], or part of the controls, lies in the region between the plane of rotation of any inboard propeller and the surface generated by a line passing through the center of the propeller hub making an angle of five degrees forward or aft of the plane of rotation of the propeller.

The image below shows an interpretation of this requirement for a wing-mounted propeller and has been produced by Torenbeek [20].



CS 25.905 Propellers

(d) Design precautions must be taken to minimise the hazards to the aeroplane in the event a propeller blade fails or is released by a hub failure. The hazards which must be considered include damage to structure and critical systems due to impact of a failed or released blade and the unbalance created by such failure or release.

AC 25.905-1 - Minimizing the Hazards from Propeller Blade and Hub Failures

7. Design Practices to Minimize Blade Fragment Hazard

   a. General

      (1) Techniques defined in AC 20-128A for minimizing the hazards following an uncontained engine rotor failure (i.e. separation of critical systems, isolation of functions, redundancy of functional elements, or shielding) are also applicable when minimizing damage from propeller blade fragments. However, the numerical assessment of fragment size defined in paragraph 9 of AC 20-128A is not applicable for the propeller.

      (2) Applicants should take all practical precautions in the airplane's design to minimize, based on good engineering judgment, the risk of catastrophic effects due to the release of part of a blade or a complete blade.

A final example that we include here is a set of requirements defined by the International Civil Aviation Organisation or *ICAO*. ICAO fosters the development of international air transport to ensure safe and orderly growth.[9] Their Annex 6 "Operations of Aircraft" prescribes how much fuel an airplane needs to take on board to ensure a safe flight. A distinction is made between *commercial aviation* and *general aviation*. Commercial aviation is that part of aviation where operators provide a transportation service through the purchase of a ticket. General aviation is everything else, except for military aviation. For general aviation, the distinction is made between flying under Instrument Flight Rules (IFR) and Visual Flight Rules (VFR). Whether you fly an airplane under IFR or VFR depends on the weather conditions. Note that the wording in the example below only implies fuel as an energy source. In this text, we assume that the same requirements would apply to battery-powered airplanes and that the word "fuel" would be replaced by the word "energy."

Example 3.20

For **commercial aviation**, ICAO Annex 6, Part I, Section 4.3.6 "Fuel Requirements" requires the fuel quantity ahead of a flight to include:

- Taxi fuel
- Trip fuel (to reach intended destination)
- Contingency fuel (5% of "trip fuel" or 5 minutes of holding flight, whichever one is more)
- Alternate destination fuel (to fly a missed approach and reach an alternate airport)
- Final reserve fuel (45 minutes of holding flight for reciprocating engines, 30 minutes for turbine engines)
- Additional fuel (if needed to guarantee ability to reach an alternate with an engine failure or at lower altitude due to a pressurization loss)
- Discretionary fuel (if the pilot in command wants it)

For **general aviation**, ICAO Annex 6 Part II, Section 2.2.3.6 "Fuel and oil supply" requires the fuel quantity ahead of a flight to include:

- For IFR, enough fuel to reach destination, then alternate (if required), plus 45 minutes
- For day VFR, enough fuel to reach destination plus 30 minutes
- For night VFR, enough fuel to reach destination plus 45 minutes

Note that the fuel requirements from the example above partially overlap with the ones we deduced from our mission profile of a transport airplane (Example 3.3). You can use the requirements from ICAO's Annex 6 to harmonize the overall requirements you defined based on your mission profile. Note that ICAO does, in some cases, require the airplane to reach an alternate airport in case of emergency without specifying how far this alternate airport is away from the destination airport. Therefore, this range should be decided and made explicit in the top-level requirements. Finally, in case you design an electric airplane, it can be assumed that the word *fuel* should be replaced with *energy*.

In this section, we have provided you with numerous examples stemming from the airworthiness regulations as presented in the CS/FAR documents. Many of the presented examples are going to be used in the design process of your airplane. Also during the de-

[9] See: online description of ICAO.

sign process itself, we might introduce more requirements stemming from the airworthiness regulations to ensure compliance later on. It is by no means necessary to know all the regulations by heart, but it is important to know where to find the relevant documents and how to use them in the design process.

3.4. SELECTION OF REFERENCE AIRPLANES

When you perform a design process, you start with very little knowledge about your design. You might have selected a concept, but how do you start analyzing it? This is a typical chicken-and-egg problem where you need an airplane to analyze something and analysis results to design an airplane. To avoid this problem, you need to make assumptions about some aspects of the design before you can perform any sizing or design step. During the design process, these assumptions might be replaced in subsequent iterations of the design. If an assumption is replaced by an analysis result, it does not need to be communicated in a design report. However, if an assumption is used throughout the design process, you should report it explicitly in a design report. This is important because the assumption might relate to an analysis result of the design and therefore affect one or more performance metrics of the design.

The assumptions that are needed to start the design process pertain to aspects of the design that are not known yet. For airplanes, we typically look at previous airplanes to quantify these aspects. The best airplanes to choose in this respect are airplanes that have been designed for a similar set of TLARs as the ones that you are using for your own design. For example, if you need to design a 6-seater passenger airplane, it makes sense to look for other airplanes in the 4-8 seat range. The airplane that you use in the design process to derive assumptions from are your *reference airplanes*. The following example illustrates this.

Example 3.21

You have been tasked to design an airplane to transport 50 passengers over a distance of 1000 km (560 nmi). Which reference airplanes would you choose to base your assumptions on?

A quick search on the internet tells us that we are dealing here with a regional airliner. Based on the TLARs of range and pax, we can quickly find the following five airplanes:

Manufacturer	Name	Pax	Range (km)
ATR	ATR 42-600	48	1300
Antonov	An 148	68	2400
Bombardier	CRJ700	70	940
De Haviland Canada	Dash 8 Q400	80	1300
Fokker	F50	50	1700

In Example 3.21, we only selected five reference airplanes. In practice, it is often beneficial to have more than that. Typically, between 5 and 10 airplanes are sufficient for the design process that we use here. Depending on the design specification, you might be able to find more or less reference airplanes. Regional airliners of 50 passengers (or around that number) have been around since the 1950s. Therefore, you could have a long list of potential reference airplanes. However, older airplanes might not have the same technology as newer airplanes. Therefore, newer airplanes get priority over older airplanes when you have to choose. Usually, the reference airplanes can differ somewhat from the specifications you have been given. In the previous example, you might have noticed that the range of most airplanes is larger than 1000 km. Similarly, the number of passengers is also somewhat higher than specified. This is almost unavoidable and should not negatively influence the design process. However, when selecting reference airplanes, priority should be given to the airplanes that mimic the specifications as closely as possible. In Example 3.21, the ATR 42-600 comes closest to the specification (see Fig. 3.10).

Figure 3.10: The ATR 42 is selected as one of the reference airplanes in Example 3.21. Photo: KlausF .

As is evident from the design process of Figure 2.2, multiple steps in the design process require input from assumptions. When we discuss the sizing and analysis methods in the Chapters 7 through 9, this is made more explicit. In a design report, it is advised to present your reference airplane in more detail, including relevant data for the design process. The following example illustrates this for the ATR-42.

Example 3.22

The ATR 42-600 is a regional airliner. It has turboprop engines, a high-wing configuration and a t-tail. The main landing gear is connected to the fuselage and is retractable. The maximum payload mass is 5.45 tonnes, the empty mass is 11.5 tonnes, and the maximum take-off mass is 18.6 tonnes. It typically seats 48 passengers and has a cruise speed of 300 kts. The wing area is 54.5 m^2 and the engines each produce a maximum of 1,300 kW of shaft power at sea-level.[10]

[10]Data from `https://en.wikipedia.org/wiki/ATR_42`. Retrieved 25 May, 2022.

In a design report, it is good practice to include your reference airplanes explicitly before you start the design process. Not only does it provide you with a source to which you can compare your own design, but preparing the description also teaches you something about the airplanes you selected. By preparing a small synopsis of the airplane, you can study each airplane superficially. Perhaps you can find interesting aspects of each particular airplane that might give you ideas for your own design. Perhaps you find out information about why one reference airplane was very successful in terms of sales, while for others only a handful were built. This provides some context to the design you are making and could potentially help you in making design decisions later in the design process.

ASSIGNMENT 3.9

Based on the TLARs of the airplane you need to design, you need to select and present a set of five reference airplanes. The following tasks and questions will help you in the process:

a. List the following TLARs for your airplane design: number of passengers, payload mass, and mission range.
b. Find five airplanes whose specifications resemble the TLARs of the airplane you need to design. For each reference airplane, list the following items in a table:
    i. Manufacturer
    ii. Airplane name
    iii. Entry into service year
    iv. Number of passengers
    v. Maximum payload mass (kg)
    vi. Maximum range
    vii. Maximum take-off mass (kg)
    viii. Empty mass or operating empty mass (kg)[a]
    ix. Wing area (m^2)
    x. Wing span (m)
    xi. Maximum take-off power or maximum take-off thrust[b]
c. For each airplane, add a photo that shows what the airplane looks like.
d. Make a diagram with, on the vertical axis, the (operating) empty mass and, on the horizontal axis, the maximum take-off mass. Plot each reference airplane as a point in this diagram.

[a]Whichever one is listed in your source
[b]Whichever one is applicable to your design choice

4

CONFIGURATION SELECTION

In the previous chapters, you have estimated the lift, drag, and efficiency characteristics of your airplane. You have made design choices on the type of energy carrier, the number of engines, the aspect ratio, etc. You have computed the characteristic masses of the airplane and you have sized the wing and the propulsion system. The question that you will answer in this chapter is: what will your airplane look like?

There is a large variation in airplane configurations that have appeared over the last century. In Figure 4.1 we show three examples. We can see that the wing can be positioned high or low and that the engines can be mounted on the wing, on the fuselage, or buried in the nose of the airplane. We also notice some variation in the tail design. The reason why these airplanes have different configurations has to do with their functionality. Each of them needs to comply with a different set of requirements. Some of these requirements might play an important role in choosing the most appropriate configuration. The second reason for preferring one configuration over another is optimizing the design objective. The assignments in this chapter help you to select an airplane configuration that meets your design requirements while optimizing your design objective.

In Section 1.1, you have seen that the third step of the engineering design process is to set up design options that can solve your design problem. The choice of your airplane configuration is an example of such a design option. If you select more than one design option, you must complete the entire design process for each option before you can

Figure 4.1: Examples of various airplane configurations. Photos by Alan Wilson, Pedro Aragao, and Anna Zvereva.

perform a trade-off and choose one of them as your design solution. In this chapter, we ask you to choose a single design option to get experience with this process. Contrary to the quantitative work that we performed in Chapters 5 and 7, this chapter is much more qualitative in nature. We also limit the discussion of airplane configurations to "conventional" configurations that all have the same ingredients: a fuselage to store the payload, a wing to provide lift, a propulsion system to provide thrust, a tail to provide stability and control, and a landing gear to interface with the ground surface.

In this chapter, we have broken down the airplane configuration into four main aspects: the integration of the energy and propulsion system (Section 4.1), the wing positioning (Section 4.2), the landing gear configuration (Section 4.3) and the tail configuration (Section 4.4). While we treat these configurational aspects in distinct sections of this chapter, we will also stress important cross-links between them. Going through the examples and assignments in this chapter, you can make a well-funded decision about the airplane configuration you are designing.

4.1. ENERGY AND PROPULSION SYSTEM INTEGRATION

All powered airplanes have a propulsion system on board that provides thrust to the airplane. To provide thrust, we need an energy source and a machine to convert that energy source into thrust. In this textbook, we consider two types of energy sources: chemical energy and electrical energy. Chemical energy is stored in a fuel, while electrical energy is stored in a battery. Fuel can be converted to mechanical power (or *shaft power*) by combusting it in a piston engine or a gas turbine. A gas turbine that drives a propeller is termed a *turboprop* engine. An example of a turboprop engine is the Pratt & Whitney PW120, which is depicted in Figure 5.10. Electrical power can be converted to shaft power by an electric motor. With the available shaft power, a propeller can be driven, which produces thrust. In a turbojet engine, fuel is combusted to produce thrust by accelerating the combustion products through a nozzle. If a fan (i.e., a multi-bladed ducted propeller) is connected to a shaft of the turbojet engine, we call this a *turbofan* engine. These four propulsion systems are schematically depicted in Figure 4.2.

In Chapter 3 we presented some of the requirements that pertain to the integration of propellers and jet engines. If you are designing a propeller airplane, it is advised to read through Examples 3.17 and 3.19. They describe the clearance requirements and how to prevent catastrophic failure in case of a blade failure of the propeller. You can consult Example 3.18 for turbine engines such as turbofans and turboprops. It tells you to position the engine in such a way that structural failure of a turbine disk fragment does not lead to catastrophic failure of the airplane. This should be considered when positioning a propeller/engine with respect to other propellers/engines on the airplane or with respect to critical systems such as the fuel system, the flight control system, or the fuel tanks.

In this section, you will decide what energy carrier your airplane will have, what propulsion system your airplane will have, and how the propulsion system will be integrated. We will present various engine integration options and discuss the advantages and disadvantages of each. Then, at the end of the section, you are asked to decide how you wish to integrate your propulsion system with your airframe.

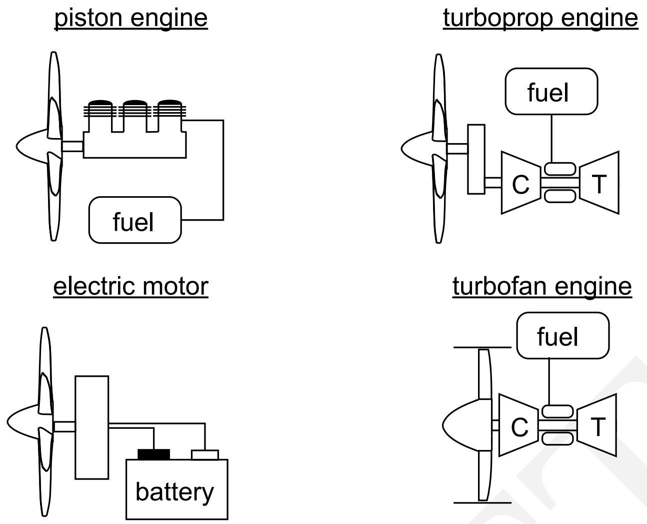


Figure 4.2: Schematics of four different propulsion systems that are considered in this book. C = compressor, T = turbine.

4.1.1. CHOOSING YOUR ENERGY CARRIER

In this book, we consider two types of energy carriers: batteries and fuel. Batteries store electrical energy in electrochemical cells stacked together in a pack. Batteries can be charged or discharged, but their mass remains constant. In aviation, typically, lithium-ion batteries are used due to their relatively high mass-specific energy. Fuel is a generic term for a liquid or gas that chemically reacts to produce heat or electricity. The most commonly used aviation fuels are avgas and Jet A (kerosene). Other fuel types include mogas, diesel, (liquefied) natural gas, synthetic kerosene, biofuel, and (liquefied) hydrogen. Fuels that do not result in a net production of carbondioxide in their lifecycle are referred to as *sustainable aviation fuels* (SAFs). In the following paragraphs, we will shortly discuss these energy carriers. In Ref. [2], a more detailed elaboration of the various energy carriers can be found.

Electric energy is typically stored in electrochemical cells. Many cells stacked together form a battery pack. In addition to the cells, a battery pack comprises a structure, internal wiring, and, often, a thermal conditioning system. The latter is required to prevent batteries from overheating during fast charging or discharging and to prevent the batteries from being too cold, reducing their energy-storing capacity. Lithium-ion batteries are typically used in electric aircraft, which offer relatively high specific energy and little degradation over their lifetime. When selecting a battery as your energy source, it should be noted that it should never be completely discharged during normal operation to prevent internal damage. Therefore, a 5-10% *state of charge* should remain. It should also be noted that charging a battery is much slower than fueling an airplane. This may be important for commercial airplanes that require a certain *turnaround time*, the time it takes to complete all the ground processes between arrival and departure. The advantage of battery technology is that the conversion of electric energy from the source to the propeller is very high compared to any of the options below. Therefore, less energy is “wasted” in the form of heat. Also, an electric aircraft produces no emissions during flight, although emissions could be generated in the production and transport of electricity. An example of a regional passenger airplane design with batteries as the primary energy carrier is shown in Figure 4.3.

Figure 4.3: Conceptual design of 90-passenger electric aircraft with a range of 800 km. Photo: Elysian-E9X. Used with permission.

Piston engines often use *avgas* (aviation gasoline). Avgas typically includes a toxic lead-containing additive to increase the octane level and reduce the chance of *knocking*, i.e., the premature detonation of fuel inside the cylinder. Because the lead-containing avgas causes polluting emissions, which can affect public health, *mogas* can be an alternative. Mogas, short for automotive gasoline, is an unleaded alternative. It is similar to gasoline for cars but does not allow any ethanol to be mixed into it, which is allowed for car engines. Mogas has a lower octane level than avgas, which increases the change of knocking, particularly at high pressure. Also, mogas is more susceptible to vaporization in the fuel lines when positioned close to the hot engine. This can cause bubbles in the fuel lines and interruption of the fuel flow. A third alternative is to use diesel in combination with a diesel engine. Like mogas, diesel is an unleaded fuel and pollutes less than avgas. However, diesel engines are heavier and bulkier than gasoline engines and, therefore, have a lower power-to-mass ratio and need a larger cowling. In terms of emissions, diesel engines produce more soot and *nitrogen oxide* (NO_x) emissions than gasoline.

In jet engines, particular grades of kerosene, such as Jet A fuel, are used. While avgas burns rapidly, which is ideal for piston engines, Jet A burns more steadily and is therefore more suitable to jet engines and turboprop engines. Similar to diesel, Jet A has lower volatility than avgas, which means it is safer to transport and handle. In 2024, kerosene fuel was used for 95% of all aircraft. Therefore, it is the dominant source of carbon dioxide (CO_2) production in aviation. To reduce the production of CO_2, biofuels and synthetic fuels can be used instead. Before discussing these alternatives, let us first look at the emissions created by a jet engine or turboprop engine.

Like other carbohydrate fuels, the combustion of kerosene produces water vapor and carbon dioxide as the primary combustion products. In addition, sulfur oxides, nitrogen oxides, and soot are being produced. Carbon dioxide is a greenhouse gas that stays in the atmosphere for a long period of time. We call these *long-lived* emissions. Nitrogen oxides react with other compounds within the atmosphere and cause a long-lasting warming effect. Water vapor itself has a small, short-lived effect on global warming. However, condensation trails can form when water vapor condensates on soot particles. The formation of these condensation trails is highly dependent on the local weather conditions, and they can have both a warming effect during the night and a cooling effect during the day. However, the net effect of these so-called *contrail cirrus* is a short-lived global warming effect. The global warming effect of these various emission species is described in more detail in Ref. [5].

The first alternative to traditional kerosene is *biofuel*. As the word suggests, biofuel is a fuel obtained through a biological process. Various feedstock options exist to create biofuel, ranging from food crops to waste cooking oil. When combusting biofuel in a jet engine, the CO_2 that has been absorbed from the atmosphere by the feedstock is brought back into the atmosphere. In other words, biofuel has a much reduced *carbon footprint* compared to kerosene. One aviation-grade biofuel is based on HEFA (Hydrotreated Esters and Fatty Acids), a process to turn waste cooking oil into jet fuel. One step beyond biofuel is synthetic kerosene, a fuel created from hydrogen and CO_2, where the hydrogen is produced from electricity and water through electrolysis. The required CO_2 can either be captured from existing smoke stacks or harvested from the air or ocean using a variety of processes. While the properties of synthetic kerosene are similar to fossil-based kerosene, it produces much fewer soot particles when combusted. Therefore, it reduces the formation of contrail cirrus at high altitudes.

Hydrogen is a fuel that does not contain carbohydrates and, therefore, produces no carbon dioxide upon combustion. Compared to kerosene, hydrogen has a much wider range of flammable concentrations, which means it ignites more easily. Compared to synthetic kerosene, it requires fewer production steps and can be made purely from electricity. However, since the combustion product is only water vapor, hydrogen combustion can lead to the formation of more persistent contrails compared to fossil kerosene, depending on the local atmospheric conditions. The effect of these persistent contrails on global warming is the subject of scientific studies. To effectively store hydrogen, it needs to be liquefied by reducing the temperature to 20 K. Therefore, dedicated cryogenic tanks are required to keep this low temperature and sustain a certain pressure to keep the hydrogen in a liquid state. While the fuel itself has a very low mass per unit of energy, the required mass of the tank partially negates this advantage. Secondly, under cryogenic conditions, liquid hydrogen (LH2) occupies four times as much volume as kerosene for the same energy content. This means that much larger fuel tanks are required. Instead of combusting hydrogen, it can also be used in a *fuel cell* where it is combined with oxygen to produce water and electricity. We qualify this as a *hydrogen-electric* airplane. Depending on the atmospheric conditions, contrails could be produced by the water from fuel exhaust. However, contrary to the combustion of hydrogen, there is no formation of NO_x. An example of a hydrogen-electric airplane is shown in Figure 4.4.

Figure 4.4: Example of an airplane with liquid hydrogen fuel combined with a single fuel cell to power a 4-seat airplane. Photo: Felix Oprean for DLR

Liquefied natural gas (LNG) might be seen as a simpler alternative. LNG is the liquefied form of natural gas, a fossil fuel primarily consisting of methane. It needs to be stored at 111 K and has two-and-a-half times the energy per unit volume compared to hydrogen. Because of its chemical composition, LNG reduces the formation of CO_2 by 25%, produces less soot, and less NO_x compared to fossil kerosene. While the combustion of natural gas produces more water, the reduction in soot is likely to reduce contrail formation compared to fossil kerosene.

To compare the various energy carriers, Table 4.1 lists their density, ρ, their mass-specific energy e, and their energy density, ρ_e. These values can be used when estimating the mass and volume required to store the energy on board. In Chapter 5), we distinguish between the mass-specific energy of fuel, e_f, and batteries, e_{bat}. For the lithium-ion battery, the specific energy is typically reported in Wh/kg, which is easily converted to J/kg. The following relation converts Watt-hour (Wh) to Joules (J):

$$1\ (\text{Wh}) = 1\left(\frac{\text{J}}{\text{s}}\right) \times 3600\ (\text{s}) = 3600\ (\text{J}) \tag{4.1}$$

The values in Table 4.1 can mostly be used directly in calculations of the fuel mass. However, for LNG and LH2 airplanes, the mass of the fuel tank is rather large due to the fact that they need to hold pressurized cryogenic fuel. To that end, we define a gravimetric efficiency as follows:

$$\eta_f = \frac{m_f}{m_f + m_t} \tag{4.2}$$

Table 4.1: Properties of various energy carriers.

	ρ (kg/m³)	e (MJ/kg)	ρ_e (MJ/l)
avgas	730	44	32
mogas	750	43	32
diesel	840	43	36
LH2	70	120	8.4
LNG	450	49	22
Jet A	820	43	35
biofuel (HEFA)	780	44	34
Synthetic kerosene	805	44	35
Li-ion battery (300 Wh/kg)	2000	1.1	2.2

where m_{f} is the fuel mass and m_{t} is the tank mass. For a hydrogen tank, the gravimetric efficiency is highly dependent on the dimensions, insulation, and maximum pressure. Values in the literature range from 0.2 to 0.35, although optimistic values above 0.7 can also be found. For LNG tanks, this value is between 0.4 and 0.5. The effective specific energy of these cryogenic fuels can then be calculated as follows:

$$e_{\mathrm{eff}} = \eta_{\mathrm{f}} \frac{\rho_e}{\rho} \tag{4.3}$$

Secondly, liquid hydrogen tanks and LNG tanks need to allow volume for the liquid fuel to evaporate, which requires 10-15% more volume than allocated based on the required energy. In other words, the *volumetric efficiency*, $\eta_V \approx 0.85$. The *effective* energy density of a cryogenic fuel can therefore be computed as follows:

$$\rho_{e,\ \mathrm{eff}} = \eta_V \rho_e \tag{4.4}$$

The following example shows how this works.

Example 4.1

In this example, we compute the effective specific energy and effective energy density of liquid hydrogen fuel.

Based on the literature, we assume a gravimetric efficiency of $\eta_{\mathrm{f}} = 0.4$. This means that of the total tank plus fuel mass, 40% is fuel mass, and 60% is tank mass. Secondly, we assume a volumetric efficiency of $\eta_V = 0.85$ The effective mass-specific energy of the fuel is then:

$$e_{\mathrm{eff}} = 0.40 \cdot \frac{8.4 \cdot 10^9}{70} = 48 \text{ (MJ/kg)}$$

The effective energy density becomes:

$$\rho_{e,\ \mathrm{eff}} = 0.85 \cdot 22 = 19 \text{ (MJ/l)}$$

With this assumed value of the gravimetric efficiency, the effective mass-specif energy of liquid hydrogen is close to the value for Jet A (see Table 4.1).

The values for the mass-specific energy and energy density are plotted in Figure 4.5. Towards the right in this figure, a large cluster of fossil fuels and their substitutes can be seen. The battery is all the way in the bottom right corner of this figure. The cryogenic fuels have a blue marker indicating their physical properties and red markers indicating their effective properties for assumed values of gravimetric and volumetric efficiency.

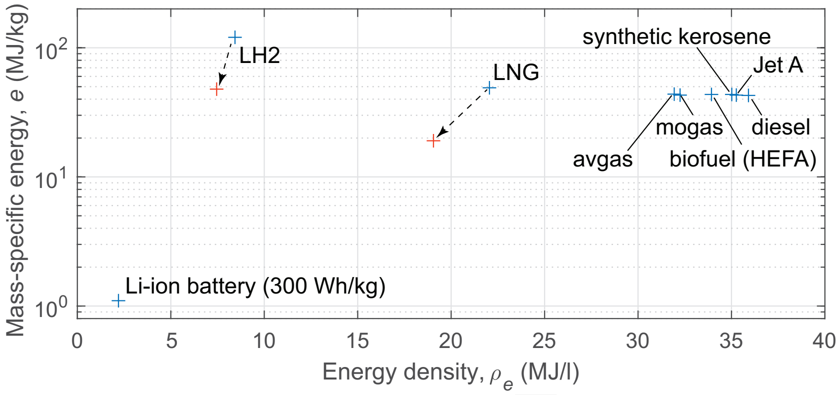


Figure 4.5: Graphical representation of the energy per unit mass and the energy per unit volume for various energy carriers used by fixed-wing airplanes. The red markers for the cryogenic fuels show the effective values assuming $\eta_f = 0.4$ and $\eta_V = 0.85$.

ASSIGNMENT 4.1

In this assignment, you choose the energy carrier of your airplane.

a. What energy source do you choose for your airplane? Substantiate your choice.
b. What is the specific energy of your energy source?
c. If you have chosen a cryogenic fuel, what values have you assumed for the gravimetric efficiency and volumetric efficiency?
d. If you have chosen a cryogenic fuel, what is the *effective* specific energy of your fuel?

4.1.2. INTEGRATION OF THE ENERGY CARRIER

Once an energy carrier has been chosen, the question arises where to stow it. In theory, any airplane component that has a hollow internal volume could be used to store energy. However, that might not be a very practical choice. The most likely locations to store the energy carrier are in the wing and/or in the fuselage. Depending on the type of energy carrier, either one might be preferred over the other.

There is a large advantage of storing your energy in the wing. The wing volume is typically not used for storing payload, so it does not require additional volume in the fuselage. When you store fuel in the wing, the bending moment induced by the lift distribution is reduced. We refer to this as *bending moment relief.* This translates into a lower wing weight compared to the scenario where all the energy would be stored in the fuselage. So regardless of the energy carrier, the good rule of thumb to abide by is: “If it can be stored in the wing, it should be stored in the wing.” Airplanes with fuel as their energy carrier typically have fuel tanks in their wings or have the wing structure double as fuel tanks.

However, there could be good reasons not to store the fuel in the wings. Small aircraft, for example, could have detachable wings, which make it inconvenient to store fuel there. Cryogenic fuels should be stored in pressurized tanks, which typically do not fit inside the wing. They could, therefore, be integrated into the fuselage. Alternatively, external tanks could be suspended from the wing. Batteries

ASSIGNMENT 4.2

In this assignment, you make a preliminary decision on how to integrate your energy carrier with the airplane.

a. Where do you intend to store the energy carrier that you selected in Assignment 4.1? Explain why you have made this decision.

4.1.3. INTEGRATION OF SINGLE-PROP PROPULSION SYSTEMS

For the single-prop propulsion systems, we typically put the propulsion in the symmetry plane of the airplane. Most single-prop airplanes have a *tractor* configuration, meaning the propeller is positioned in the nose of the airplane, ahead of the center of gravity. The propeller airplanes in Figure 4.13 are examples of single-prop airplanes where the propulsion system is integrated into the nose of the airplane. If the propeller is positioned behind the center of gravity, we call this a *pusher* configuration. . An example of either configuration is provided in Figure 4.6.

Figure 4.6: Example of single-engine propeller airplanes with a tractor configuration (left) and a pusher configuration (right). Photos: Pipistrel and Stephen Kearney.

As can be seen in Figure 4.6, a propeller in the back of the fuselage prevents the integration of a conventional tail. Therefore, a foreplane is added to balance the airplane around the center of gravity. Due to the relatively heavy propulsion system, the wing is positioned aft on the fuselage to provide lift as well as stability. For a tractor configuration, the wing is located further forward on the fuselage, and a long arm is formed between the tail and the wing. One advantage of the tractor configuration is the fact that the propeller receives clean flow, while the air to a pusher propeller has been influenced by the presence of the fuselage upstream. This means that the propeller efficiency for a tractor configuration is somewhat higher than for a pusher configuration.

4.1.4. INTEGRATION OF MULTI-PROP PROPULSION SYSTEM

If you design an airplane with two propellers, there are several options to integrate the propeller and engine/motor with the airframe. The most common configuration has the two propeller engines mounted to the wing in a tractor configuration. Alternatively, one can combine a tractor propeller and a pusher propeller in the *in-line* configuration (see Figure 4.7). The in-line configuration has the advantage that a single engine failure does not result in a *yawing moment*.[1] This could reduce the required size of the vertical tailplane. As for all pusher propellers, care must be taken that the propeller does not touch the ground when the airplane rotates during take-off. Therefore, the main landing gear must be sized appropriately. Furthermore, integration with the tail could prove challenging. Therefore, the example of Figure 4.7 shows a twin vertical tail supported by two tail booms.

Figure 4.7: Integration of two propeller engines can be done in line with the fuselage (left) or attached to the wing (right). Photos: Wally Cacsabre and Michael Miley.

The configuration with twin propellers mounted below the wing allows the nose to be tailored for excellent pilot visibility. Furthermore, the mass of the engine causes a bending moment in the root of the wing that opposes the bending moment introduced by the distributed lift force on the wing. The resulting wing-root bending moment is, therefore, lower compared to the in-line engine configuration. This results in a lower mass of the wing box. When positioned in a tractor configuration as in Figure 4.7, the slipstream of the propeller causes the lift over the wing to increase. This has implications for the stability and control of the airplane that goes beyond the scope of this book but is explained in Ref. [8]. The airplane of Figure 4.7 also shows a strut supporting the wing. While the strut increases the wetted area of the airplane and, thereby, the drag, it does reduce the bending moments in the wing significantly and, thereby, the wing mass.

Figure 4.23 shows that also a wing-mounted pusher configuration is possible for a twin prop. This has the advantage that the propeller noise is less high in the passen-

[1] A yawing moment is a moment about the airplane's vertical axis

ger cabin. However, care must be taken that the wing wake and engine exhaust do not cause an increase in noise when they pass through the propeller disk. This could create a higher fly-over noise for people on the ground. Two other examples where the engines are located on the rear fuselage are shown in Figure 4.8. To support the engine and propeller, a relatively large *pylon* is required. The pylon is the structural component that connects the engine (or motor) to the airframe. The propfan shown in the right-hand picture of Figure 4.8 is a cross-over between a turboprop engine and a turbofan engine.

Figure 4.8: Example of fuselage-mounted propellers (left) and fuselage-mounted propfans (right). Photos by Tim Rees and Andrew Thomas, respectively.

If more than two propellers are selected, then they are typically mounted to the wing. Figure 4.9 shows examples of four-engined airplanes with wing-mounted engine installation. In a four-engine configuration there is statistically a larger chance that one of the engines fails. Furthermore, more engines also mean that the maintenance costs are higher. Finally, smaller engines suffer from a reduction in thermal efficiency when they are smaller. However, there can still be good reasons to choose more than two engines. One advantage is that in case of an engine failure, the loss in thrust is only 25% for a four-engined airplane. Secondly, there might not be an engine that is powerful enough to provide half of the required take-off power. In that case, you have to spread the take-off power over (at least) four engines. A final reason could be that the combination of flight speed and engine power would result in a poor match between the propeller and the engine/motor. In Chapter 8, we will further detail how the propeller can be designed to absorb the power of the engine/motor.

Although less common, choosing more than four engines can prove a feasible design solution. Electric motors, for example, do not suffer from a loss in efficiency with size. Furthermore, due to the lack of hinging and sliding components within the motor, they are also less maintenance prone. To have a good match between the propeller and the motor, it can be desirable to distribute the power over more than four motors. This concept of *distributed propulsion* has additional benefits: the aerodynamic interaction with the wing is spread over the span, which improves the span-wise lift distribution in cruise flight, the one-engine inoperative condition has little impact on the flight performance, and the smaller propellers make it easier to integrate a short landing gear in case of a low-wing configuration.

Figure 4.9: Four-engined airplanes with a high-wing configuration (left) and a low-wing configuration (right). Photos by Ronnie Macdonald and RuthAS, respectively.

4.1.5. INTEGRATION OF JET ENGINES

While single-engine jet airplanes exist (see Figure 3.6), this section is focused on the integration of two or more engines with the airframe. Low bypass-ratio engines, such as those found in military combat airplanes or in the first generation of passenger jets, can be embedded in the fuselage or wing root. However, modern turbofan engines have such a large fan diameter that integration in the fuselage or wing is virtually impossible. Therefore, for twin-jet airplanes, the choice comes down to fuselage-mounted or wing-mounted. In Figure 4.21, we have already shown these options, and we will discuss some more characteristics of each option.

For wing-mounted engines, the mass of the engines relieves the bending moment of the wing, which reduces its structural mass. On a low-wing, the engine can be easily reached for inspection or for maintenance. However, the aerodynamic and structural integration with the wing are quite challenging. While the engine receives clean airflow regardless of the angle of attack or sideslip angle of the airplane, the wing experiences a flow field that is affected by the engine. Care must be taken to prevent so-called *interference drag* in the vicinity of the pylon, wing, and nacelle. Also, aerodynamic modifications are required to prevent an early onset of stall at the location where the pylon interrupts the leading edge of the wing. Finally, engines that are close to the ground can easily suck in debris when they are operated at low (taxi) speeds and relatively high thrust.

The fuselage-mounted engines result in a clean wing, which prevents the complicated aerodynamic and structural interaction of the wing-mounted engine. However, the structural mass of the wing is higher for this engine configuration. On the other hand, a one-engine inoperative condition results in a smaller yawing moment because the engines are located closer to the symmetry plane. Also, the distance from the thrust vector to the center of gravity is smaller, which reduces the change in *pitching moment* due to thrust.[2] Positioning the engine on the rear fuselage reduces the cabin noise for most passengers. Also, the wing partially shields the fan noise to ground observers, which reduces the fly-over noise. On the other hand, the large mass of the propulsion system causes the wing to be positioned further rearward for balancing reasons. This leads to a larger center-of-gravity excursion during the loading and unloading of fuel and passengers (see Chapters 9, which 12) results in a larger horizontal tailplane. Finally, due

[2]A pitching moment is a moment about the lateral axis of the airplane

to the exhaust of the engine, the horizontal tailplane needs to be raised, resulting in a cruciform tail or a T-tail (see Section 4.4).

An alternative integration solution is to use the so-called *over-the-wing* engines (see Figure 4.10). In such a configuration, the engines are positioned on pylons on the upper surface of the wing. This allows for a short landing gear, the engine mass reduces the wing bending moment, and the thrust vector is aligned with the center of gravity of the airplane. There are two main challenges with this configuration: (1) the aerodynamic interaction between the upper surface of the wing and the engine inlet and (2) the structural integration of a vibrating engine on top of a flexible pylon. To reduce the aerodynamic interaction, the engines are vertically separated from the wing with a relatively large pylon, which increases the structural mass and the friction drag of the airplane.

Figure 4.10: Integration of a jet engine on a large passenger airplane (left) and a business jet (right). Photos by Alan Wilson and Michael Pereckas CC BY, respectively.

Three-engined jets and four-engined jets reduce the impact of an engine failure on the flight performance of the airplane. On a three-engined airplane, the integration of the third engine is typically done in the tail of the airplane. A critical aspect of the integration of this engine is the fact that the engine needs to be easily removable from the airframe for maintenance. The engine can either be embedded in the aft fuselage or positioned in the vertical tailplane as shown in Figure 4.11. When integrated into the fuselage, an s-duct connects the inlet to the engine. Both configurations have their challenges regarding the structural integration of the aft fuselage, tailplanes, and engine. Also, aerodynamically, there is a strong interaction between these components, which requires careful aerodynamic design in the preliminary design phase.

Modern four-engined jet airplanes all have the same configuration: four engines hanging under the wing (see Figure 4.12). The engines are distributed to find a balance between a low wing mass on the one hand and an acceptable vertical tail size to balance a one-engine-inoperative yawing moment. For very large airplanes, the selection of four engines can be the only option to have sufficient thrust with the available engines. While the under-the-wing engines are the only feasible option for high-bypass-ratio engines, other integration solutions have been found for low-bypass-ratio engines such as integration in the wing root (de Havilland Comet), pairwise installation on the rear fuselage (Ilyushin 62 and Vickers VC-10), or pairwise installation under the wing (Concorde). The B-2 stealth bomber even has its four engines completely embedded in the fuselage.

Figure 4.11: The integration of the center-engine can be done in the tail (left) or embedded in the aft fuselage (right). Photos by Boushh-TFA and New York-air, respectively

Figure 4.12: Two examples of four-engined jet airplanes: the Airbus A380-800 (in the air) and the Boeing 747-8 (on the ground). Photo by Kiefer.

Figure 4.13: Examples of different wing positions: high-wing (left, mid-wing (center), low-wing (right). Photos: Péter Czégény, Alan Wilson, Textron

ASSIGNMENT 4.3

In this section, we have shown the advantages and disadvantages of various options to integrate the propulsion system with the airframe.

a. What propulsion system do you choose for your airplane? Why?
b. What is the cruise speed or cruise Mach number of your airplane?
c. How many engines/motors have you selected for your design? Please, motivate this decision.
d. How do you intend to integrate the engines/motors with your airframe? Why do you choose this integration solution?
e. For propeller airplanes, how do you intend to comply with blade failure requirements for the chosen configuration? See Example 3.19 for more details on blade failure.
f. For turbine engines, how do you intend to comply with turbine-disk failure requirements for your chosen configuration? See Example 3.18 for more details on turbine-disk failure.
g. Has the integration of the propulsion system changed your design decision regarding the wing configuration? If so, explain.

If you have not stated your climb gradient requirement(s) because these are dependent on the number of engines, you can now revisit **Assignment 3.7** and explicitly state the climb gradient requirement(s) for your airplane.

4.2. WING POSITIONING

In this section, we discuss the wing configuration: i.e., where the wing is positioned with respect to the fuselage. We only consider the vertical position: low-wing, mid-wing, and high-wing (see Figure 4.13). The longitudinal position of the wing will be determined later in Chapter 9. We will show the advantages and disadvantages of these three different wing positions. At the end of the section, you will be asked to make a well-funded decision about the position of the wing of the airplane you are designing.

The vertical position of the wing with respect to the fuselage influences many aspects of the airplane such as:

1. Aerodynamic interaction between wing and fuselage
2. Aerodynamic interaction between wing and horizontal tailplane

3. Accessibility to the fuselage
4. Type and positioning of engines
5. Positioning of landing gear
6. Airplane lateral stability
7. Airframe structural design
8. Crashworthiness
9. Pilot visibility

We will present each of the three wing positions and highlight how each of these aspects is affected.

4.2.1. HIGH-WING CONFIGURATION

Many military transport airplanes feature a high-wing configuration. The Lockheed C-5 is an example of a (very) large military transport airplane. In figure 4.14, we can see that one of the main advantages of the high-wing configuration is that it allows the fuselage to be close to the ground, which allows easy unloading of cargo from the airplane. This is an important requirement for this airplane, as it services airports with few facilities. Without the aid of cargo lifts, ramps can be used on the front side and rear side to simultaneously load or unload the cargo. To reduce the steepness of the loading ramps, the fuselage-mounted landing gear can be lowered, such that the airplane "kneels" during this process.

Figure 4.14: The Lockheed C-5 Galaxy is a military transport airplane with a high-wing configuration. Photos: Phylyp, Roland Balik, US Army Europe, and Eric Purcell.

The high-wing configuration of this airplane also allows for a larger vertical distance between the engines and the ground. This makes it easy to comply with ground clearance requirements without the need for tall landing gear. For airplanes that need to take off from unpaved runways, sufficient clearance between the ground and the engines needs to be preserved to prevent the engines from ingesting dirt or debris from the runway. This can cause so-called *foreign object damage* (FOD) to the engine. Figure 8.14 illustrates the dust and debris that is kicked up during take-off from a gravel runway. Another reason to increase the ground clearance could be to increase the distance from the deployed flaps to the ground. Seaplanes need to keep the wing and the installed engines out of the reach of the water spray. They also benefit from a high-wing configuration.

Figure 4.15: High-wing airplanes with wing-mounted engines allow for a larger ground clearance and reduce the risk of debris ingestion during landing and take-off. Photos: US Airforce.

If a high-wing configuration is selected, the main landing gear can either be connected to the wing or mounted to the fuselage (see Figure 4.16). The latter option has been chosen for the C-5 because the resulting landing gear height would be very large when connected to the high wing. When integrating the main landing gear with the fuselage, a support structure is added to the side of the fuselage, which ensures the landing gear has a sufficiently wide track. A fairing covering this structure and the retracted landing gear is called a *sponson* and causes an increase in (friction) drag. The additional support structure between the landing gear struts and the frames also increases the structural weight.

Figure 4.16: For a high-wing configuration, the landing gear can be mounted to the wing (left) or to the fuselage (right). Photos: Aero Mongolia, Anna Zvereva

A high-wing configuration also increases the *lateral stability* of the airplane, compared to the other two configurations. To understand the concept of lateral stability, we first examine what happens to an airplane when it experiences a *sideslip* condition. In a sideslip, the airplane is moving laterally with respect to the airflow direction. Such an event could occur when a lateral wind gust occurs or when the airplane is landing in cross-wind conditions. As the airplane is sideslipping, a *rolling moment* could develop.

[3] This is notionally shown in Figure 4.17. In this figure, three scenarios are shown: one where a restoring rolling moment due to sideslip occurs (stable), one where no rolling moment occurs (neutral), and one where a rolling moment occurs that aggravates the bank angle and increases the sideslip of the airplane (unstable). CS/FAR-23.177 and CS/FAR-25.177 require all non-aerobatic airplanes to have a stable rolling moment due to sideslip.

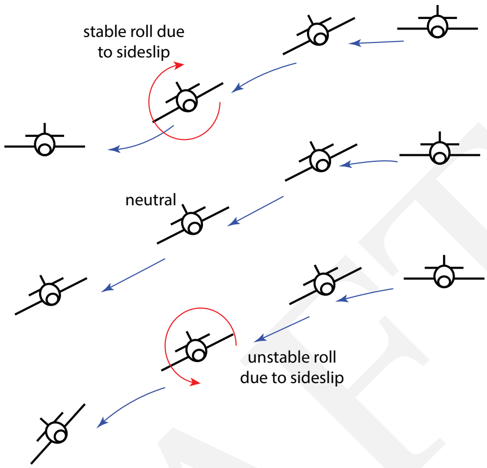


Figure 4.17: The rolling moment due to sideslip is dependent on the vertical position of the wing. Image after Whitford [27].

There are three ways to influence the lateral stability of an airplane with the wing:

1. Dihedral angle of the wing
2. Leading-edge sweep angle of the wing
3. Vertical position of the wing w.r.t. the fuselage

In Chapter 8 we will further discuss the effect of wing dihedral and wing sweep. Here, we focus on the vertical position of the wing with respect to the fuselage. To understand the effect of wing position on the rolling-moment-due-to-sideslip, consider Figure 4.18. When the airplane is slipping, a relative side wind component exists that causes the streamlines over the fuselage to migrate from left to right. For a high-wing configuration, the windward side of the wing therefore experiences a local increase in angle-of-attack near the fuselage while the leeward side of the wing experiences a local decrease in angle of attack near the fuselage. This results in an anti-symmetric force couple that causes a clockwise rolling moment. For the low-wing airplane, the effect of the cross-flow streamlines is the opposite: on the windward side, the lift reduces, while on the leeward side the lift increases. This causes a rolling moment in the counter-clockwise direction. A high-wing configuration of the wing therefore increases the lateral stability of the airplane, while a low-wing airplane decreases the lateral stability.

[3] A rolling moment is a moment about the longitudinal axis of the airplane that creates a *bank angle*.

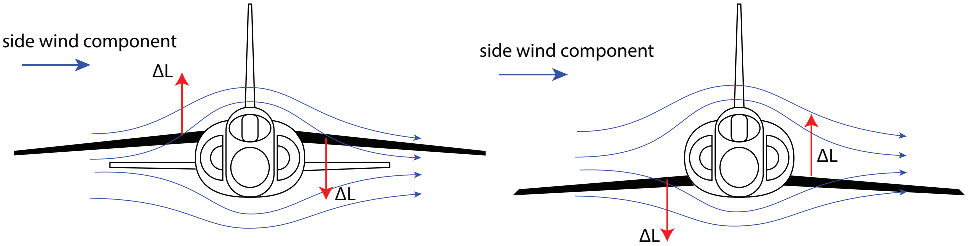


Figure 4.18: Change in lift over the wing due to side-wind for a high-wing configuration (left) and a low-wing configuration (right)

4.2.2. Low-Wing Configuration

The low-wing configuration is often seen on jet-powered civil-transport airplanes along with many business jets and general aviation airplanes. The low wing allows for the integration of the landing gear with a relatively short strut and a relatively wide track. Furthermore, the landing gear loads are introduced into the *wing box*, i.e., the load-carrying structure of the wing. Therefore, the additional structure that is required to transfer the landing gear loads towards the fuselage is relatively small. If the landing gear is retractable, the (short) main landing gear can be simply retracted using a single hinge point that swings the wheels laterally and stows them in the fuselage. This is at the expense of little wetted area increase. Alternatively, the landing gear can be stored in the nacelle behind the engine for airplanes with a wing-mounted propeller engine. In other words, the low-wing configuration allows for a low-drag and low-weight integration on the landing gear.

Figure 4.19: A low-wing configuration allows for an efficient integration with the landing gear. Photos: Poudou99, Bob Adams, Peter Haas.

Another advantage of the low-wing configuration is that the wing is easily accessible. For large airplanes, this allows the airplane to be easily fueled from below. Also, inspection of the wing can be easily performed on the platform prior to every departure. If engines are installed on or below the wing, they, too, can be easily checked and serviced.

Large passenger airplanes often have a low-wing configuration. Therefore, the fuselage is relatively far off the ground. This means that airports need to be equipped with all kinds of lifts and conveyor belts to transport luggage and cargo onto the airplane's cargo deck (Figure 4.20). Passengers need to board via air stairs or through a dedicated air bridge. Also, the trucks that service the galleys need a lifting platform that can interface with the main passenger deck. Without all of this ground equipment, the airplane cannot

Figure 4.20: Low-wing passenger airplanes need elevators and conveyor belts to transport cargo and luggage to the freight deck. Photos: Downtowngal, Jamesshliu

be properly serviced. However, there are also examples of airplanes with a low-wing configuration that are self-supporting, indicating that a low-wing transport airplane could also do without all of this equipment. In that case, creative solutions need to be found to elevate passengers and goods to the passenger deck and freight deck, respectively.

The gear should be tall enough for a low-wing configuration with a wing-mounted propulsion system to provide sufficient ground clearance. You can imagine that when the airplane's shock absorber and tire are completely compressed, the engine cowling or propeller blade should still have a proper clearance to the ground (see also Chapter 9). For wing-mounted jet engines with a high bypass ratio, requires the engines to be positioned ahead of the wing and require a long crane beam to connect the engines to the wing box. To increase the ground clearance between the engine and the ground, a few degrees of wing dihedral can help.[4] Alternatively, one can choose to position the engines on the fuselage, which might allow a shorter landing gear. However, this has other implications, as we discuss in Section 4.1. Figure 4.21 shows these two options.

Figure 4.21: For low-wing configurations, jet engines can either be integrated under the wing (left) or on the fuselage (right). Photos by Anna Zvereva and Dmitry Mottl.

The low-wing configuration reduces the lateral stability of the airplane, as we discussed in Sec. 4.2.1. However, increasing the dihedral angle of the wing can offset this effect and allow the airplane to comply with the lateral stability requirements.

[4]The dihedral angle is the angle between the wing and a horizontal line in the front view.

From a structural perspective, the easiest way to integrate a low wing with the fuselage is to mount the fuselage on top of the wing. This allows for an unobstructed fuselage structure connected to an unobstructed wing structure. For (near-) circular pressurized fuselages, a relatively large streamlined fairing between the wing and the fuselage is required. Such a wing-body fairing is common among business jets, and it increases the wetted area of the airplane. However, the unobstructed fuselage structure is relatively light. Another commonly found solution is to let the wing go through the fuselage below the main deck. This requires a smaller wing-body fairing and also brings the fuselage closer to the ground (see Figure 4.22). However, it does increase the mass of the fuselage structure as the structurally efficient circular tube has a large cut-out. A beneficial effect of the wing-body fairing is the increased available volume to stow the landing gear, house part of the flap-deployment mechanism, and provide space for the pressurization and air conditioning kit (PACK).

Figure 4.22: For pressurized fuselages, the wing structure can be positioned below the fuselage (left) or pierce through the fuselage below the passenger deck (right). Photos: James, Leo067

4.2.3. MID-WING CONFIGURATION

The mid-wing configuration is often applied to military combat airplanes. This wing configuration does not require a wing-body fairing and therefore minimizes the friction drag. To reduce the adverse aerodynamic interaction between wing and fuselage, often wing-body blending is applied. Furthermore, the lateral stability of the airplane is not influenced by the mid-fuselage location of the wing. For combat airplanes, it also results in excellent visibility for the pilot. In Figure 4.23, two examples are shown of airplanes with a mid-wing configuration.

From a structural point of view, the mid-wing configuration can pose a challenge. The highest bending moments occur at the root of the wing. Therefore, we prefer an unobstructed wing box from tip to tip. A passenger airplane with a mid-wing configuration would then have the wing box through the passenger cabin. For military combat airplanes, this problem is less acute because there are no passengers there. However, a large intake duct to the engine can be present, which could interfere with the wing structure. Another possibility is to position the wing behind the passenger cabin as is shown on the right-hand side of Figure 4.23. However, you then have to be sure that you can balance the airplane properly. In other words, you must make sure that the center of gravity is close to the center of lift in terms of their respective longitudinal positions. In