Figure 4.23: A military combat airplane (left) and general-aviation airplane (right) with a mid-wing configuration. Photos: José Luis Celada Euba and Tim Felce

the example of Figure 4.23 this is ensured by adding a foreplane.

> **ASSIGNMENT 4.4**
>
> In this section, you have seen examples of three wing configurations. Based on your top-level requirements and your design objective, what wing configuration do you choose for your airplane? Explain why you have chosen that configuration.

## 4.3. LANDING GEAR CONFIGURATION

In this section, we present the design options for the landing gear configuration. We first make a distinction between bicycle and tricycle landing gears. For the tricycle landing gears, which are most common, we then present the advantages and disadvantages of the two dominant configurations: the *tricycle landing gear* and the *conventional landing gear* (also known as "tail dragger"). Both configurations are shown in Figure 4.24. At the end of the section, you will be tasked to make a decision on the configuration of your landing gear. In Chapter 9 you will subsequently learn how to integrate your chosen landing gear with the airframe.


Figure 4.24: Example of a conventional landing gear for an aerobatic airplane (left) and a tricycle landing gear for a very large transport airplane (right). Photos by Velodenz and Vasiliy Koba, respectively.

### 4.3.1. CONVENTIONAL LANDING GEAR

The conventional landing gear consists of a main landing gear and a tail wheel. The main landing gear is positioned ahead of the center of gravity of the airplane to make sure the airplane can be balanced by the tail wheel. This configuration has been applied since the beginning of aviation; hence the term “conventional.” The integration of the tail wheel is relatively simple. It is small and it can easily be connected to the control cables of the rudder (or to the rudder itself) to provide steering to the airplane. When brakes are applied, the force on the landing gear increases, which makes the brakes very effective.

A tail-wheel airplane allows for a three-point landing or tail-wheel landing by stall. A tail-wheel landing is a maneuver where the pilot tries to land the airplane on the tail wheel first, then lets the airplane decelerate while keeping the stick back with the tail wheel on the runway until the wing stalls, and then the main gear touches the ground. The nose-high attitude on the ground with the stalled wing causes a lot of drag, quickly reducing the airspeed and the landing field length. A three-point landing is similar but with all three wheels touching the runway at the same time. Whether a three-point landing or a tail wheel landing is performed depends on the stall angle of the wing. Another advantage of the nose-high attitude is that during taxiing, the propeller has large ground clearance, preventing gravel or debris to be sucked into the propeller.

On the negative side, the high attitude during taxiing results in poor over-the-nose visibility for the pilot. Also, the advantage of high drag during landing can be a disadvantage during take-off, which reduces the acceleration of the airplane until the speed is high enough to raise the tail. In the case of a two-wheel landing with a large decent rate, the inertia of the airplane causes an increase in angle-of-attack, which might result in the airplane bouncing off the runway again.

While braking is more effective with conventional landing gear, it also comes with a nose-over and a ground loop risk. Both events are due to the fact that the airplane’s center of gravity is located behind the main landing gear. The nose-over event can occur when the main wheels suddenly come to a standstill, and the momentum of the airplane tips the airplane on its nose. This causes the propeller to hit the ground or sometimes even to flip the airplane on its back. A ground loop can occur when asymmetric braking is applied during deceleration. The resulting curved track that the airplane follows can be aggravated by the centrifugal force that acts at the center of gravity of the airplane, which resides behind the main landing gear. This destabilizing force can lead to large yawing and rolling motions if not appropriately corrected for by the pilot. In short, landing an airplane with a conventional landing gear is more difficult compared to landing an airplane with a tricycle landing gear.

### 4.3.2. TRICYCLE LANDING GEAR

The tricycle landing gear has the center of gravity ahead of the main landing gear. This prevents the problems of an unstable ground loop or a nose-over event. It also ensures that the airplane is in an upright position, which reduces the drag and provides a horizontal floor for the passengers and freight when on the ground. Furthermore, the pilot has good over-the-nose visibility during take-off and landing. Finally, in case of a large decent rate and a two-wheel landing, the inertia of the airplane causes the angle of attack to decrease, thereby shedding the lift and putting the nose wheel firmly on the ground.

Figure 4.25: A retractable landing gear requires volume to be stowed (left) and comprises many hinging parts (right). Photos by Victor and Julian Herzog, respectively.

All of these advantages make the tricycle configuration the most widely adopted landing gear layout. However, it comes with a price: mass. The nose landing gear (NLG) is much heavier than the tail wheel because it is larger and needs to sustain higher (dynamic) dynamic loads. Furthermore, to carry the point load of the nose wheel with the surrounding structure, local strengthening of the airframe is required, which increases the mass even further.

Despite the higher landing gear mass, the tricycle landing gear is used in all airplanes, ranging from very small two-seater airplanes (see Figure 3.4) to very large passenger airplanes (Figure 4.24). While some large airplanes feature multiple struts combined with many wheels on the main landing gear (as in Figure 4.12), we still characterize this as a tricycle landing gear. If you look at the examples that have been presented in this chapter, you can see many more tricycle landing gear layouts.

### 4.3.3. FIXED OR RETRACTABLE LANDING GEAR?

A landing gear generates quite some drag when it is deployed. To reduce the drag of the landing gear, you can decide to make it retractable or add a low-drag wheel fairing. In Example 7.7, we showed that the zero-lift drag coefficient and the resulting drag polar depend on this decision. When the landing gear is not retractable, it affects the airplane's performance and, therefore, also the maximum take-off mass, the take-off power, and the wing size. So, how do you decide between a fixed or a retractable landing gear?

A fixed landing gear (e.g., Figure 3.4) is relatively simple and can consist of very few components. It is, therefore, lighter, cheaper to produce, and easier to maintain. Furthermore, it is reliable: it is always deployed. The downside is the higher drag that was mentioned above. A retractable gear (Figure 4.25) can substantially reduce the drag at the cost of increased complexity, cost, weight, and maintenance. It also requires a (large) volume inside the airplane to be stowed, which complicates the integration of all the subsystems on the airframe. Apart from the kinematic mechanism required to safely deploy the landing gear, this is often combined with doors that open and close, which further increases the complexity. Finally, it is inherently less reliable than the fixed landing gear.

Whether you choose a fixed or retractable landing gear for your airplane depends on the type of airplane you are designing. Using the methods Chapters 5 and 7, you can quantify what the impact is of using a fixed or retractable landing gear. Because these methods are not sensitive to qualitative aspects such as complexity and maintenance, you need to make a (qualitative) trade-off. The question is: are the improvements in the performance metrics (maximum take-off mass, take-off power, wing size) worth the increase in complexity, part count, and maintenance cost?

ASSIGNMENT 4.5

In this assignment, you decide your landing gear layout.

a. What landing-gear layout do you choose for your airplane? Motivate your answer.
b. To what airframe components do you choose to connect your landing gear? Explain your decision.
c. Has the decision of your landing-gear layout changed your decision on the vertical wing position or how to integrate the propulsion system?
d. Is your landing retractable or fixed? Explain your decision.

## 4.4. TAIL CONFIGURATION

The function of the tail is three-fold: to balance, to stabilize, and to control the airplane. In this section, we present the design options for the tail. We will present the conventional tail configurations: low tail, cruciform tail, and T-tail, as well as more unconventional tail configurations like the V-tail and the inverted Y-tail. For each of these configurations, we will present the advantages and disadvantages. We will also present possible synergies between the tail configuration and the wing configuration. Or, oppositely, we will explain which combinations of tail configuration and wing position could lead to problematic interactions between these two lifting surfaces. Finally, we will prompt you to make a decision on the tail configuration for your airplane.

### 4.4.1. COMMON TAIL CONFIGURATIONS

The most common tail configurations comprise a single horizontal tailplane and a single vertical tailplane that are located at the rear of the fuselage. The horizontal tailplane balances the airplane around the center of gravity (trim), provides longitudinal stability, and provides pitch control through the elevator. The vertical tailplane provides balance (e.g., in a one-engine inoperative condition), directional stability, and yaw control by means of the rudder. The low-tail configuration, also known as *conventional tail*, is the most common. In this configuration, the vertical tail and horizontal tail are both connected to the fuselage. It provides the simplest integration option compared to the cruciform tail and the T-tail (see Figure 4.26). The low tail also allows for simple staggering of the horizontal and vertical tailplane on the fuselage, meaning one is positioned ahead of the other. This reduces the aerodynamic interference between the vertical and horizontal tail.

While the low-tail configuration might be the most common, there can be a good reason to raise the horizontal tailplane and mount it on the vertical tailplane. For twin-prop airplanes with wing-mounted tractor propellers, the propeller slipstream can re-


Figure 4.26: Three commonly used tail configurations: low tail (left), cruciform tail (center), and T-tail (right). Photos by Caribb CC BY-NC-ND, dylan3300 CC BY-SA, and Nabil Molinari CC BY-NC-SA, respectively.

duce the stabilizing effectiveness of the horizontal tailplane. Raising the horizontal tail improves the stabilizing effectiveness of the tailplane. Jet-powered airplanes with engines on the fuselage need to raise the horizontal tailplane to get it out of the hot exhaust plume. Also, the tail cone of the fuselage could be occupied by other systems, such as an air brake system or an auxiliary power unit that complicates the integration of the horizontal tailplane with the fuselage.

If a low tail is not possible, you can either select a T-tail or a cruciform tail. A cruciform tail has the advantage that it is still relatively close to the fuselage. This implies that only part of the vertical tail structure needs to be reinforced in order to transfer the loads of the horizontal tailplane to the fuselage. This also allows the vertical tailplane to have a lower *taper ratio*[5], which reduces its weight.

The T-tail configuration ensures the largest vertical separation between the wing and the horizontal tailplane. This maximizes the stabilizing effectiveness of the horizontal tailplane. When positioned on a vertical tailplane with a sweep-back angle, the arm between the horizontal tailplane and the airplane's center of gravity is maximized. This implies that the horizontal tailplane can be smaller, which reduces drag. The horizontal tailplane also acts as an end plate to the vertical tailplane, improving the effectiveness of the vertical tailplane as a stabilizing surface.

On the other hand, the T-tail requires a strong vertical tail to transfer the loads to the fuselage. A low aspect ratio helps to reduce the weight of the vertical tailplane but also reduces its effectiveness as a stabilizing surface. The taper ratio of the vertical tailplane is close to one in order to provide sufficient volume at the tip of the vertical tailplane to mount the horizontal tailplane.

T-tails are also notorious because of the so-called *deep stall* that they can cause. A deep stall is an unstable stall where the airplane keeps on increasing its angle of attack without the pilot being able to correct for this by means of the elevator. Such a situation can occur when the wake of a stalled wing immerses the horizontal tailplane (Figure 4.27). The low dynamic pressure in the wake results in a reduction in the aerodynamic force that can be generated by the horizontal tailplane and the elevator. To avoid this, many airplanes use a system that pushes the nose down before the deep-stall angle of attack is reached. Such a system might be mechanical (i.e., a *stick pusher*) or embedded in software as part of the *flight envelope protection* system.

[5] The taper ratio of a lifting surface is the ratio of tip chord to root chord

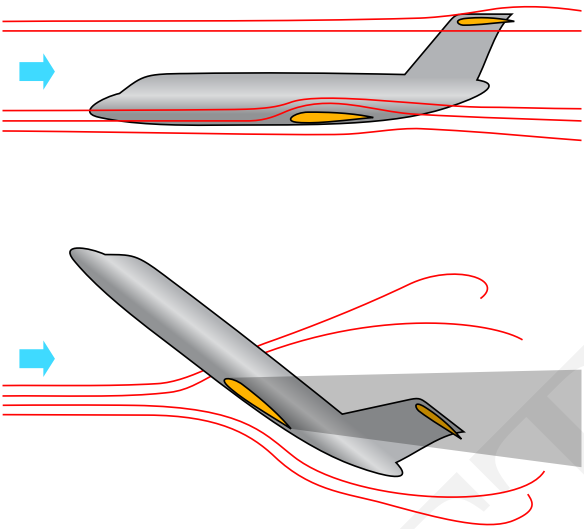

Figure 4.27: When stalling, the wing wake immerses the horizontal tailplane causing a deep stall. Image by GrahamUK, Mysid, and Michael32710 

### 4.4.2. OTHER TAIL CONFIGURATIONS

While the conventional tail, cruciform tail, and T-tail are most common, there are also alternatives that might be a better solution for a given set of top-level requirements. First, let us consider the possibility of having multiple vertical tail surfaces. Examples of twin vertical tails and triple vertical tails are shown in Figure 4.28. Distributing the vertical tails adds redundancy to the directional stability and control system, which is important for military vehicles. When positioned at the end of the horizontal tailplane, the effectiveness of the horizontal tail as a stabilizing surface increases due to the end-plating effect. As the vertical tail forms the highest point on the airplane, distributing the vertical tail area over multiple tails can ensure the airplane can meet a maximum height requirement. Such a requirement could stem from the height of available hangars for the airplane. A final motivation to choose multiple vertical tail surfaces is the aerodynamic interaction with the wing or fuselage at high angles of attack. Depending on the stall characteristics of the airplane, a twin fin might be more effective than a single fin in such a situation.

The downside of having multiple tails is that the number of individual parts increases. A higher part count typically implies higher manufacturing costs. If vertical tails are close together, they also influence each other aerodynamically. This means that two small tails are less effective than a single large tail with the combined size of the two small tails. The further you position your tails apart, the lower this interference effect becomes. Placing the vertical tails at the tips of the horizontal tail therefore minimizes the interference between the two vertical tailplanes. Also in supersonic conditions, the interference between the two tailplanes is reduced due to the formation of oblique shock waves from the leading edge of each tailplane.


Figure 4.28: Examples of twin vertical tails (left and right) and a triple tail (center). Photos by Robert Sullivan, Jez, and pqgw, respectively.

In Figure 4.29 you can see a V-tail, an inverted Y-tail, and a twin-boom tail. The V-tail provides stability and control in pitch and yaw through two tailplane surfaces. This reduces the required total tail area and thereby reduces the (friction) drag of the tail. As the control surfaces on the V-tail are used to control pitch and yaw, they are typically revered as *ruddervators*. When deployed to yaw the airplane to the left, the port-side ruddervator deflects down, while the starboard-side ruddervator deflects up. This creates a yawing moment to the left but also a rolling moment to the right! This effect is not present on the inverted Y-tail, where a yawing moment to the left is accompanied by a rolling moment to the left. The downside of the inverted Y tail is its close proximity to the ground. This reduces the rotation angle that the airplane can make on the ground. The inverted Y-tail in Figure 4.29 also protects the pusher propeller from striking the ground during take-off and landing maneuvers.


Figure 4.29: Examples of less common tail configurations: the V-tail (left), the inverted-Y tail (center) and the twin-boom tail (right). Photos by Huhu Uet, NASA/GA-ASI, and Wally Cacsabre, respectively.

The twin-boom tail configuration shown in Figure 4.29 is actually closely related to the jet engine location of the airplane. In Figure 4.7, we saw another example of the twin-boom configuration for a twin-prop airplane with a pusher propeller at the back of the fuselage. The engine location prevents a single tail boom extending from the centerline of the airplane. Therefore, two tail booms are attached to the wings, each having a vertical tailplane at the end. The T-tail configuration of the twin-boom configuration in Figure 4.29 prevents the hot exhaust of the jet engine from immersing the horizontal tailplane.

### 4.4.3. TAILLESS AIRPLANES

Airplanes that lack a horizontal and/or vertical tailplane have been flying since the 1920s. The absence of tail surfaces reduces the weight and drag of the airplane and contributes to a higher overall efficiency. However, the wing needs to perform the functions of directional and longitudinal stability and control. In Figure 4.30, we show three examples of tailless airplanes. Each of these airplanes still features vertical tail surfaces to provide directional stability and control. Roll and pitch control is provided by means of *elevons*, , which combine the function of elevator and aileron. The swept-back wings ensure sufficient longitudinal stability and increase the pitching moment due to symmetric elevon deflection. On the flying-wing airplane and the flying-V airplane, the vertical tails double as winglets. This reduces the lift-induced drag resulting in a higher Oswald factor ($e$). Also, per unit of useful volume, each of the tailless airplanes of Figure 4.30 have less wetted area ($S_{\text{wet}}$), which reduces their zero-lift drag. Therefore, tailless configurations like the blended-wing-body and the flying-V airplane are efficient for carrying large volumes of passengers, cargo, and fuel. For hydrogen-fueled airplanes, which require at least four times the volume for the same amount of energy, these configurations could be advantageous over the traditional distinct wing-fuselage configuration.

Figure 4.30: Examples of tailless airplanes: a flying-wing airplane (left), a blended-wing-body airplane (center) and a flying-V airplane (right). Photos by Harveyellis (cc)(by)(sa), S. Ramadier for Airbus, and J. van Oppen for TU Delft, respectively.

Tailless airplanes have many challenges when it comes to integrating all the functionalities of a wing and a horizontal tailplane into a single wing. The generation of lift, providing trim, providing pitch control, and providing longitudinal stability needs to be performed with the same lifting surface. This requires compromises in the design of the wing. As the stability and control functionalities cannot be directly mapped onto the design of distinct tail surfaces, the design process is also more complicated. Therefore, this textbook does not provide methods to design tailless airplanes. The examples of Figure 4.30 show that while the design process is more complicated, tailless airplanes can be successfully flown in a stable and controlled manner.

**ASSIGNMENT 4.6**

In this assignment, you will decide on your airplane's tail configuration.

a. Given the requirements of your airplane and the possible tail configurations laid out in this section, what tail configuration do you choose for your airplane? Motivate your answer.
b. Does the tail configuration of your choice have an influence on the other design choices you have made regarding the configuration? In other words, would you change any of the design decisions you have made before, now that you have chosen your tail configuration? Motivate your answer.

# 5

# PRELIMINARY MASS ESTIMATION

Now that the design process has been defined (Figure 2.1) and the set of top-level requirements has been compiled, it is time to take our first steps on the design ladder. These first two steps are collectively termed "preliminary sizing," i.e. using a subset of our top-level requirements to determine the size of three key aspects of the airplane we are designing: the airplane's maximum take-off mass, the wing area, and the thrust (or power) of the powerplant. However, to perform the sizing activities, we need to know how the lift and drag of the airplane are correlated. In other words, we need a *drag polar*, a mathematical expression that expresses the drag coefficient of an airplane as a function of its lift coefficient.

Here, we arrive at our first chicken-and-egg problem: how to derive a drag polar from an airplane for which we do not know anything yet? Fig. 2.1 already gives us a hint: we need to make assumptions. Assumptions that we replace by analysis results later on in the design process. Not only do we need assumptions for the drag polar, but we also need assumptions for the characteristics of the powerplant. As the outcome of the preliminary sizing process depends on the mathematical expression of the drag polar, it is inherently correlated to the assumptions that we make. Badly chosen assumptions might result in an airplane design with either too favorable or too pessimistic performance. So, how do you make sure your assumptions are good when you embark on the sizing process?

For that, we turn to our reference airplanes that we have defined in Section 3.4. We will use both the qualitative and quantitative information that we have gathered in Assignment 3.9. The fact that we use information from the past to start the design process, does not mean that the resulting design will be simply an average of our reference airplanes. On the contrary, we only use it to get the process started. Just think of it as an "egg" that we need to produce our first chicken. Once we have our first chicken, we can ask it to lay its own eggs, i.e., replace the initial assumptions with analysis results. The closer the initial assumptions are to the analysis results, the lower the number of design iterations that are needed for the design process to converge. This is yet another reason to choose your reference airplanes carefully.

In this chapter, we explain how you can estimate the mass properties of the airplane.

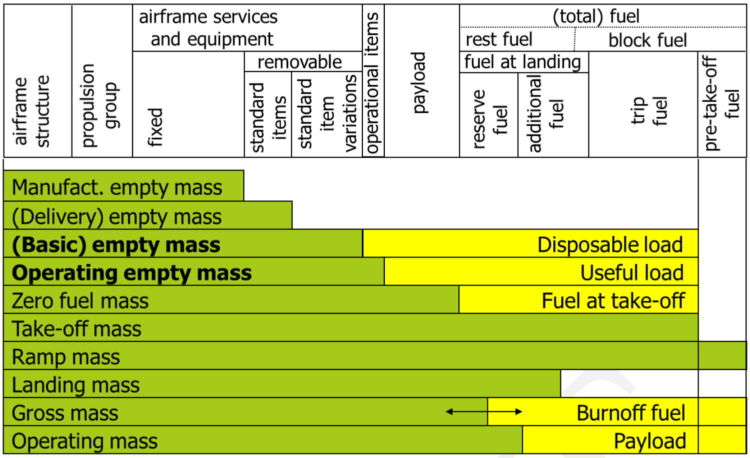


Figure 5.1: Graphical interpretation of the various mass definitions for an airplane. Note that typically the word "weight" is used in the open literature. After: Torenbeek [20]

First, we show what the various mass definitions mean. We subsequently show how, based on the top-level airplane requirements, you can estimate the maximum take-off mass, the fuel mass, and the operating empty mass. Finally, we demonstrate how to construct the payload-range diagram.

## 5.1. MASS ESTIMATION METHOD: THE UNITY EQUATION

Before we present our mass estimation method, let us start with various mass definitions. Figure 5.1 graphically shows these mass definitions for a fuel-consuming airplane. There are three mass groups that are important for this chapter: the (operating) empty mass ($m_{\mathrm{OE}} + m_{\mathrm{ec}}$), the payload mass ($m_{\mathrm{pl}}$), and the fuel mass. The (basic) empty mass of an airplane comprises the mass of the structure (i.e. the airframe), the propulsion group, and the fixed *and* removable equipment. The *operating empty* mass also includes the mass of all the items and people that are needed to operate the airplane: cabin crew, cockpit crew, drinks, meals, potable water, non-potable water, operating manuals, etc. The payload mass ($m_{\mathrm{pl}}$) comprises the mass of passengers, luggage, and cargo. The total energy mass ($m_{\mathrm{ec}}$) encompasses the energy required to fly the mission as well as the reserve energy.

The maximum take-off mass of the airplane ($m_{\mathrm{MTO}}$) is the sum of these three mass components, i.e.:

$$m_{\mathrm{MTO}} = m_{\mathrm{pl}} + m_{\mathrm{OE}} + m_{\mathrm{ec}} \tag{5.1}$$

We intentionally have the letter "O" in $m_{\mathrm{OE}}$ in grey because for small airplanes, we typically use the empty mass, while for larger airplanes, we use the operating empty mass in this equation, as will be explained below. Energy can be stored as fuel in a tank or as electricity in a battery. By dividing the left-hand side and right-hand side by $m_{\mathrm{MTO}}$, we find the so-called *unity equation*:

$$\frac{m_{\text{pl}}}{m_{\text{MTO}}} + \frac{m_{\text{OE}}}{m_{\text{MTO}}} + \frac{m_{\text{ec}}}{m_{\text{MTO}}} = 1 \tag{5.2}$$

As $m_{\text{pl}}$ is known from the top-level airplane requirements, we need to make an estimation for $(m_{\text{OE}}/m_{\text{MTO}})$ and $(m_{\text{ec}}/m_{\text{MTO}})$ in order to solve (5.2).

## 5.2. ESTIMATING THE ENERGY-MASS FRACTION

To estimate $(m_{\text{ec}}/m_{\text{MTO}})$ in (5.2), we have to distinguish two scenarios. In the first scenario, the stored energy comes in the form of fuel and is burned during flight. In this case, the airplane becomes lighter during the course of the flight. In the second scenario, the energy is consumed, but the mass of the energy carrier remains the same. In that case, the airplane has the same mass throughout the entire flight. If we consider an airplane that only cruises (i.e. no take-off, climb, etc.), then the following Breguet range equations can be derived [18, 25]:

$$R = \eta_{\text{eng}}\eta_{\text{p}}\left(\frac{L}{D}\right)\left(\frac{e_{\text{f}}}{g}\right)\ln\left(\frac{m_{\text{OE}}+m_{\text{pl}}+m_{\text{f}}}{m_{\text{OE}}+m_{\text{pl}}}\right) \qquad \text{for } m_{\text{ec}} = m_{\text{f}} \tag{5.3}$$

$$R = \eta_{\text{em}}\eta_{\text{p}}\left(\frac{L}{D}\right)\left(\frac{e_{\text{bat}}}{g}\right)\left(\frac{m_{\text{bat}}}{m_{\text{OE}}+m_{\text{pl}}+m_{\text{bat}}}\right) \qquad \text{for } m_{\text{ec}} = m_{\text{bat}} \tag{5.4}$$

where $\eta_{\text{em}}$ and $\eta_{\text{eng}}$ are the electric motor and engine efficiency, respectively, $\eta_p$ is the propulsive efficiency, $L$ and $D$ are the lift and drag force, respectively, $e_{\text{f}}$ and $e_{\text{bat}}$ are the specific energy of fuel and battery, respectively, and $g$ is the gravitational acceleration.

In either equation, the aerodynamic efficiency $(L/D)$ is linearly related to the range as well as to the propulsive efficiency $(\eta_p)$ of the powerplant. Note that within the mass fraction of (5.3), the maximum take-off mass is in the numerator, while it is in the denominator of the mass fraction of (5.4). For the fuel-burning airplane, the engine $(\eta_{\text{eng}})$ in (5.3) is a thermodynamic efficiency for converting fuel into mechanical power. The electric motor efficiency $(\eta_{\text{em}})$ in (5.4) is an electric efficiency of converting electric current into mechanical power. The *mass-specific energy* or simply *specific energy* of fuel is denoted with $e_{\text{f}}$ in (5.3), which is a property of the type of fuel that you have selected (Assignment 4.1). For batteries, the specific energy $(e_{\text{bat}})$ is a function of the battery chemistry that you select as well as the packaging of the battery pack. In the context of this textbook, $e_{\text{bat}}$ refers to the specific energy of the battery pack rather than the individual cells that are in the battery pack. Finally, $g$ is the gravitational acceleration (i.e. $9.81\ \text{m/s}^2$). You can use Equation 5.3 when your selected energy carrier is a fuel, while (5.4) should be used when you selected batteries as the energy carrier for your airplane.

The Breguet equations relate the cruise range $R$ to the energy consumption. However, we can imagine that we need even more energy to account for acceleration, climb, diversion, and contingency. Therefore, we define an *equivalent range*, $R_{\text{eq}}$ that is the cruise range that could be flown on the total energy that is on board the airplane, i.e. without acceleration, climb, diversion, and contingency (see also Section 5.2.3). We use the equivalent range in (5.1) and (5.6) to compute the energy-mass fraction $(m_{\text{ec}}/m_{\text{MTO}})$

in (5.2). We rewrite (5.3) and (5.4) and substitute $R = R_{\mathrm{eq}}$:

$$\frac{m_{\mathrm{f}}}{m_{\mathrm{MTO}}} = 1 - \exp\left[\frac{-R_{\mathrm{eq}}}{\eta_{\mathrm{eng}}\eta_{\mathrm{p}}(e_{\mathrm{f}}/g)(L/D)}\right] \tag{5.5}$$

$$\frac{m_{\mathrm{bat}}}{m_{\mathrm{MTO}}} = \frac{R_{\mathrm{eq}}}{\eta_{\mathrm{EM}}\eta_{\mathrm{p}}(e_{\mathrm{bat}}/g)(L/D)} \tag{5.6}$$

Let us examine the ingredients of these equations and how we can compute them. The energy carrier of your airplane is a design choice, and you can look up the specific energy of the energy carrier of your choice (see Table 4.1). The lift-to-drag ratio ($L/D$) is a property of the airplane. In Section 5.2.1, we show how to estimate the lift-to-drag ratio in cruise conditions. The engine efficiency ($\eta_{\mathrm{eng}}$) or motor efficiency ($\eta_{\mathrm{EM}}$) are a function of the propulsion system. In Section 5.2.2, we show how to estimate these properties along with the propulsive efficiency ($\eta_{\mathrm{p}}$). Finally, in Section 5.2.3, we show how you can estimate an equivalent range ($R_{\mathrm{eq}}$). Note that if you have selected a cryogenic fuel for your energy carrier, the *effective* mass-specific energy should be used in Equation 5.5, which can be computed with Equation (4.3). In that case, $m_{\mathrm{f}}$ should be replaced with $m_{\mathrm{f}}/\eta_{\mathrm{f}}$ on the RHS of Equation 5.5 and in all the equations showing $m_{\mathrm{f}}$ downstream from here.

### 5.2.1. ESTIMATING THE LIFT-TO-DRAG RATIO

In this section, we show how you can estimate the aerodynamic efficiency, also known as the *lift-to-drag ratio*. Therefore, we need a mathematical expression that relates the drag coefficient to the lift coefficient of the airplane. Remember that the lift and drag coefficients are defined as follows:

$$C_L = \frac{2L}{\rho V^2 S_{\mathrm{w}}} \tag{5.7}$$

$$C_D = \frac{2D}{\rho V^2 S_{\mathrm{w}}} \tag{5.8}$$

where $\rho$ is the density of air, $V$ is the velocity and $S_{\mathrm{w}}$ is the wing reference area. The reference area is typically based on the planform area of the wing. Without derivation, we present that for subsonic, fixed-wing airplanes up to moderate angles of attack, the drag coefficient of the airplane can be related to the lift coefficient according to:

$$C_D = C_{D_0} + C_{D_{\mathrm{i}}} = C_{D_0} + \frac{C_L^2}{\pi A\!\!R_{\mathrm{w}} e} \tag{5.9}$$

where $C_{D_0}$ is the zero-lift drag coefficient and $C_{D_{\mathrm{i}}}$ the induced-drag coefficient. The latter is a function of the lift coefficient, the aspect ratio ($A\!\!R$), and the *Oswald factor e*. The aspect ratio is a non-dimensional parameter that describes the wing slenderness. It is defined as follows:

$$A\!\!R_{\mathrm{w}} = \frac{b_{\mathrm{w}}^2}{S_{\mathrm{w}}} \tag{5.10}$$

Equation 5.9 is the *drag polar*, a two-term, second-order equation in $C_L$. As you can see, the equation describes a parabola with a minimum at $C_L = 0$. The value of the drag coefficient at an arbitrary lift coefficient depends on two things: the value of $C_{D_0}$ and the value of $1/(\pi AR_\mathrm{w} e)$. Figure 5.2 shows how we define these reference dimensions. Note that part of the wing reference area is located *inside* the fuselage.

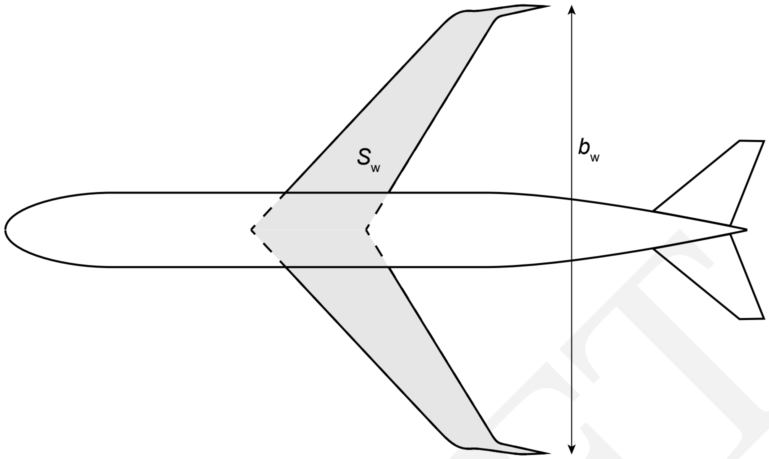


Figure 5.2: Definition of the reference wing area ($S_\mathrm{W}$) and the reference span ($b_\mathrm{W}$) for a generic airplane

Now that we have established a relationship between the lift and drag coefficient, we can estimate the maximum lift-to-drag ratio, which is equivalent to the minimum drag-to-lift ratio. We derive the following:

$$\frac{\partial}{\partial C_L}\left(\frac{C_D}{C_L}\right) = 0 \quad \rightarrow \quad C_D = 2C_{D_0} \text{ and } C_L = \sqrt{\pi AR_\mathrm{w} e C_{D_0}} \tag{5.11}$$

Substituting these values into the expression for lift-to-drag ratio results in a maximum lift-to-drag ratio of:

$$\left(\frac{L}{D}\right)_\mathrm{max} = \frac{1}{2}\sqrt{\frac{\pi AR_\mathrm{w} e}{C_{D_0}}} \tag{5.12}$$

This is the lift-to-drag ratio that we substitute in (5.5) and (5.6) when we compute the fuel mass fraction or battery mass fraction, respectively. There are three ingredients that we need to estimate in order to quantify the maximum lift-to-drag ratio: $AR_\mathrm{w}$, $C_{D_0}$, and $e$. We will subsequently discuss how to determine each of them.

**Example 5.1**

Assume that we are designing a jet transport airplane and that we have estimated a zero-lift drag coefficient and Oswald factor in cruise configuration of $C_{D_0} = 0.0180$ and $e = 0.80$, respectively. Compute the maximum lift-to-drag ratio.

**solution** We employ (5.12) to compute the lift-to-drag ratio. We choose an aspect ratio of $AR_\mathrm{w} = 8$. Then we have:

$$\left(\frac{L}{D}\right)_\mathrm{max} = \frac{1}{2}\sqrt{\frac{\pi \cdot 8 \cdot 0.80}{0.0180}} = 16.7$$

### CHOOSING THE ASPECT RATIO

The aspect ratio ($AR_{\mathrm{w}}$) is a design parameter that you have to choose. Noting that $AR_{\mathrm{w}}$ is in the numerator of (5.12), it might seem convenient to choose $AR_{\mathrm{w}}$ very high. This would result in very low lift-induced drag and, consequently, a high lift-to-drag ratio. However, such a slender wing would also be very thin and wide. Therefore, it would need a very heavy structure inside to carry the weight of the fuselage and its payload. In general, we can say that the lower the aspect ratio, the lighter the wing. While we cannot quantify the effect of your decision at this stage, it is something to keep in the back of your mind when making a choice for the aspect ratio. Naturally, you can look at your reference airplanes and chart their wing aspect ratios. This could give you some indication of what is practical. However, it is up to you to choose the value of your aspect ratio.

**ASSIGNMENT 5.1**

1. For each of your reference airplanes, compute the reference aspect ratio, $AR_{\mathrm{w}}$.
2. Decide which aspect ratio you choose for your airplane. Motivate your design decision by including the following aspects:
   - How does the choice in aspect ratio compare to the ones found on your reference airplanes?
   - What is the envisioned effect of your choice on the lift-induced drag?
   - What is the envisioned effect of your choice on the wing mass?
   - What is the envisioned effect of your choice on internal wing volume for fuel and systems?

### ESTIMATING THE ZERO-LIFT DRAG COEFFICIENT

When the aspect ratio has been decided, only two quantities are left in Eq. 5.9: $C_{D_0}$ and $e$. For both of these values, we need to make an assumption. We know that for a well-designed subsonic airplane, the zero-lift drag consists mostly of skin-friction drag plus a small fraction of pressure drag. The non-dimensional form of the local skin-friction stress ($\tau$) is the skin-friction coefficient $c_f$:

$$c_f = \frac{\tau}{\frac{1}{2}\rho V^2} \tag{5.13}$$

The local skin friction coefficient varies over the airplane as does the local pressure drag at zero lift. Therefore, we introduce the concept of an *equivalent skin-friction coefficient*, $\bar{C}_f$: the average friction coefficient over all of the *wetted* surface area of the airplane. The wetted area ($S_{\mathrm{wet}}$) is defined as all of the airplane's surface area that is touched by the airflow during flight. The zero-lift drag coefficient and the equivalent friction coefficient are related as follows:

$$\bar{C}_f S_{\mathrm{wet}} = C_{D_0} S_{\mathrm{w}} \tag{5.14}$$

The product of $C_{D_0}$ and $S_w$ is referred to as the *parasite drag area.* We can rearrange this equation to have an estimate for the zero-lift drag coefficient as a function of the equivalent friction coefficient and the ratio between the wetted area and reference area:

$$C_{D_0} = \bar{C}_f \frac{S_{wet}}{S_w} \tag{5.15}$$

In order to estimate the zero-lift drag coefficient, we "only" need to find an estimate for $\bar{C}_f$ and $(S_{wet}/S_w)$.

The ratio between the wetted area and reference area is a function of the type of airplane that you are designing. You can imagine that if you design a pure flying-wing airplane, this ratio is a bit larger than 2. However, if you design a classical airplane with a fuselage and a tail, the ratio increases quite rapidly. Figure 5.3 shows examples of airplanes and where they fall on the $(S_{wet}/S_w)$-chart. You can see that commercial airplanes have a ratio of around 6, a twin-prop commuter is around 5, and a small single-engine airplane sits around 4. Based on this very limited information, you can make an assumption of $(S_{wet}/S_w)$ for your airplane. Keep in mind that this assumption will be replaced by analysis once we have finished the first iteration of the airplane geometry (see Chapter 10).

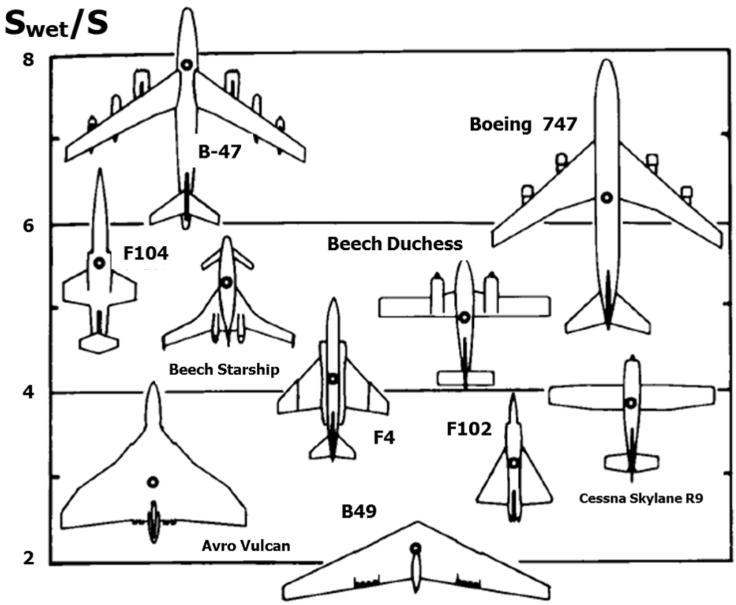


Figure 5.3: The ratio between wetted area and reference area $(S_{wet}/S_w)$ depends on the type of airplane you design. Adapted from [12].

The equivalent friction coefficient is dependent on the size of the airplane, the roughness of the skin, and the shape of the components. In addition, manufacturing imperfections, rivet heads, gaps, air inlets and exhausts, steps in the exterior geometry, and external protrusions (like antennas) also contribute to the equivalent friction coefficient.

None of that information is known at this stage, but we can still make an estimate of the equivalent friction coefficient based on the estimated wetted area of the airplane. In assignment 5.2, you can see which steps to take in order to do this. One key aspect is the relation between size and equivalent friction coefficient. For a given production technology, it can be generalized that the larger the airplane, the smaller its equivalent friction coefficient.

Figure 5.4 shows the friction-drag coefficient as a function of wetted area for a large number of subsonic business jets and jet transports. The trend line that is given shows that there is a nonlinear relation between the equivalent friction coefficient and the wetted area. It shows that for very small airplanes, the equivalent friction drag coefficient can be as high as 0.0045, while many single-aisle commercial transports are around 0.0030. The asymptotic behavior suggested by the trend line shows that larger airplanes have a lower average friction coefficient. However, you should keep in mind that this trend only holds as long as the surface finish is the same [8].

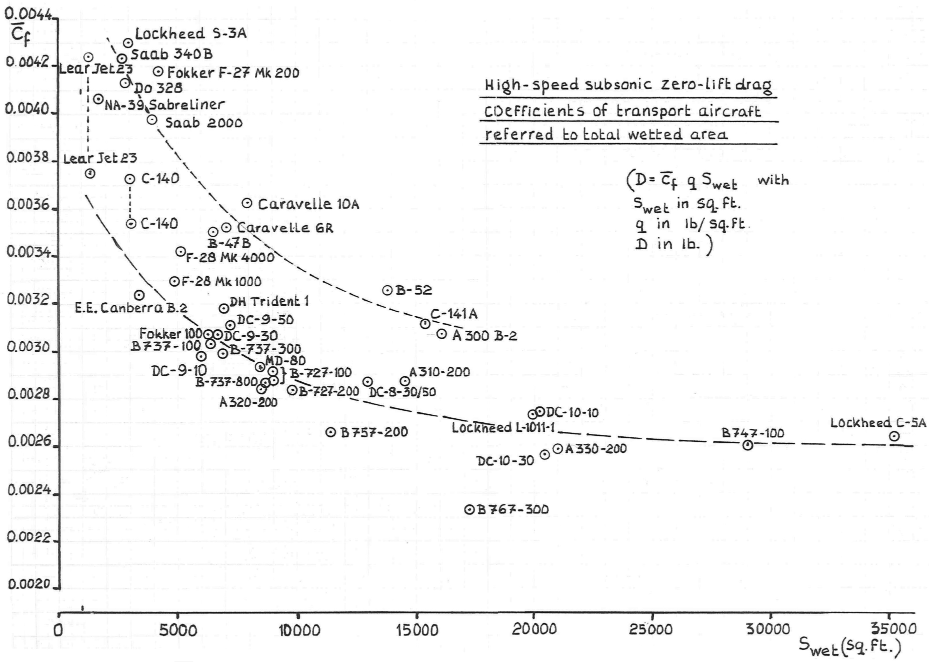


Figure 5.4: Equivalent friction coefficient versus wetted area for turbine-powered airplanes. Data from [8].

In summary, we propose the following sequence to estimate the zero-lift drag coefficient:

*Step 1* Make assumption for the wetted-area to wing-area ration, $S_{\text{wet}}/W_{\text{w}}$.

*Step 2* Make an approximation of the wetted area of your reference airplanes.

*Step 3* Estimate the wetted area of your airplane.

*Step 4* Estimate the equivalent friction-drag coefficient.

*Step 5* Compute the zero-lift drag coefficient using (5.15).

The following example shows how this is done for a turbofan-powered commercial airplane.

**Example 5.2**

In this example, we show the process of estimating the zero-lift coefficient of a commercial turbofan-powered airplane in cruise configuration.

*Step 1* We have a commercial airplane, so looking at Figure 5.3, we estimate $S_{\mathrm{wet}}/S_{\mathrm{w}}$ to be close to the value of the Boeing 747. We assume $S_{\mathrm{wet}}/S_{\mathrm{w}} = 6.0$.

*Step 2* Using our reference airplanes, we estimate their respective wetted areas assuming $S_{\mathrm{wet}}/S_{\mathrm{w}} = 6.0$:

| Name | A320NEO | B737-8 | C919ER | MC-21-300 | MD-90-30 |
|---|---|---|---|---|---|
| $S_{\mathrm{w}}$ (m$^2$) | 123 | 127 | 129 | 120 | 112 |
| $S_{\mathrm{wet}}$ (m$^2$) | $\approx 740$ | $\approx 760$ | $\approx 770$ | $\approx 720$ | $\approx 670$ |

*Step 3* We estimate the wetted area of our airplane by taking the average of our reference airplanes, i.e., $S_{\mathrm{wet}} = 730\ \mathrm{m}^2 \approx 7900\ \mathrm{ft}^2$.

*Step 4* Using the lower graph from Figure 5.4, we estimate $\bar{C}_f = 0.0030$.

*Step 5* With (5.15) we compute $C_{D_0} = 0.0180$.

ASSIGNMENT 5.2

In this assignment, you are going to estimate the zero-lift drag coefficient of your airplane. To complete this assignment, you need data from Assignment 3.9.

a. Make an estimate of the ratio of wetted-area to reference area, i.e. ($S_{\mathrm{wet}}/S_{\mathrm{w}}$) based on Fig. 5.3, your reference airplanes, and your own notion of what the airplane will look like.
b. Using the value of ($S_{\mathrm{wet}}/S_{\mathrm{w}}$) from a. and the wing areas of your reference airplanes, compute for each reference airplane the approximate wetted area.
c. Estimate the wetted area, $S_{\mathrm{wet}}$, of your airplane.
d. Using Figure 5.4 and the estimated wetted areas of your reference airplanes, estimate the equivalent friction coefficient for your airplane.
e. Using (5.15), estimate the zero-lift drag coefficient of your airplane.

### ESTIMATING THE OSWALD FACTOR

Using the steps of Assignment 5.2, you can now make an estimation of the first unknown parameter of the drag polar. The second parameter is known as the Oswald efficiency factor, $e$, and is a correction factor that represents the change in lift-induced drag of an airplane compared to an ideal wing of minimum lift-induced drag having the same aspect ratio as the airplane's wing. The Oswald efficiency factor pertains to a complete airplane and is therefore different from the *span efficiency factor*, which pertains only to a wing. While the Oswald factor is related to the span efficiency factor, they are not the same. Apart from the spanwise lift distribution, the Oswald factor depends on the orientation of the fuselage, the flap setting, as well as the presence of engine nacelles. For an airplane without flap deflection, the lift-induced drag can therefore be estimated as follows:

$$C_{D_\mathrm{i}} = \left(\psi + \frac{1}{\pi \textit{Æ}_\mathrm{w} \varphi}\right) C_L^2 = \frac{1}{\pi \textit{Æ}_\mathrm{w} e} C_L^2 \tag{5.16}$$

where $\psi$ is the parasite drag dependent on the lift coefficient and $\varphi$ is the span efficiency factor of the wing. Figure 5.5 shows how $\frac{1}{\pi \textit{Æ}_\mathrm{w} e}$ changes as a function of aspect ratio for large transport airplanes. The trend line that is shown assumes a span efficiency of $\varphi = 0.97$ and a lift-dependent parasite drag parameter of $\psi = 0.0075$. The corresponding Oswald factor can then be computed according to:

$$e = \frac{1}{\pi \textit{Æ}_\mathrm{w} \psi + \frac{1}{\varphi}} \tag{5.17}$$

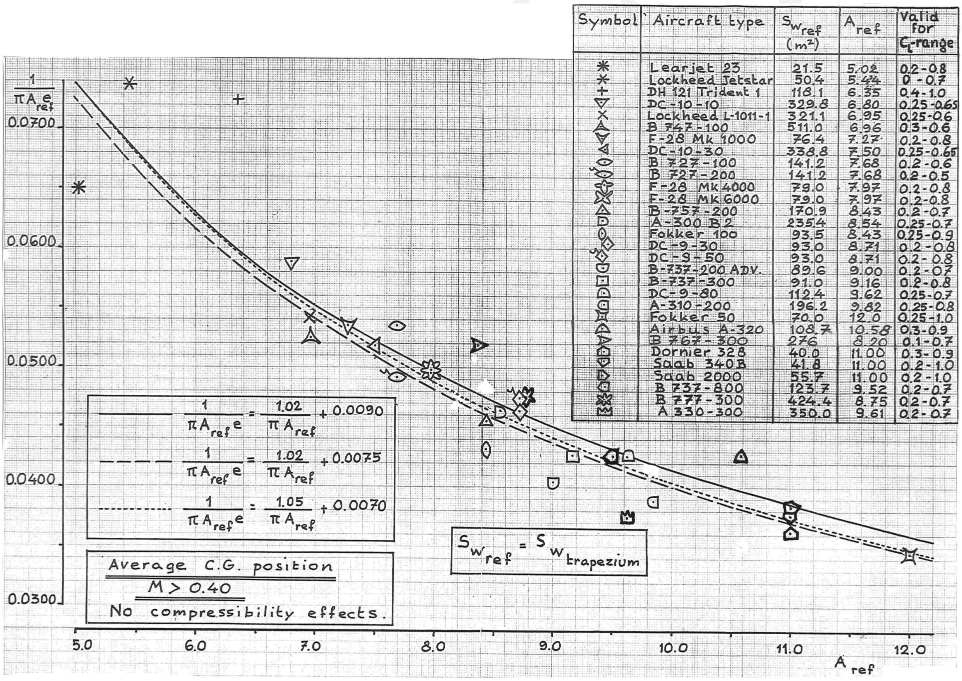


Figure 5.5: The Oswald factor, $e$, as a function of aspect ratio, $\textit{Æ}_\mathrm{w}$, for a variety of transport airplanes. Data from [8].

In the following example, we show how to estimate the Oswald factor of your airplane.

**Example 5.3**

In this example, we estimate the Oswald factor for a battery-powered 4-seater airplane.

*Step 1* For the parasite drag parameter, we assume $\psi = 0.0075$, similar the value used for the dashed line in Figure 5.5.

*Step 2* As we do not yet know the aerodynamic properties of our wing, we assume $\phi = 0.83$.

*Step 3* Using (5.17), we compute $e = 0.71$.

---

ASSIGNMENT 5.3

In this assignment, you are going to estimate the Oswald efficiency of your airplane.

a. What value for the parasite drag parameter, $\psi$, do you assume?
b. What value of the span efficiency factor, $\varphi$, do you assume?
c. Based on these assumptions, what is the Oswald efficiency factor, $e$ of your airplane?

### 5.2.2. ESTIMATING THE PROPULSION SYSTEM EFFICIENCIES

In this section, we show how you can estimate the propulsion system efficiencies, $\eta_{\mathrm{EM}}$, $\eta_{\mathrm{f}}$, and $\eta_{\mathrm{p}}$. We distinguish between electric motors and fuel burning engines but also between propeller propulsion and jet propulsion.

For electric motors, $\eta_{\mathrm{EM}}$ can often be found in online databases for electric motors. Typically, electric motors have a very high efficiency (i.e. 95%). However, extracting electricity from a battery and transporting it through cables also leads to losses. If cables and batteries are properly designed, this discharge efficiency can be close to 100%. In this textbook, we neglect transmission efficiency and only consider electric motor efficiency.

For fuel-burning engines, often the thrust-specific fuel consumption (TSFC) or power-specific fuel consumption (PSFC) can be found in the open literature. For piston engines and turboprop engines, the PSFC is related to the (thermal) efficiency of the engine, $\eta_{\mathrm{eng}}$. For a given engine, the following relations can be used in (5.3):

$$e_{\mathrm{f}}\eta_{\mathrm{eng}} = \frac{1}{\mathrm{PSFC}} \tag{5.18}$$

where $e_{\mathrm{f}}$ is the specific energy (i.e. the energy per unit mass) of the fuel that is burned in the engine. The power-specific fuel consumption for propeller engines is often expressed in lb/hp/hr. To use these published values in the context of the methods presented above, these units need to be converted to SI units. In the following example, we show how the efficiency of a reciprocating engine can be computed.

**Example 5.4**

The Lycoming O-320 (Fig. 5.7) is a piston engine with a maximum power of 160hp (120 kW) that burns avgas (aviation gasoline). The Lycoming O-320 power-specif fuel consumption is approximately 0.48 lb/hp/hr. Converting that to SI units, this results in PSFC = 0.48 [lb/hp/hr] $\cdot \frac{0.45\ [\text{kg/lb}]}{745\ [\text{W/hp}]\cdot 3600\ [\text{s/hr}]} = 8.05 \cdot 10^{-8}$ kg/W/s. We can now directly compute ($e_\text{f}\eta_\text{eng}$) using (5.18):

$$e_\text{f}\eta_\text{eng} = \frac{1}{8.05 \cdot 10^{-8}} = 1.24 \cdot 10^{7}\ \text{J/kg}$$

Using a specific energy of 44.7 MJ/kg for avgas, we can also compute the engine efficiency:

$$\eta_\text{eng} = \frac{1.24 \cdot 10^{7}}{4.47 \cdot 10^{7}} = 0.28 \tag{5.19}$$

---

To compute the engine efficiency for a turboprop engine, the same procedure can be used for a piston engine.

Propulsion systems that drive a propeller are typically rated by their mechanical power ($P$). The thrust ($T$) is the resultant force that is generated by a propeller. To maintain a certain level of thrust at a given speed ($V$), the propeller is spun by an engine or motor. Therefore, part of the shaft power is required to overcome the aerodynamic torque produced by the propeller. We therefore use the propulsive efficiency of the propeller ($\eta_\text{p}$) to correlate the shaft power to the thrust and speed of the airplane as follows:

$$\eta_\text{p} P = TV \tag{5.20}$$

Here, $\eta_\text{p}$ is the *propulsive efficiency* of the propeller. It should be noted that this relation does not work when $V \to 0$ but works well for all flying conditions.

The propulsive efficiency, $\eta_\text{p}$, is a measure for how much of the shaft power is converted to propulsive power. The propulsive efficiency of a propeller depends on the geometry of the propeller: the number of blades, the diameter of the propeller, the chord distribution of each blade, the twist in the blades, and the airfoils that are chosen. Clearly, none of that information is available to us at this stage. The efficiency also depends on the speed regime where the propeller is used, whether it has fixed or variable pitch, and the rotational speed it has. We hypothesize that downstream in the design process, a propeller is designed to maximize propulsive efficiency in cruise conditions. To allow us to move forward in the design process, we, therefore, have to assume a value for the propulsive efficiency at this stage. In other words, we need to make a *reasonable assumption* on the propulsive efficiency of the propeller. Table 5.1 can be consulted to help you make that assumption.

For a jet engine, the thrust-specific fuel consumption is the weight of fuel that is consumed per unit of thrust per unit of time. The TSFC is related to the thermal efficiency and propulsive efficiency of the engine, which are often lumped into a single efficiency value, i.e. $\eta_\text{j} = \eta_\text{eng}\eta_\text{p}$. The published value of TSFC is typically found in the design cruise condition. We can relate $\eta_j$ to the TSFC as follows:

$$e_\text{f}\eta_\text{eng}\eta_\text{p} = e_\text{f}\eta_\text{j} = \frac{V_\text{CR}}{\text{TSFC}} \tag{5.21}$$

Table 5.1: Tabulated values for $\eta_\mathrm{p}$ for various airplane types. Data from Ref. [13].

| airplane type | $\eta_p$ |
|---|---|
| homebuilts | 0.70 |
| single engine/motor props | 0.80 |
| twin engine props | 0.82 |
| regional turboprops | 0.85 |

where $V_\mathrm{CR}$ is the cruise speed. Often imperial units are used to express the TSFC, which results in lb/lb/hr. The following example shows how to compute the engine efficiency of a turbofan engine.

**Example 5.5**

The CFM-56 (Fig. 5.6) is a turbofan engine that burns kerosene and has a thrust of 30,000 lb (132 kN) at sea level. A turbofan engine has a gas turbine at its core that drives a ducted fan. The thrust-specific fuel consumption of a CFM-56 is 0.55 lb/lb/hr.[1] If we convert that to SI units, we have TSFC $= 0.55$ [lb/lb/hr] $\cdot \frac{0.45\ \text{[kg/lb]}}{4.4\ \text{[N/lb]}\cdot 3600\ \text{[s/hr]}} = 1.56\cdot 10^{-5}$ [kg/N/s] $=$ 15.6 [g/kN/s]. We assume that this engine is used on a transport airplane that has a cruise speed of $V_\mathrm{cr} = 230$ m/s. We can now compute the value of $(e_\mathrm{f}\eta_\mathrm{eng}\eta_\mathrm{p})$ according to (5.21):

$$e_\mathrm{f}\eta_\mathrm{eng}\eta_\mathrm{p} = \frac{230}{15.6\cdot 10^{-6}} = 14.7\ \mathrm{MJ/kg}$$

The resulting engine efficiency is then:

$$\eta_\mathrm{j} = \eta_\mathrm{eng}\eta_\mathrm{p} = \frac{14.7}{43} = 0.34$$

where we use a specific energy of 43 MJ/kg for kerosene.

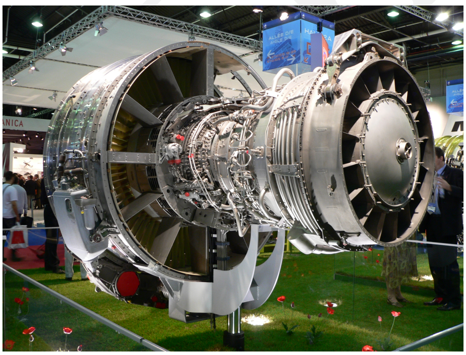

Figure 5.6: The CFM56 is a turbofan engine belonging to Example 5.5. Photo: David Monniaux.

Figure 5.7: The Lycoming O-320 is a piston engine belonging to Example 5.4. Photo: A. Hunt.

[1] Source: wikipedia.org

The TSFC of turbofan engines is dependent on the altitude and Mach number at which the engine is operated. In the context of this book, we only consider the cruise conditions when we consider the TSFC. Naturally, the TSFC of an engine depends on the characteristics of the engine components. In particular, the overall pressure ratio, the total temperature at the entry of the turbine, and the *bypass ratio* ($B$) have an impact on the fuel consumption of the engine. The bypass ratio is the ratio between the mass flow going through the gasturbine core (h, for hot) and the mass flow that bypasses the gasturbine (c for cold) and only flows through the fan:

$$B = \frac{\dot{m}_{\mathrm{c}}}{\dot{m}_{\mathrm{h}}} \tag{5.22}$$

The bypass ratio has a strong impact on the efficiency and size of the engine. For the same thrust, increasing the bypass ratio improves the engine efficiency and increases the engine diameter.

Despite the complex relation between engine design parameters and engine efficiency, the bypass ratio is the only turbofan design variable that we consider in this book. Based on engine data from the open literature, we have deduced the following relation between TSFC in cruise conditions and the bypass ratio, $B$:

$$\mathrm{TSFC} = 22B^{-0.19}\ [\mathrm{g/kN/s}] \quad \text{for} \quad 1 \leq B \leq 15 \tag{5.23}$$

This relation gives you an estimate of the TSFC based on $B$ with an accuracy of $\pm 5\%$, as can be seen in Figure 5.8. You can subsequently use (5.21) to compute the corresponding jet efficiency, $\eta_{\mathrm{j}}$. Note to be careful with using the correct units for TSFC in your calculations as $1\ [\mathrm{g/kN/s}] = 1 \cdot 10^{-6}\ [\mathrm{kg/N/s}]$.

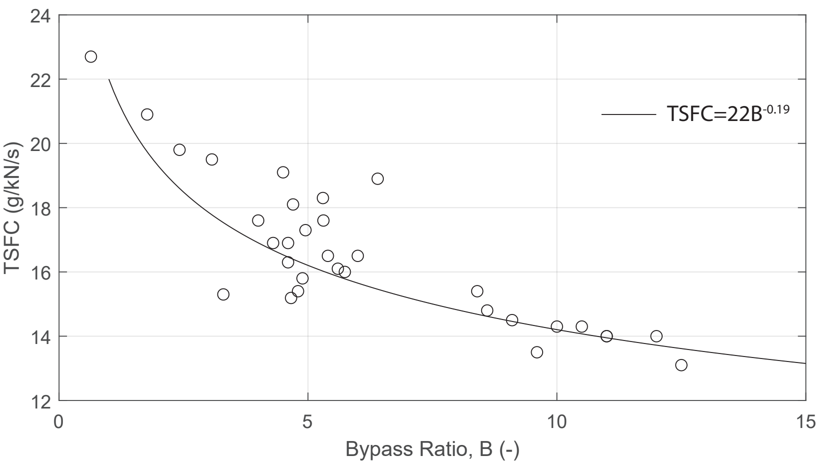


Figure 5.8: Published values cruise TSFC and bypass ratio for existing turbofan engines. Data source: wikipedia.org.

ASSIGNMENT 5.4

In this assignment, you will determine the efficiency of your propulsion system. Depending on your choice of engine (from Assignment 7.4), answer the following questions:

a. For electric motors, what electric-motor efficiency, $\eta_{\text{em}}$, do you assume?
b. For piston engines or turboprop engines, What (thermal) efficiency, $\eta_{\text{eng}}$, do you assume?
c. When using a propeller, what propulsive efficiency, $\eta_{\text{p}}$, do you assume?
d. For turbofan engines, what bypass ratio do you choose? What jet efficiency, $\eta_{\text{j}}$, do you compute?

### **5.2.3.** ESTIMATING THE EQUIVALENT RANGE

In this section, we compute the equivalent range, $R_{\text{eq}}$ that we will substitute for $R$ in (5.6) and (5.6).

Let us return to the mission profile that we introduced in Section 3.2. Examples 3.3 and 3.4 show that apart from the cruise phase, other phases make up the complete mission of the airplane. You can imagine that during take-off and climb, we throttle up the engine or motor to accelerate and climb to the cruise altitude. However, during descent and deceleration, we throttle down. From experience, we know that over the complete mission, we use more energy than the energy that is needed to fly between two points in cruise conditions. In other words, part of the range is "lost," in the sense that the energy is not converted to mission range.

This lost range can be computed as a function of the cruise speed and cruise altitude as follows [21]:

$$R_{\text{lost}} \approx \frac{1}{0.7}\left(\frac{L}{D}\right)_{\text{CR}}\left(h_{\text{CR}} + \frac{V_{\text{CR}}^2}{2g}\right) \tag{5.24}$$

where 0.7 accounts for the fact that we do not fly as efficiently during acceleration and climb as we do during cruise.

Additional energy may be required to cover a diversion range ($R_{\text{div}}$) and/or a specified endurance time ($t_{\text{E}}$) to hold at an alternate airport. We call the energy required for diversion and holding *reserve energy*.

If we know approximately how much this lost range ($R_{\text{lost}}$) is, we can apply the Breguet range equations and include the complete mission profile in a single equation. We therefore define the *equivalent range* as follows:

$$R_{\text{eq}} = \left(R_{\text{des}} + R_{\text{lost}}\right)\left(1 + f_{\text{con}}\right) + R_{\text{eq, res}} \tag{5.25}$$

In this equation, the design range ($R_{\text{des}}$) and the lost range in the first bracket determine the trip fuel. This is multiplied by $(1 + f_{\text{con}})$, where $f_{\text{con}}$ is the fraction of trip fuel that is used for contingency (e.g. headwind or rerouting). Example 3.20 showed that $f_{\text{con}} = 0$ for general-aviation airplanes and $f_{\text{con}} = 5\%$ for commercial-aviation airplanes. The equivalent reserve range ($R_{\text{eq, res}}$) is the range that can be flown under cruise conditions using the reserve energy. It is computed as follows:

$$R_{\text{eq, res}} = 1.2R_{\text{div}} + t_{\text{E}}V_{\text{CR}} \tag{5.26}$$

The factor 1.2 in (5.26) accounts for 20% additional energy that is needed for non-optimal flight during the diversion part of the mission. The following example shows how to compute the equivalent range for a turbofan-powered airplane.

**Example 5.6**

In this example, we compute the equivalent range for a turbofan-powered airplane. From Example 3.3 we have the following requirements:

1. The design range of the airplane with all passengers on board in a typical seating configuration and without additional cargo is 3000 km.
2. The airplane shall cruise at a Mach number of $M = 0.80$
3. 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.

Looking at the mission profile, we see that starting at take-off, we have the following phases where fuel is burned: take-off, climb, cruise, descent, diversion climb, diversion cruise, hold, descent, land, and taxi. We choose a cruise altitude of $h_{\mathrm{CR}} = 11$ km. The cruise speed can be computed to be $V_{\mathrm{CR}} = 236$ m/s when employing (7.30) in combination with (7.11). We compute the lost range using (5.24) while using the maximum aerodynamic efficiency of Example 5.1: $(L/D)_{\mathrm{CR}} = 16.7$:

$$R_{\mathrm{lost}} = 330 \text{ km}$$

With $t_{\mathrm{E}} = 1800$ s, and $R_{\mathrm{div}} = 400$ km, we compute the equivalent reserve range using (5.26):

$$R_{\mathrm{eq,\,res}} = 910 \text{ [km]}$$

As this is a commercial jet, we have $f_{\mathrm{con}} = 0.05$. With $R_{\mathrm{des}} = 3000$ km, (5.25) and

$$R_{\mathrm{eq}} = 4400 \text{ [km]}$$

Note that the equivalent range is almost fifty percent more than the design range. Also, note that we round the value of the range from $R = 4401.8$ km to $R = 4400$ km. We do this because the numbers that we use in the computation also do not have more than two significant figures. The number of significant figures represents the accuracy of a calculation result. In our case, much of the calculation input does not have more than two significant figures. We therefore also report the result of the calculation in two significant figures. Not only does this do justice to the calculation accuracy, but it also allows for a much quicker interpretation of the results.

### 5.2.4. COMPUTING THE ENERGY MASS FRACTION OF YOUR AIRPLANE

In the previous sections, we showed how to compute, choose, or estimate each of the ingredients of energy mass fraction equations: (5.5) and (5.6). In the next two examples, we show how the fuel mass ratio and battery mass ratio for a turbofan airplane and electric airplane can be computed.

**Example 5.7**

In this example, we compute the fuel mass fraction for a turbofan-powered airplane. Let us use the equivalent range of Example 5.6: $R_{\text{eq}} = 4400$ km. We assume the same efficiency as the CFM56 of Example 5.5: $\eta_{\text{j}} = 0.33$ and a maximum aerodynamic efficiency of Example 5.1: $(L/D)_{\text{CR}} = 16.7$. Now, the fuel mass ratio can be computed using (5.5) knowing that $\eta_{\text{j}} = \eta_{\text{eng}}\eta_{\text{p}}$ and using $R = R_{\text{eq}}$:

$$\frac{m_{\text{f}}}{m_{\text{MTO}}} = 0.14$$

---

The previous example shows that the fuel-mass fraction ($m_{\text{f}}/m_{\text{MTO}}$) is 14 percent. By varying the input values to this calculation, you can determine how sensitive the fuel-mass fraction is to the assumed values of $L/D$ and $\eta_{\text{j}}$. In the next example, we consider a general-aviation airplane that is battery-powered.

**Example 5.8**

Assume we design a single-prop, electric-powered airplane that uses batteries to store the electric energy. This is a general-aviation airplane that operates under visual flight rules (VFR). Therefore, an energy reserve of 30 minutes is required according to ICAO Annex 6 (see Example 3.20). It is to have a design range of 450 km. Furthermore, it has a cruise lift-to-drag ratio of $L/D = 13.8$, a motor efficiency of $\eta_{\text{EM}} = 0.94$, and a battery pack with a specific energy of $e_{\text{bat}} = 350$ Wh/kg. To compute ($m_{\text{bat}}/m_{\text{MTO}}$), we first need to estimate the equivalent range with (5.25). The lost range is computed with (5.24) for $h_{\text{CR}} = 1800$ m and $V_{\text{CR}} = 70$ m/s:

$$R_{\text{lost}} = 40 \text{ km}$$

With $f_{\text{con}} = 0$ and $R_{\text{div}} = 0$, and $t_{\text{E}} = 1800$ seconds we have:

$$R_{\text{eq}} = 620 \text{ km}$$

We then convert $e_{\text{sp}} = 350$ [Wh/kg] = 1.26 [MJ/kg]. We subsequently employ (5.6) to find:

$$\frac{m_{\text{bat}}}{m_{\text{MTO}}} = 0.46$$

---

You may observe that the battery-mass fraction of the previous example is much higher than the fuel-mass fraction of Example 5.7, even though the range is much smaller. This is due to the lower specific energy of batteries compared to fuel. However, the battery-mass fraction is much more sensitive to changes in $L/D$ than the fuel-powered airplane. This means that the battery-mass fraction would decrease quite rapidly with an increased lift-to-drag ratio. Furthermore, you might have seen that the assumed $\eta_{\text{EM}}$ is quite high compared to the ones we computed for the CFM56 (Example 5.5) and the Lycoming O-320 (Example 5.4). The Pipistrel E-811 electric motor (Fig. 5.9) is an example of an electric motor with a high transfer efficiency.

The previous examples featured a turbofan engine and an electric motor. The procedure for estimating the fuel mass fraction for an airplane with a reciprocating engine or a turboprop engine is the same as for the turbofan engine (see Example 5.7).

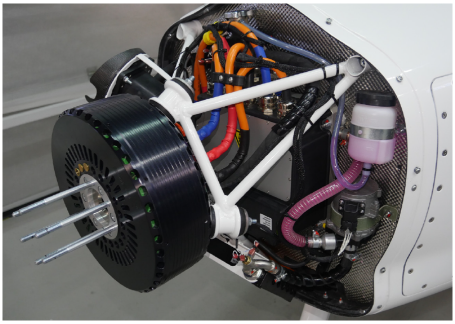

Figure 5.9: Pipistrel E-811 60kW electric motor. Photo: Pipistrel.

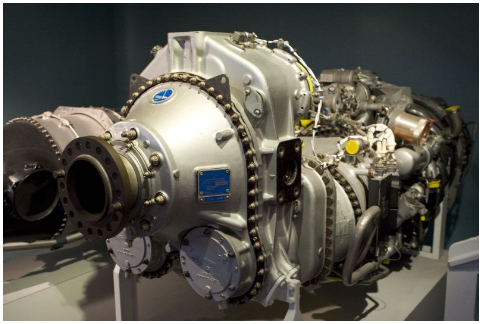

Figure 5.10: Pratt and Whitney PW120 1.3 MW turboprop engine. Photo: Imnop88a

We have shown how you can compute the fuel-mass fraction and battery-mass fraction of the airplane you are designing. The examples above have demonstrated that these fractions are sensitive to assumed input parameters such as $(L/D)$, $\eta_\mathrm{p}$ and $\eta_\mathrm{EM}$, but also to the design choice on the aspect ratio ($AR$) and properties of the energy carrier ($e_\mathrm{fuel}$ or $e_\mathrm{bat}$) in the next assignment you are going to practice with this.

**ASSIGNMENT 5.5**

In this assignment, you compute the fuel-mass fraction or battery-mass fraction of your airplane.

a. Based on your aspect ratio (Assignment 5.1), zero-lift drag coefficient (Assignment **??**) and Oswald factor (Assignment 5.3), compute the maximum lift-to-drag ratio of your airplane in cruise configuration.
b. Using (5.5) or (5.6), compute the fuel-mass fraction or battery-mass fraction, respectively.
c. If applicable, compute the reserve energy mass fraction.

## 5.3. ESTIMATING THE OPERATING EMPTY MASS FRACTION

Now that you know how to compute the mass fraction of the energy carrier, it is time to move our attention to the other unknown in the unity equation (5.2): $m_\mathrm{OE}/m_\mathrm{MTO}$, known as the (operating) empty-mass fraction.

However, before we compute this, let us first look at the difference between *empty* mass and *operating empty* mass. For small airplanes, the crew and the payload typically overlap. For example, a four-seater airplane would have a payload mass corresponding to four passengers. Little additional mass is added for operational items. Therefore, for small airplanes, we typically find that their empty mass is used in (5.2). On the other hand, larger airplanes typically do operate with a crew that is separate from the payload. In that case, we count the crew plus all additional operational items to the *operating* empty mass. Whether you should use $m_\mathrm{E}$ or $m_\mathrm{OE}$ can be decided by looking at your reference airplanes. Can you only find empty-mass quotes? Then use empty mass in (5.2). Can you only find operating empty mass quotes? Then you had better use operating empty mass in (5.2).

To compute $m_{\mathrm{OE}}/M_{\mathrm{MTO}}$, we use the data from our reference airplanes. While we design a new airplane, we need to have a good starting point for the estimation of the (operating) empty mass fraction. The best we can do at this stage is to use data from airplanes that have come before, knowing that in Chapter 12, we will analyze our airplane and come up with an improved estimation of the (operating) empty mass. The following example demonstrates how to estimate this.

**Example 5.9**

In this example, we estimate the (operating) empty mass fraction for the design of a twin piston-prop airplane. For this airplane, we have the following top-level requirements:

- Number of passengers: 4
- design range: 1700 km

Based on these top-level requirements, we have found six reference airplanes that each carry four passengers but have various mission ranges. The relevant data from these airplanes is tabulated below:[2]

| name | pax | $R$ (km) | $m_{\mathrm{MTO}}$ (kg) | $m_{\mathrm{E}}$ (kg) | $m_{\mathrm{E}}/m_{\mathrm{MTO}}$ | year |
|---|---|---|---|---|---|---|
| Diamond DA-42 | 4 | 2250 | 1999 | 1251 | 0.63 | 2004 |
| Beechcraft Duchess | 4 | 1440 | 1769 | 1116 | 0.63 | 1978 |
| Cessna 310 | 4 | 1600 | 2087 | 1293 | 0.62 | 1954 |
| Let-200D | 4 | 1710 | 1950 | 1330 | 0.68 | 1957 |
| Tecnam PT2006 | 4 | 1204 | 1230 | 819 | 0.67 | 2010 |
| Piper PA-44 | 4 | 1695 | 1724 | 1086 | 0.63 | 1976 |

Note that the open literature quotes the empty mass ($m_{\mathrm{E}}$) of the reference airplanes. Therefore, we also use the empty mass in our computations. To determine $m_{\mathrm{E}}/m_{\mathrm{MTO}}$, we can simply take the average of the values listed for the reference airplanes. However, we can also take two other factors into account: the range and the year the airplane was first flown. The range of the airplane impacts the fuel-mass fraction of an airplane and, therefore, also the empty-mass fraction. The year an airplane was first flown says something about the technology that was used in constructing the airplane. Newer airplanes are likely to be more representative than older airplanes. In this instance, the two newest planes (DA-42 and PT2006) have an empty-mass ratio of 0.63 and 0.67, respectively. The design range of our airplane is in between the ranges of those airplanes. The Let-200D and PA-44 are closest in terms of range and have an empty-mass ratio of 0.68 and 0.63, respectively. However, the Duchess and Cessna 310 are not far off in terms of range, having an empty-mass ratio of 0.63 and 0.62, respectively. Based on this qualitative assessment, we choose to assume a value of $m_{\mathrm{E}}/m_{\mathrm{MTO}} = 0.64$, which is close to the arithmetic mean of all the airplanes.

This example has demonstrated how one can make a well-funded assumption for the empty-mass fraction of the airplane you are designing. Not only is it important to select the right reference airplanes, but it is equally important to assess how close the range of

[2] All data from wikipedia.org.

the airplane is to the range that is specified in the top-level requirements. If the range of a reference airplane is much less or much more than the required range of your airplane, you might want to discard that airplane when computing the (operating) empty-mass fraction. Secondly, it is important to take the year of first flight into account. Newer airplanes typically take advantage of improved technology levels, which typically reduce the weight of the airframe, the powerplant, and the systems. In the following assignment, you estimate the (operating) empty-mass fraction for your airplane.

**ASSIGNMENT 5.6**

In this assignment, you will estimate the (operating) empty mass of your airplane. You need to consult the reference airplanes that you have selected in Assignment 3.9.

a. For each reference airplane, find the maximum take-off mass value in kg.
b. For each reference airplane, find the empty mass or operating empty mass in kg.[a]
c. Compute the (operating) empty-mass fraction for each reference airplane.
d. Make an estimation of the (operating) empty-mass fraction of your airplane.

[a]whichever one is listed in the open literature

So what do you do if there is no or a limited number of reference airplanes to consult? In that case, you try to work with what you do know. At the time of writing this textbook, few battery-powered airplanes were fielded. So, how do you estimate a battery-powered airplane's (operating) empty-mass fraction? First, it is good to know what (operating) empty-mass fractions are feasible from a practical point of view. On the upper end, we can imagine an airplane without payload and zero battery mass (essentially a glider), which has a theoretical $m_{\mathrm{OE}}/m_{\mathrm{MTO}} = 1$. On the lower end, we can think of airplanes that have been operated over an extremely long range (i.e. high $m_{\mathrm{f}}/m_{\mathrm{MTO}}$) and therefore had to have a low $m_{\mathrm{OE}}/m_{\mathrm{MTO}}$. The Rutan Voyager (Figure 5.11) and Rutan Global Flyer (Figure 5.12), were both designed to fly around the world. Their respective empty-mass ratios were 0.23 and 0.17.[3] Naturally, these airplanes were experimental, did not need compliance with the airworthiness regulations, and had little comfort for the crew. Therefore, they were very lightweight. Nonetheless, the empty-mass fractions of these airplanes could serve as an indicative lower bound for the feasible (operating) empty-mass fraction range. Secondly, you might want to consult studies into battery-electric airplanes. You can use data from design reports or scientific publications to estimate your (operating) empty-mass fraction.

## 5.4. ESTIMATING THE MAXIMUM TAKE-OFF MASS

Now that we have computed the energy-mass fraction and the (operating) empty mass fraction, we can compute the maximum take-off mass of our airplane. We rearrange (5.2)

[3]Source: wikipedia.org

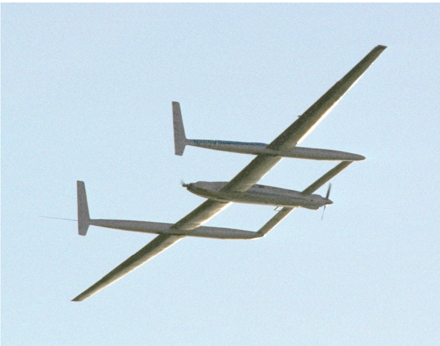

Figure 5.11: The Rutan Voyager has an empty-mass fraction of 0.23. Photo: NASA/Thomas Harrop

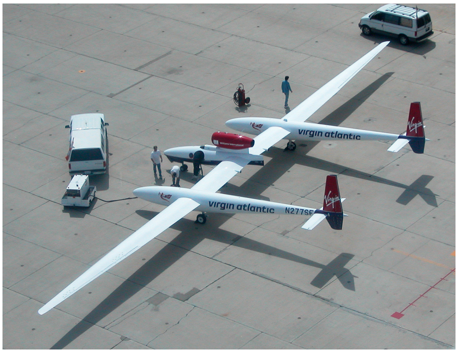


Figure 5.12: The Rutan Global Flyer has an empty-mass fraction of 0.17. Photo: Alan Radecki

to bring $m_{\mathrm{MTO}}$ to the left-hand side of the equation:

$$m_{\mathrm{MTO}} = \frac{m_{\mathrm{pl}}}{1 - \left(\frac{m_{\mathrm{OE}}}{m_{\mathrm{MTO}}}\right) - \left(\frac{m_{\mathrm{f}}}{m_{\mathrm{MTO}}}\right)} \tag{5.27}$$

The right-hand side of this equation is the payload mass, for which the design payload mass should be used. The other two mass fractions have been explained in the previous text. The next example demonstrates this.

**Example 5.10**

In this example, we consider the twin prop of Example 5.9. We have $m_{\mathrm{pl}} = 400$ kg, $(m_{\mathrm{E}}/m_{\mathrm{MTO}} = 0.64)$ and $(m_{\mathrm{f}}/m_{\mathrm{MTO}}) = 0.15$. Substituting this in (5.27) results in:

$$m_{\mathrm{MTO}} = 1900 \text{ kg}$$

When comparing this value to the reference airplanes, we see that it falls within the range of values that are listed under $m_{\mathrm{MTO}}$. With the payload mass fractions, we can now directly compute the fuel mass and empty mass for this airplane:

$$m_{\mathrm{E}} = 1200 \text{ kg} \qquad m_{\mathrm{f}} = 290 \text{ kg}$$

Again, we round the numbers in these calculations to two significant figures to represent the accuracy of our calculations. However, we store the unrounded numbers to employ them in the calculations that are yet to come. In the next assignment, you compute the characteristic masses for your airplane. Use the correct number of significant figures when reporting the masses.

ASSIGNMENT 5.7

In this assignment, you will compute the characteristic masses of your airplane. Perform the following tasks:

a. Compute the maximum-take-off mass of your airplane
b. Compute the (operating) empty mass of your airplane
c. Compute the energy mass of your airplane in the design condition
d. Compute the reserve energy mass of your airplane.
e. In the diagram of Assignment 3.9, plot the data point of your airplane. How does it correspond to the data points of your reference airplanes?

Now that you have calculated the take-off mass of your airplane, look back at the design process shown in Figure 2.1. You can see that the maximum take-off mass is one of the important metrics needed for many of the next steps. For CS/FAR 23 airplanes, you now also know whether your airplane is going to be above or below 6,000 lb (2722 kg), telling you which set of climb gradient requirements to select (see Example 3.13). Also, think about the decisions you have already made. You have selected the type of energy carrier. You have selected an aspect ratio. You have selected a propulsion system. You may have even selected whether you wish to fly this airplane under IFR or only under VFR. These decisions are all design choices and are an important step in the design of your airplane.

## 5.5. CONSTRUCTING THE PAYLOAD-RANGE DIAGRAM

In this section, we are going to construct one of the most important performance diagrams of an airplane: the payload-range diagram. An example of a payload-range diagram for a large passenger airplane is shown in Figure 5.13. This diagram shows the payload mass on the vertical axis and the range on the horizontal axis. The solid pink line in the diagram forms a boundary: any combination of payload and range that lies within this boundary can be flown by the A350-1000. Any point outside this boundary cannot. The diagram, therefore, informs us about the possible productivity of the airplane. With this diagram, we can plan which routes can be served by this airplane and what the maximum payload can be on each route. The dashed line indicates the payload mass for a typical design payload: 366 passengers plus their luggage. The associated range is approximately 15,600 km. You can see that there is a (large) difference between the design payload and the *maximum structural payload*. The maximum structural payload is the maximum payload that the structure of the airplane can carry. The latter is almost twice as much as the former.

In Example 3.2, we showed that there can be multiple payload-range requirements imposed on an airplane at the same time. Only one of these requirements leads to the highest maximum take-off mass. The associated requirements are termed the *design payload* and the *design range*. Every other combination of the required range and payload will fall within the boundaries of the payload-range diagram. In Figure 5.13, the design range and design payload is the second kink in the payload-range diagram: $m_{\text{p, des}} = 32$ t and $R_{\text{des}} = 15,800$ km. This means that the fuel tanks are dimensioned to carry the fuel required to transport the payload over the nominal design range. If we wish

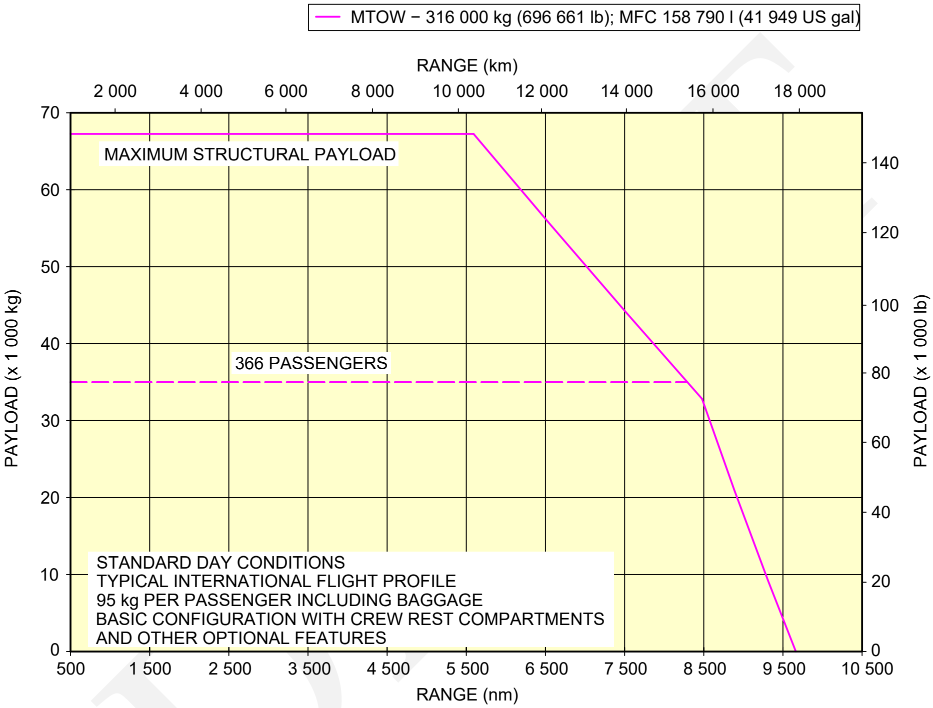


Figure 5.13: Payload-range diagram of the A350-1000 as published by Airbus in the document “Airplane Characteristics Airport and Maintenance Planning.”

to fly further, then we can reduce the payload mass, causing an increased range through the Breguet range equations (5.3) or (5.4). We can reduce the mass of the payload to zero, which results in the maximum range that the empty airplane can fly: the *ferry range*. This works the same for fuel-powered airplanes as for battery-powered airplanes. When computing the range through (5.3) or (5.4), we need to be careful to subtract the so-called *auxiliary range*, i.e., the difference between the equivalent range and the design range:

$$R_{\text{aux}} = R_{\text{eq}} - R_{\text{des}} \tag{5.28}$$

The auxiliary range is the range of the airplane that would be covered if the energy required for climbing, accelerating, holding, and diverting would be used to fly in a straight line under cruise conditions. The following example shows how this works for a battery-powered airplane:

**Example 5.11**

Let us take the airplane of Example 5.8 and produce the payload-range diagram.

1. The design range of this airplane is 450 km with a payload of 200 kg. Since the battery mass does not change if we fly a shorter mission, the design payload mass equals the maximum structural payload mass. This gives us two points in the payload range diagram, which are tabulated below.
2. Before we compute the range at zero payload mass, we first compute the auxiliary range, i.e., the equivalent range that is needed for climbing, accelerating, and holding (5.28):
$$R_{\text{aux}} = 620 - 450 = 170 \text{ [km]}$$
3. We employ (5.4) to find the range that can be flown at $m_{\text{pl}} = 0$:
$$R = \eta_{\text{em}}\eta_{\text{p}}\left(\frac{L}{D}\right)\left(\frac{e_{\text{bat}}}{g}\right)\left(\frac{m_{\text{bat}}}{m_{\text{OE}} + m_{\text{bat}}}\right) - R_{\text{aux}} = 530 \text{ [km]}$$
We add this value to the table below as the theoretical ferry range of this airplane.
4. We realize that this is a two-person airplane and that at least one person is needed to fly the airplane. In other words, the payload cannot go to zero when we have a pilot that is part of the payload mass. When we assume that a single pilot (plus luggage) has a mass of 100 kg, we find a practical ferry range of 490 km.
5. We use the data points to construct the payload range diagram of Figure 5.14.

| $m_{\text{pl}}$ | [kg] | 200 | 200 | 0 |
|---|---|---|---|---|
| $R$ | [km] | 0 | 450 | 530 |

The previous example showed that the maximum structural payload of a battery-powered airplane is equal to the design payload. Fuel-powered airplanes, however, have the ability to trade fuel mass for payload mass while keeping the maximum take-off mass constant. If there exists a requirement to fly with a payload mass higher than the design payload mass, that payload mass becomes the maximum structural payload mass $m_{\text{pl max}}$.

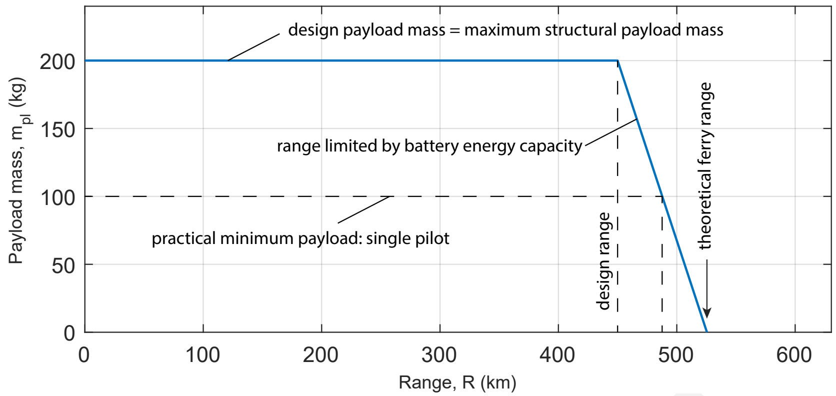


Figure 5.14: Payload-range diagram for a two-seat electric airplane.

The fuel mass available to fly with maximum structural payload mass can be computed as follows:

$$m_f = m_{MTO} - m_{OE} - m_{pl\,max} \tag{5.29}$$

To compute the associated range, we can employ (5.3). However, we need to be careful as the resulting range is the *equivalent* range. In order to compute the design range, we still need to subtract the auxiliary range $R_{aux}$.

**Example 5.12**

We construct the payload-range diagram for a jet airplane with a design range of 3000 km and a design payload of 19 tonnes. Furthermore, this airplane has a maximum structural payload requirement of 23 tonnes. Using the methods introduced in the previous section and the values of Example 5.7, we have the following mass characteristics for the design condition (all in tonnes):

| $m_{MTO}$ | $m_{OE}$ | $m_f$ | $m_{pl}$ |
|---|---|---|---|
| 63 | 35 | 9.1 | 19 |

We can now start constructing the payload range diagram.

1. The first point in the payload range diagram that can be easily constructed is zero range combined with our maximum structural payload (first column in the table below).
2. The second point is the combination of design range and design payload (third column in the table below).
3. Before we compute any of the other non-zero range points, we first compute the auxiliary range with (5.28):

$$R_{aux} = 4400 - 3000 = 1400 \text{ [km]}$$

4. We compute the available fuel mass at the maximum structural payload using (5.29):

$$m_f = 5.1 \text{ [t]}$$

5. Now, we compute the range at maximum structural payload with (5.3) for $m_{\text{pl}} = 23$ tonnes:

$$R = \eta_{\text{eng}}\eta_{\text{p}}\left(\frac{L}{D}\right)\left(\frac{e_{\text{f}}}{g}\right)\ln\left(\frac{m_{\text{OE}} + m_{\text{pl}} + m_{\text{f}}}{m_{\text{OE}} + m_{\text{pl}}}\right) - R_{\text{aux}} = 990 \text{ [km]}$$

   We can add this combination of payload mass and range to the table below (second column)

6. We compute the ferry range by employing (5.3) with $m_{\text{pl}} = 0$ and using the same fuel mass as for our design range, i.e. $m_{\text{f}} = 9.1$ tonnes:

$$R = \eta_{\text{eng}}\eta_{\text{p}}\left(\frac{L}{D}\right)\left(\frac{e_{\text{f}}}{g}\right)\ln\left(\frac{m_{\text{OE}} + m_{\text{pl}} + m_{\text{f}}}{m_{\text{OE}} + m_{\text{pl}}}\right) - R_{\text{aux}} = 5100 \text{ [km]}$$

   We add this data point to our table in the last column below.

7. Based on the payload-range points in the table below, we can now construct the payload-range diagram in Figure 5.15.

| $m_{\text{pl}}$ | [t] | 23 | 23 | 19 | 0 |
|---|---|---|---|---|---|
| $R$ | [km] | 0 | 990 | 3000 | 5100 |

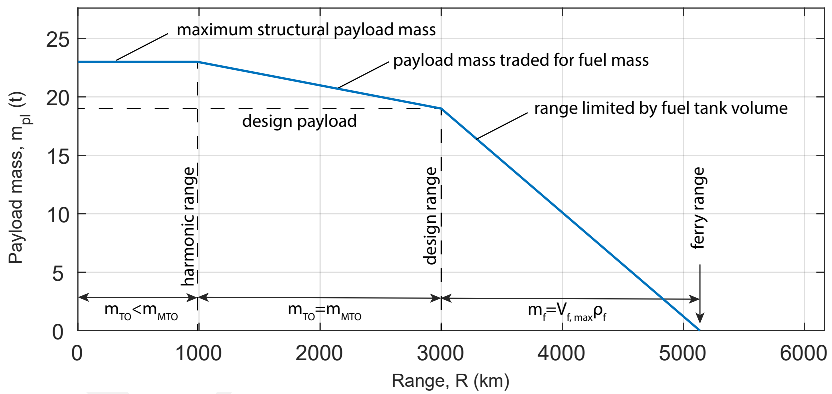


Figure 5.15: Payload-range diagram for a 190-seat turbofan airplane.

The previous example produced a payload-range diagram that looks similar to the one we saw for the A350-1000 in Figure 5.13. We can observe that an increase in payload mass rapidly reduces the available range of the airplane. On the other hand, if we fly this airplane with a payload mass of only 10 tonnes, we can achieve a range of 4000 km. You can see that below the design payload, the available range is limited by the tank volume, $V_{\text{f}}$. We have implicitly assumed that the tanks of our airplane are sized to the design-range requirement. However, we might find out in Chapter 10, that there is more volume available in the airplane to store fuel. If this is the case, we can choose to trade payload for fuel beyond the design range. We can do this until the tank volume is completely filled with fuel. The second kink, as well as the ferry-range point, will then shift to the right in our payload-range diagram.

**ASSIGNMENT 5.8**

In this assignment, you will construct the payload-range diagram of your airplane:

a. State the combination of design range and design payload.
b. Do you have a payload-mass requirement that qualifies as a maximum structural payload? If so, state the maximum structural payload mass $m_{\mathrm{pl\,max}}$ and compute the range at the maximum structural payload.
c. Compute the ferry range of the airplane at $m_{\mathrm{pl}} = 0$
d. Using the answers to the previous questions, construct the payload range diagram.

# 6

# Fuselage Design

In this chapter, you are going to design the fuselage of your airplane. The primary function of a fuselage is to house the payload. The volume of the payload is, therefore, a driving requirement for the size of the fuselage. The type of payload also influences the design of the fuselage. If the payload consists only of cargo, it can be very densely packed, resulting in a relatively small fuselage compared to the cargo volume. Passengers, on the other hand, require more volume than the volume that is occupied by their bodies. They need sufficient space to feel comfortable. Furthermore, additional space is needed for lavatories, galleys, and carry-on luggage. Also, aisles are needed to reach the seats and provide a path for emergency evacuation. To facilitate the loading and unloading of the passengers and/or cargo, the fuselage also needs to provide access doors.

The fuselage does not only provide a volume to store the payload, it also provides the payload protection from the environment in terms of wind, extreme temperatures (both hot and cold), and noise. For airplanes that fly at high altitudes, it also ensures that passengers are protected from the low ambient pressure. At this low pressure, the air does not contain enough oxygen, which can lead to hypoxia.

The fuselage also provides space for the cockpit and an array of systems. For small airplanes, this includes the *avionics* (aviation electronics), the engine or motor, a fire extinguisher, part of the fuel system, part of the flight control system, fuel or batteries, and a battery. Larger airplanes can have an environmental control system (ECS), an auxiliary power unit (APU),, and landing-gear bays. Also, hydraulic, pneumatic, and electric lines are running throughout the fuselage to connect all these systems. Furthermore, weather radar and various antennas are often installed to facilitate communication. Finally, the fuselage also provides connection points for the wing, the landing gear, and the tail.

To design the fuselage of your airplane, we are going to take a step-by-step approach. In Section 6.1, we first discuss some design considerations regarding payload, emergency evacuation, aerodynamics, ground clearance, structures, ground handling, and pilot visibility. Prior to designing the fuselage, we first analyze the payload in terms of its volume (Sec. 6.2). Then, we first show how to design the cross-section of the fuselage (Sec. 6.3). We subsequently design the fuselage in the top view (Sec. 6.4) and in the side view (Sec. 6.5).

The design of the fuselage is the starting point of a three-view design of the complete airplane. In Chapters 8 and 9, you will dimension the wing, the propulsion system, the landing gear, and the tail. The fuselage acts as the connecting element between all of these components and is, therefore, our starting point for the three-view drawing of the airplane.

## 6.1. DESIGN CONSIDERATIONS

The fuselage is one of the largest components of the airplane. When it is designed, there are many aspects to consider. Some aspects of its design have been clearly quantified in terms of design requirements. An example is the requirement to store the payload. However, requirements regarding ground handling can also be quite explicit. Other aspects, such as aerodynamics and structures, are often implicit in the design objective. We wish to minimize the drag and minimize the structural weight. In the subsequent sections, we will explain which considerations are at play in the mind of the airplane designer when designing the fuselage.

### 6.1.1. PAYLOAD AND ENERGY

As stated above, the fuselage size is primarily determined by the payload requirements. Figure 6.1 shows the unloading of fuselage barrels from the fuselage of a cargo airplane. This cargo airplane has clearly been designed to fit large components inside its fuselage. As these components do not require oxygen, only the cockpit of the cargo airplane needs to be pressurized. The size of the fuselage is determined by the size or volume of the cargo that you want to transport. Most passenger airplanes also have storage volume for goods. This can be luggage or cargo. In either case, the volume of these goods determines how much volume is needed in the fuselage to store them.


Figure 6.1: The payload requirements for a cargo airplane can be completely different from a passenger airplane leading to different design solutions. Photo by David .

When your airplane transports passengers, you need to think about the level of comfort you wish to include in your design. More space per passenger means a more comfortable flying experience. However, it also means a larger, heavier, draggier fuselage. In other words, more comfort for the passengers means more energy is required to transport them. This goes against the objective of minimum fuel burn, minimum operating cost, or minimum climate impact. You should, therefore, make a careful trade-off between your design objective and the comfort level you want to provide.

Apart from passengers and cargo, the fuselage can also provide volume for large systems. Many fighter aircraft, for example, have their engines buried within the fuselage. Also (part of) the fuel or batteries can be stored in the fuselage. Long-range aircraft that do not have sufficient volume in their wings to store kerosene can use additional fuselage tanks to accommodate the total fuel volume. Pressurized hydrogen or LNG tanks can be difficult to fit into a planar wing and can, therefore, be integrated into the fuselage. The same argument can be made for battery packs. An example of a concept aircraft with a hydrogen fuel tank integrated into the fuselage can be seen in Figure 6.2.


Figure 6.2: Concept airplane with hydrogen tanks integrated into the fuselage. Photo: Airbus.

### 6.1.2. EMERGENCY EVACUATION

In Example 3.15 and Example 3.16, we presented the requirements that CS-23 and CS-25 airplanes need to comply with regarding emergency evacuation. In Assignment 3.8, you decided on the number of emergency exits and where each exit should be located (left or right). In this chapter, you need to determine the size of each door and what its longitudinal position on the fuselage needs to be. In order to make that decision, you can keep the following considerations in mind:

- distribute the emergency exits over the length of the fuselage as "uniformly as possible." If all exits are cluttered together, it will take longer for all passengers to get out.
- If you have a pair of emergency exits, position them across from each other such that they can be connected via a cross aisle.
- If you have over-the-wing exits, make sure they are positioned at the location of the wing. If you do not know the wing location, position them at a longitudinal location where you expect the wing root to be positioned. Update the longitudinal location of the exit(s) once the position of the wing root is known.
- One or more emergency exits are likely to be used for boarding of passengers and/or crew. Other emergency exits might be used for servicing the airplane (see Section 6.1.7). Therefore, you may want to increase the size of the emergency exits to improve these processes.

For CS-23 airplanes, the minimum size of the emergency exit should be 48-by-66 cm


Figure 6.3: Example of a Type II exit on a business jet. Photo: Anna Zvereva

(width × height). This corresponds to a so-called Type IV exit according to CS-25 regulations. This minimum size is not very comfortable for boarding an airplane. Therefore, the boarding door is usually larger than the minimum size. In Figure 6.3, you can see an example of a CS-23 airplane with its main exit open. According to the regulations, this airplane should have at least two Type IV exits. However, the designers have chosen to have one Type IV exit and one Type II exit. The type II exit is larger and is located on the left-hand side of the fuselage. The Type IV exit is an overwing exit and is located on the opposite side of the fuselage.

CS-25 specifies various types of exits. In Example 3.16, we showed how many passengers are allocated per pair of emergency exits of a given type. The most common are exits located at cabin floor level and overwing exits. Their dimensions are shown in Figure 6.4. For the overwing exits, a maximum step height is defined from the inside to the door ($h_1$) and from the door onto the wing ($h_2$). Table 6.1 shows the dimensions of the most common door types. An exhaustive list of emergency exit types and associated dimensions can be found in CS-25.807.

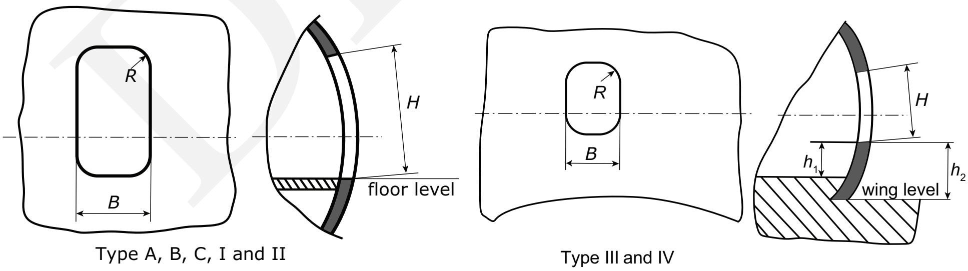


Figure 6.4: Definition of emergency exit dimensions After Ref. [20].

### 6.1.3. AERODYNAMICS

The fuselage is generally not intended to generate any lift force. However, when subjected to an angle of attack or angle of sideslip, it generates resulting forces. These forces

Table 6.1: Dimensions of emergency exit types as defined in CS-25.807. Dimensions are defined in Figure 6.4. 1 inch = 2.54 cm.

| Type | Location | min. dimensions $B \times H$ (in.) | max. step height $h_1$:$h_2$ (in.) |
|---|---|---|---|
| Type A | floor level | 42 × 72 | - |
| Type B | floor level | 32 × 72 | - |
| Type C | floor level | 30 × 48 | - |
| Type I | floor level | 24 × 48 | - |
| Type II | floor level | 20 × 44 | - |
| | overwing | 20 × 44 | 10:27 |
| Type III | overwing | 20 × 36 | 24:27 |
| Type IV | overwing | 19 × 26 | 29:36 |

impact the longitudinal, lateral, and directional stability of the airplane. In addition, the fuselage has a large contribution to the drag of the airplane, mainly due to its large wetted area. How the fuselage shape relates to the drag is further expanded below.

In Figure 6.5, we show how the total drag of a fuselage-like body scales with the inverse of its so-called *fineness ratio*. The fineness ratio is the ratio between the length of the body ($l$) and the diameter of the body ($d$). We note that the friction drag reduces with the inverse of the fineness ratio and has a minimum when the body is a sphere. This makes sense because a sphere has the smallest ratio of surface area to internal volume. However, a sphere is not an aerodynamically shaped body. Therefore, it has a high *pressure drag*. If we stretch the sphere into a tube, the pressure drag reduces. Simultaneously, the friction drag starts to increase because the wetted area increases for the same internal volume. A minimum in total drag is found when the fineness ratio is $l/d \approx 3$.

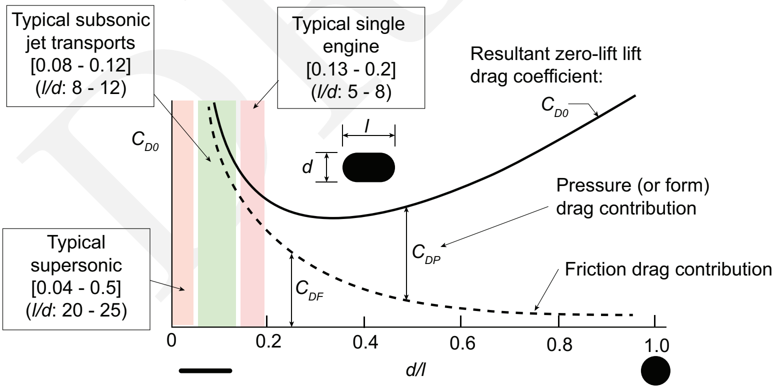


Figure 6.5: The zero-lift drag coefficient of the fuselage is dependent on the fineness ratio, $l/d$. After Corke [3]
.

Superimposed on the curves of Figure 6.5 are colored bands that indicate typical fineness ratios of actual fuselage shapes. These all have a considerably higher fineness ratio than the one for minimal drag. For all airplanes, this has to do with the fact that a longer fuselage creates a longer arm from the center of gravity to the tailplanes. A longer arm means that a smaller horizontal and vertical tail is required to provide adequate stability and control (see also Chapter 9). For pressurized fuselages, the circumferential (hoop) and axial loads in the skin that result from pressurization scale linearly with the diameter. Therefore, a higher fineness ratio results in a lower diameter and, therefore, a thinner and lighter fuselage skin. Supersonic airplanes have a very high fineness ratio because the pressure drag stemming from conical shock waves correlates quadratically to the fineness ratio of the nose and tail of the fuselage [19]. A higher fineness ratio, therefore, reduces the supersonic drag of these airplanes.

### 6.1.4. PILOT VISIBILITY

In order to provide a safe flying environment when flying under visual flight rules (VFR), the pilot needs to have good visibility of the surroundings of the airplane. Therefore, CS-23 and CS-25 include requirements that provide a minimum level of pilot visibility. This dictates the minimum size of the cockpit windows. Naturally, you can choose to go beyond this minimum level by creating larger cockpit windows than would be required according to the regulations. The visibility of the pilot is described in terms of a field of view measured in degrees with respect to the horizontal line of sight. The horizontal line of sight is a notional horizontal line that starts at the pilot's eye (pe) and is parallel to the longitudinal axis of the fuselage. With respect to this line, the field of view in the horizontal plane can be described by the rotation of the pilot's head. In addition, the vertical field of view is described by how much the maximum upward view angle and the maximum downward view angle. This is schematically shown in Figure 6.6.

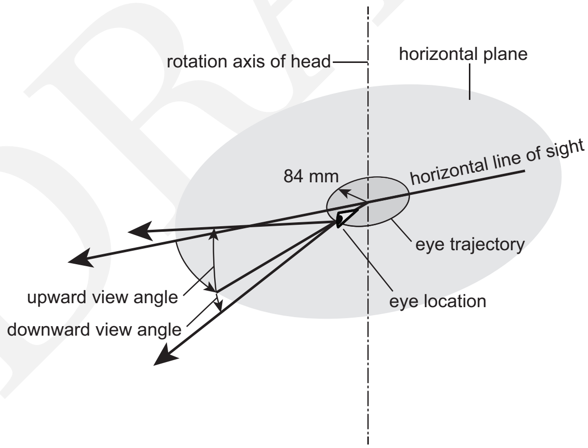


Figure 6.6: Pilot field of view in three dimensions

If we concentrate on the symmetry plane of the airplane, we show how the requirements on the downward view angle and upward view angle influence the shape of the nose and the size of the cockpit window. For simplicity, we assume that the pilot's eye is located on the symmetry plane. We also have the cross-section of the nose of the fuselage, along with the location and dimensions of the cockpit windows (see Figure 6.7).

Then, we can draw a straight line from the pilot's eye through the upper left corner of the window and measure the upward view angle. Similarly, we can draw a straight line from the pilot's eye through the lower left corner of the window and find the downward view angle. The latter angle is referred to as the *over-nose angle*, because the nose shape can influence how large the downward view angle can be. The angle between the window pane and the downward view angle is called the *grazing angle*. To prevent distortion, this angle is recommended to stay below 30 degrees, [12] although values below 20 degrees are also present in existing airplanes as can be seen in Figure 4.6.

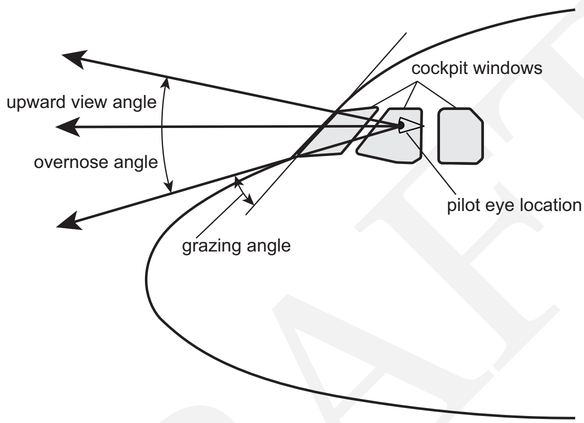


Figure 6.7: Pilot field of view in vertical plane

The requirements for pilot visibility can come from the regulations or from the customer. For example, EASA specifies the requirements in Ref. [4]. You can imagine that the pilot of a combat airplane requires a much larger field of view than the pilot of a small general aviation airplane. How the requirements affect the shape of the nose and the size of the cockpit windows depends on the visibility requirements and the attitude variation of the airplane during flight. An airplane that lands at a very high pitch attitude might require a high over-nose angle such that the pilot can see part of the runway during landing in foggy weather. The attitude of the airplane during flight is determined by the design of its wing, its speed, its flight path angle, its weight, and the deployment of trailing-edge flaps. In the first iteration of the airplane design, we cannot yet assess all of these aspects. However, after you have designed the wing and its high-lift devices (Chapter 11), you can derive what the over-nose angle needs to be.

### **6.1.5.** GROUND CLEARANCE

The design of the fuselage tail cone is influenced by the attitude of the airplane during landing. To prevent the tail of the fuselage from striking the ground during take-off or landing, the tail cone often has an upsweep. Figure 6.8 shows how the upsweep angle is defined for two different tail cone geometries. As you can see, a horizontal line is defined

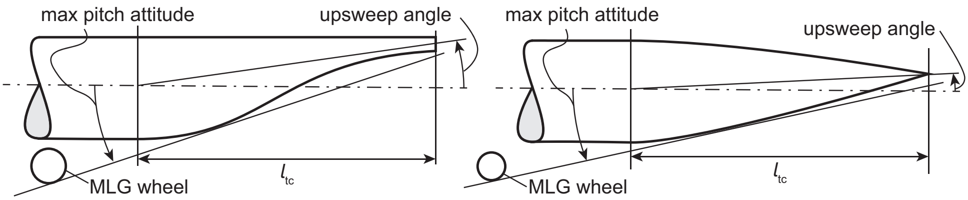


Figure 6.8: Example of a tail-cone upsweep angle on a general aviation airplane (left) and a transport airplane (right). MLG = main landing gear.

in the middle of the fuselage. The tail cone upsweep angle is defined with respect to this horizontal line. The upsweep allows for an increase in clearance between the ground and the fuselage when the airplane has an increased pitch attitude while the wheel of the main landing gear (MLG) is on the ground. The required upsweep angle to have sufficient ground clearance is, therefore, dependent on the same design parameters and operational parameters that determine the over-nose angle. In addition, the upsweep is dependent on the length of the main landing gear struts, the tire diameter, and the characteristics of the shock absorber. Therefore, in the first iteration of the design, you should choose a tail cone upsweep angle and re-evaluate this choice after having designed the landing gear (Chapter 9).

For an airplane with a tail wheel (i.e. conventional landing gear) instead of a nose wheel, an upsweep of the tail cone is not needed to ensure ground clearance. However, as the main landing gear is positioned more forward for an airplane with a conventional landing gear configuration, tail cone upsweep might still be advantageous to allow for an increase in maximum pitch attitude. An increase in pitch attitude can result in a lower stall speed in case of a tail-wheel landing (see also Section 4.3.1 on page 69). On the other hand, many airplanes with a conventional landing gear do not have upsweep in the tail cone, such as the airplane shown on the left side in Figure 4.24 on page 68. Therefore, you can decide whether or not to include an upsweep in your tail cone if you are designing an airplane with a conventional landing gear configuration.

### 6.1.6. STRUCTURES

The function of the structure of the fuselage is to protect the payload from wind, low pressure, and impact during a crash. It also provides attachment points for the wing, tail, and possibly the landing gear. It must be strong enough to carry all the loads, be durable, and be lightweight. The weight of the fuselage itself and the payload inside it produces a distributed load in the structure. The wing introduces a more concentrated load at the intersection between the wing and the fuselage. Also, the tail surfaces introduce loads. If the fuselage is pressurized, this causes additional loads. The fuselage structure should, therefore, be sized to withstand various combinations of these loads.

A fuselage structure for an airplane can be characterized as a stiffened, thin-walled structure. The skin panels transfer shear stresses and normal stresses. Connected to the skin panels are stringers that run in the longitudinal direction, also known as *longerons*. They carry part of the normal loads and prevent the skin from local buckling under compression load. *Frames* are added perpendicular to the centerline to prevent global buckling of the fuselage under bending or torsion loads. Frames are also used to introduce concentrated loads into the thin-walled structure. Some of the frames are strengthened with plates to carry loads from the wing or landing gear. These frames are called *bulkheads*. A separate floor structure is present on larger airplanes to transfer the payload loads to the frames. This floor structure typically consists of lateral floor beams and longitudinal seat tracks. Figure 6.9 shows the main structural elements in a cross-section of a pressurized fuselage.

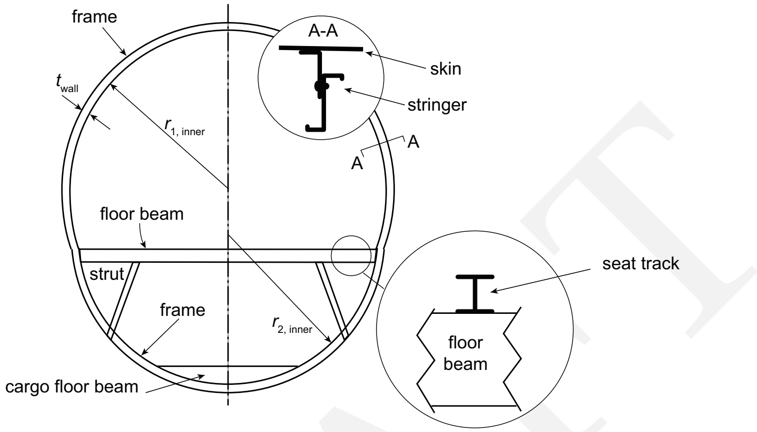


Figure 6.9: cross-section of a double-bubble fuselage with annotation of structural elements. After Niu [7].

For airplanes with pressurized cabins, a circular cross-section is preferred because it minimizes the structural weight that is required to sustain the pressurization loads. Any non-circular shell structure that is pressurized needs to be strengthened with additional structural elements to prevent large deformations. In Figure 6.9, a so-called *double-bubble* structure is shown where the floor beam connects the upper and lower fuselage shells that each have a circular shape. When pressurized, the floor beam is loaded in tension and prevents the shape from deforming. A double-bubble is sometimes preferred over a pure circular cross-section as it can result in a smaller perimeter of the fuselage cross-section and a smaller radius of the upper and lower arcs. This results in lower friction drag and lower pressure-induced loads in the skin, respectively.

In the design of the cross-section, you need to account for appropriate distances for the structural depth of the side wall. We propose the following design rules:

$$\text{Small commercial airplanes (less than 20 passengers):} \quad t_{\text{wall}} = 40 \text{ mm} \tag{6.1}$$

$$\text{Fighters and trainers:} \quad t_{\text{wall}} = 50 \text{ mm} \tag{6.2}$$

$$\text{Transport and business airplanes:} \quad d_{\text{f, outer}} = 1.045 \cdot d_{\text{f, inner}} + 0.084 \text{ m} \tag{6.3}$$

where $d_{\text{f, outer}}$ is the outer diameter of the fuselage, and $d_{\text{f, inner}}$ is the inner diameter of the fuselage. For a double-bubble fuselage (referring to Figure 6.9), you can use an equivalent inner diameter:

$$d_{\text{f, inner}} = r_{1,\text{ inner}} + r_{2,\text{ inner}} \tag{6.4}$$

The depth of the passenger floor also depends on various factors, including the fuselage diameter, the passenger load, and how the floor is supported. We propose to use a structural depth of the passenger floor between 100 mm and 300 mm, depending on the size of the airplane.

A pressurized fuselage has a bulkhead ahead of the flight deck and at the end of the cabin. The former is called the *front pressure bulkhead* and the latter bulkhead is referred to as the *aft pressure bulkhead.* This bulkhead is often spherical in shape, although some airplanes have a flat, stiffened aft pressure bulkhead.

### 6.1.7. GROUND HANDLING

When the airplane is on the ground, it interfaces with the local airport infrastructure. Regarding the design of the fuselage, you need to think about how people and/or cargo can embark or debark the airplane. For a small airplane, this means you have to think about the location of the doors and how you can access the airplane. Sometimes, a step on the wing is enough to enter the airplane, but stairs can also be required. For military transport airplanes, it can be important to be self-sufficient when it comes to the ground handling of the airplane. We have seen examples in Chapter 4 of military airplanes where the fuselage has ramps to load and unload cargo. Large civil transport airplanes, on the other hand, rely on available ground handling equipment at the airport. As can be seen in Figure 6.10, most of this equipment interfaces with the fuselage.

We shortly go over all the ground equipment that is connected to the airplane of Figure 6.10 starting at the top left quarter and making our way clockwise around the airplane. First of all, we see an air-conditioning (AC) cart, which provides fresh air to the cabin when the engines and APU are shut off. Then, we see the passenger boarding bridge (PBB), which can also be replaced by stairs at the front and/or the aft passenger door. Subsequently, we have the tow tractor (TOW), which can provide a push-back to the airplane from the gate. Next to it is the ground-power unit (GPU), which provides electricity to the airplane when engines and APU are off. Next to the GPU, we have the first catering truck. This truck has a lifted van body that connects to one of the emergency exits. The front galley is supplied from this van. Next to it is the lower-deck cargo lift (LD CL), which can lift a Unit Load Device (ULD) to the level of the lower deck. A ULD is a cargo container with standardized dimensions. We also see a dolly train that brings the ULDs to the airplane. The fuel truck interfaces with the wing, but behind the wing, we see another cargo lift and a conveyor belt (CB) for bulk cargo, i.e., cargo that is not stored in a ULD. Next to it, another catering truck that services the aft galley is present. At the tail, we find the lavatory vehicle (LV) that dispenses the wastewater. Finally, we have the potable water vehicle (WV) that provides fresh water ahead of every flight. What is not shown is a cleaning truck, which can be connected at one of the exits to allow a cleaning crew and their equipment to enter the cabin.

In the conceptual design of the fuselage, we include the location of all emergency exits. The positioning of these exits can, therefore, be influenced by the accessibility requirements imposed by the ground servicing vehicles. Producing a drawing like the example in Figure 6.10 can help in deciding where the various emergency exits, as well as cargo doors, need to be positioned. In Section 6.5, we show how the emergency exits can be positioned.

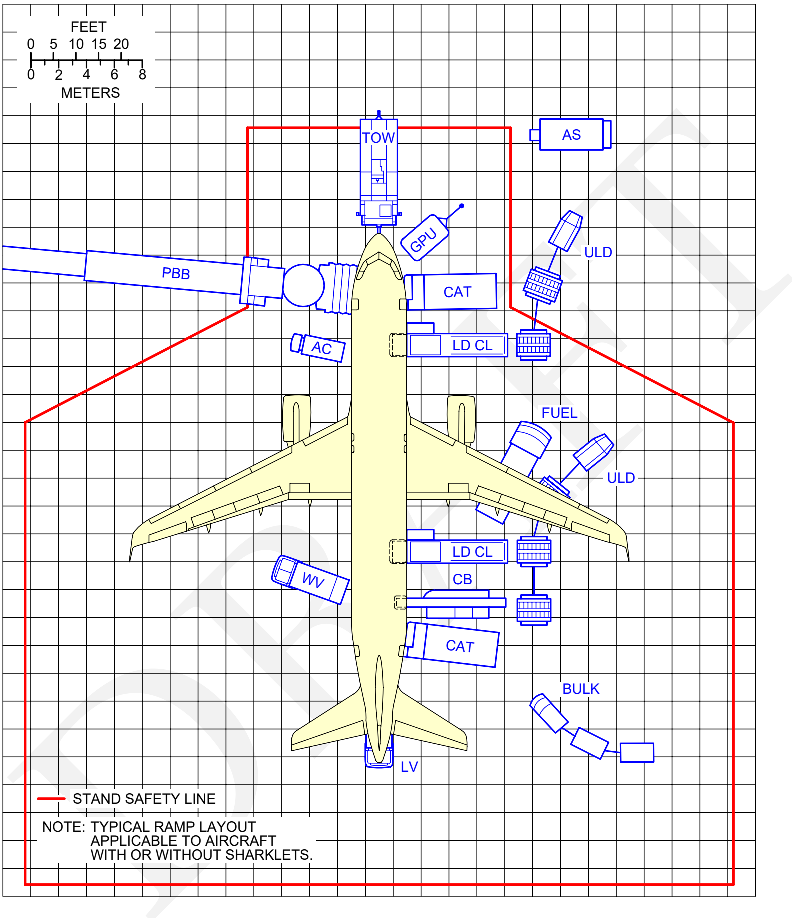


Figure 6.10: Layout of servicing vehicles at the airport stand for a large passenger airplane [1].

## 6.2. PRELIMINARY PAYLOAD VOLUME ANALYSIS

The total payload of an airplane consists of goods and/or passengers. The first question that we ask ourselves is: where is the payload stored in the fuselage? If you have passengers, luggage, and cargo, you have to decide to have that all on the same floor, to store (part of) the luggage and cargo below the floor in a separate cargo compartment, or to use separate cargo holds ahead and/or behind the passenger cabin. It is instructive to look at how cargo and luggage are stored on your reference airplanes. This could help you in deciding what works best for your design. To help you decide, it is instructive to estimate the total cargo volume in relation to the total passenger volume.

**Example 6.1**

Consider the airplane of Example 5.9 of the four-person twin-prop airplane. We have a payload mass of 400 kg for four passengers. We make the following crude assessment of the required volume for luggage and passengers:

| | $m$ (kg) | $\rho$ kg/m$^3$ | $V$ (m$^3$) |
|---|---|---|---|
| passengers | 320 | 110 | 2.9 |
| luggage | 80 | 170 | 0.5 |
| payload | 400 | - | 3.4 |

Here, we have implicitly assumed that each passenger weighs 80 kg and requires 0.7 m$^3$ of volume. So we conclude that 85% of the payload volume is taken by the passengers, and we only need 15% of the payload volume for the luggage. If we examine how the reference airplanes of Example 5.9 provide storage for luggage, we can see that cargo volumes are made ahead or behind the passenger cabin. For now, we have decided to store the luggage behind the passenger cabin.

In the previous example, it has been decided to position the luggage behind the passengers. Therefore, the luggage dimensions do not play a role when defining the cabin cross-section. However, in the next example, we show that for larger airplanes, it makes sense to store the luggage and cargo below the passenger floor.

**Example 6.2**

Consider a large transport jet. The design payload is 19 tonnes, corresponding to 180 passengers in economy class. We assume that each passenger weighs 80 kg, has 5 kg of carry-on luggage, and 20 kg of cargo-hold luggage. In addition, the maximal structural payload is 23 tonnes. This implies that an additional 4000 kg of cargo can be added, for which volume should be available. We make the following crude assessment of the volume that is required:

| | $m$ (t) | $\rho$ kg/m$^3$ | $V$ (m$^3$) |
|---|---|---|---|
| payload | 23 | - | - |
| passengers | 15 | 110 | 130 |
| carry-on luggage | 0.90 | 150 | 6.0 |
| cargo-hold luggage | 3.6 | 170 | 21 |
| cargo | 4.0 | 160 | 25 |

This crude assessment indicates that we need approximately 46 $m^3$ for cargo and luggage, against 136 $m^3$ for passengers with their carry-on luggage. If we decide to position the cargo holds ahead and/or behind the passengers, we would end up with a fuselage that is roughly 33% longer compared to a fuselage with only passengers on the main deck. Therefore, we decide to add a cargo compartment below the passenger floor. This is also in line with reference airplanes with a similar mission.

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The previous two examples have shown how a crude preliminary estimation of the required volume can be deduced from the top-level requirements. To that extent, we have used assumed values for the density of cargo and luggage. These values are only indicative but should be confirmed later on. In practice, you might be able to find more explicit requirements regarding the volume that is desired. For example, you could have the requirement that every passenger should be able to take a trolley suitcase into the cabin and store it in the overhead bins. Based on such a requirement, you can directly compute how much volume is required in the overhead bins.

Luggage and cargo can be stored in an airplane in two ways: in bulk or in standardized containers or pallets. If you choose to load luggage in bulk, you can effectively use the available cargo-hold volume. In other words, the available cargo-hold volume is equal to the required cargo-hold volume based on the sum of luggage and cargo volume. However, loading bulk luggage is a laborious process, which can be quite straining for the ground-handling crew due to the constrained space they need to work in. Therefore, you can also choose to load luggage and cargo in standardized containers or pallets. These containers are termed *unit load devices* or *ULDs*. A luggage container can be filled with luggage inside the airport terminal and can be loaded onto the airplane relatively quickly. The downside of using ULDs for luggage is their added weight. This results in higher fuel consumption than loading luggage and cargo in bulk. Small airplanes up to single-aisle transport airplanes typically have bulk loading, while all twin-aisle transports have containers. Some single-aisle airplanes are offered with either option. Then, it is up to the airline to decide whether they prefer bulk loading or containers. An overview of the weights and dimensions of ULDs used in civil aviation can be found on wikipedia.org/ULD. Figure 6.11 shows how bulk cargo and ULDs can be stored below deck. Note the use of nets for bulk cargo to prevent it from shifting during flight.

### ASSIGNMENT 6.1

In this assignment, you will decide on the distribution of your payload within the fuselage.

a State how your reference airplanes store cargo and luggage.
b State the design payload mass and maximum payload mass (if applicable).
c Assume a mass per passenger and compute the total passenger mass excluding luggage.
d Assume the average carry-on luggage mass per passenger and compute the total carry-on luggage mass.
e Assume the average cargo-hold luggage mass per passenger and compute the total cargo-hold luggage mass.

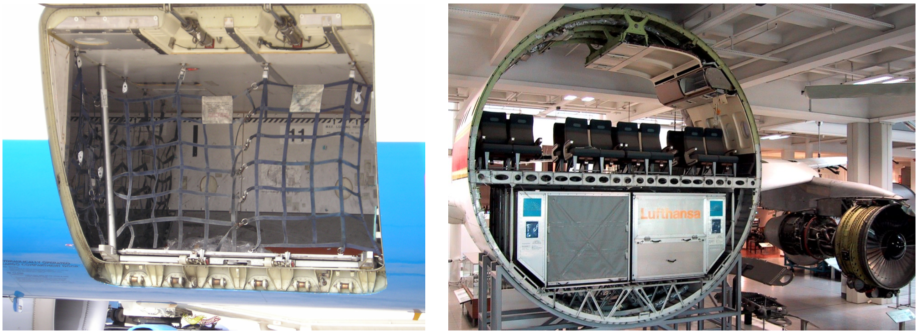


Figure 6.11: Examples of cargo holds below the passenger floor for bulk cargo (left) and unit load devices (right). Photos by User:Dtom ⓒⓘⓢ and Asiir ⓒⓘⓢ, respectively.

- f Compute the remaining cargo mass for the design payload mass.
- g Compute the remaining cargo mass for the maximum structural payload mass. Assume the same number of passengers and luggage mass as above.
- h Make assumptions on the volume per passenger, the density of luggage, and the density of cargo. State these assumptions.
- i Compute the approximate volume for passengers, carry-on luggage, cargo-hold luggage, and cargo.
- j Decide how you distribute the payload within your fuselage. Explain why you choose this distribution.
- k Decide whether to store luggage and cargo in bulk and/or in unit load devices. Motivate your decision.

## 6.3. DESIGN OF THE FUSELAGE CROSS-SECTION

In this section, we commence the design of the fuselage by designing the two-dimensional cross-section. We start with the payload requirement: the cross-section should be large enough to fit a cross-section of the payload volume. Secondly, we state the design objective: to minimize the length of the cross-sectional perimeter. This objective implicitly minimizes the friction drag of the fuselage as well as the weight.

To understand what the cross-section of the payload volume looks like, we first need to establish the dimensions of our payload volume. For a passenger airplane, we need to decide how many people are sitting abreast in the airplane. For small airplanes, this choice can be straightforward: 2-abreast is the obvious choice. However, for larger airplanes, it might be less obvious. How many seats abreast do you choose for a 100-passenger airplane? Or for a 300-passenger airplane? The following formula can provide guidance to make that decision:

$$n_{SA} = 0.45\sqrt{n_{pax}} \tag{6.5}$$

where $n_{SA}$ is the number of seats abreast and $n_{pax}$ is the number of passengers in the airplane. Note that you need to round the result of (6.5) to an integer! You can choose to round up, which results in a wider but shorter fuselage. Or you can choose to round down, which results in a narrower but longer fuselage. For airplanes where passengers do not have direct access to an emergency exit, an aisle is required. The minimum dimensions of the aisle are specified in CS/FAR 23/25 and are printed in Table 6.2. You can see that depending on the number of passengers, the requirements change. An airplane with more passengers requires a wider aisle. We use $h$ as a measure for the vertical distance from the floor. The aisle width below the armrest height (i.e., $h < 64$ cm) may be narrower than above the armrest ($h \geq 64$ cm). As a designer, you have the freedom to choose a larger aisle width to improve the boarding process as well as the level of comfort experienced by the passengers. You need to trade this against the resulting increase in weight and drag.

Table 6.2: Required aisle dimensions according to CS/FAR 23.815 and CS/FAR 25.815. Some conditions apply.

| | distance from floor | |
|---|---|---|
| $n_{\text{pax}}$ | $h < 64$ cm | $h \geq 64$ cm |
| $\leq 10$ | 30 | 38 |
| 11-19 | 30 | 51 |
| $\geq 20$ | 38 | 51 |

If we know how many seats are positioned abreast and the width of the aisle, we can start to construct a set of rectangular boxes that bound the cross-sectional area of the passenger volume. This is notionally shown in Figure 6.12, where various dimensions are defined, such as the width of the seat cushion ($w_{\text{seat}}$), the shoulder height ($h_{\text{shoulder}}$) and the armrest width ($w_{\text{armrest}}$). The value of dimensions are neither requirements nor assumptions. You have to choose a value for each of them based on available data on seats as well as common sense. Table 6.3 shows an example of seat dimensions and pitches for three different passenger airplanes.

Table 6.3: Example of seat dimensions and pitches for three different airplanes belonging to the same airline. Data from seatguru.com, retrieved on December 12, 2022. Using common airline fair basis codes: J - Business Class, W - Premium Economy, Y - Economy/Coach.

| | Short range | | | Medium range | | | Long range | | |
|---|---|---|---|---|---|---|---|---|---|
| Class | J | W | Y | J | W | Y | J | W | Y |
| Seat pitch (cm) | 81 | 81 | 76 | 84 | 84 | 76 | 107 | 89 | 79 |
| Seat width (cm) | 43 | 43 | 43 | 43 | 43 | 43 | 51 | 44 | 44 |
| armrest (cm) | 5 | 5 | 5 | 5 | 5 | 5 | 8 | 5 | 5 |

Having chosen a value for each dimension, we define the width of the cabin as follows:

$$w_{\text{cabin}} = n_{\text{SA}}\, w_{\text{seat}} + (n_{\text{SA}} + n_{\text{aisle}} + 1)\, w_{\text{armrest}} + n_{\text{aisle}}\, w_{\text{aisle}} + 2\, w_{\text{clearance}} \tag{6.6}$$

where $n_{\text{SA}}$ is the number of seats abreast and $n_{\text{aisle}}$ is the number of aisles. When we have no more than six abreast, we may have a single aisle. When we have between 7 and 12 abreast, we need two aisles (according to CS/FAR 25.817). We have chosen to measure the aisle width between the two armrests, assuming they are below 64 cm from the floor, i.e. $h_{\text{armrest}} < 64$ cm. The clearance width ($w_{\text{clearance}}$) has been added to allow for some space between the armrest and the inner cabin wall. The first rectangle to be enclosed by the fuselage inner perimeter, therefore, has a width of $w_{\text{cabin}}$ and a height of $h_{\text{shoulder}}$. The second box is the one around the aisle, which has a height of $h_{\text{aisle}}$ and a width of $w_{\text{aisle}}$. The third box ensures that the feet of the passengers stay within the inner wall. It has a width of:

$$w_{\text{floor}} = w_{\text{cabin}} - 2\left(w_{\text{armrest}} + w_{\text{clearance}}\right) \tag{6.7}$$

The final box ensures that there is enough head space for the passengers. It has its corner points at a height of $h_{\text{headroom}}$ above the floor and a width of:

$$w_{\text{headroom}} = w_{\text{floor}} - w_{\text{seat}} \tag{6.8}$$

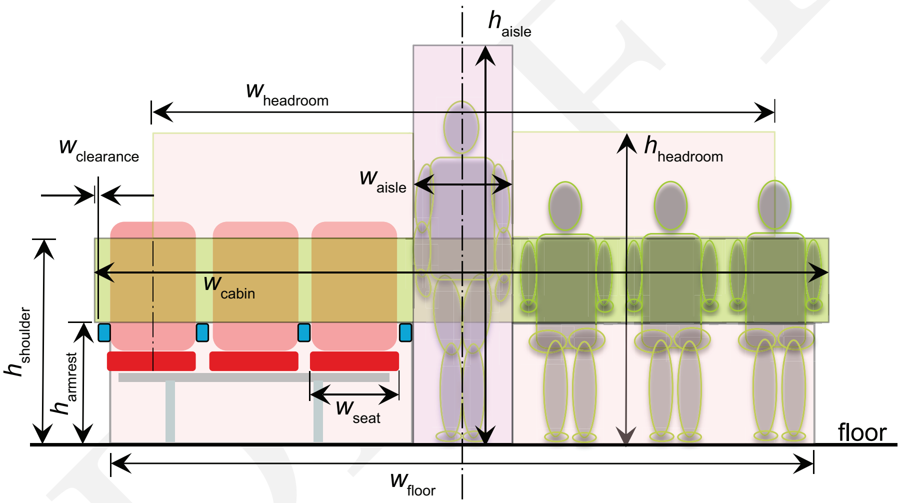


Figure 6.12: Definition of dimensions in the cabin cross-section.

Now that we have defined the bounding box for the passengers on the floor, it is time to turn our attention to the volume below the floor. If there is no under-floor volume required for cargo or luggage, then this step can be skipped. If you choose to have bulk cargo under the floor, it is advised to select a minimum distance between the passenger floor and the cargo floor so that the ground-handling crew has enough space to move and distribute the luggage. If you choose to use a ULD for luggage and cargo, the dimensions of the ULD should be included in the cross-section.

In the following examples, we will show how to construct a cross-section for various airplane types. To structure this process, we use the following sequence:

*Step 1* Compute dimensions of the passenger cabin

*Step 2* Chose a scale for your drawing

*Step 3* Draw upper-deck space allocation

*Step 4* Add floor structure depth

*Step 5* Draw cargo deck space allocation

*Step 6* Draw smallest circle(s)[1] around corners and measure the inner diameter, $d_{\text{f, inner}}$.

*Step 7* Compute $d_{\text{f, outer}}$ and draw outer perimeter.

*Step 8* Add cargo floor and overhead-bin space

**Example 6.3**

In this example, we compute the cross-section of the four-seat airplane of Example 6.1. Prior to performing the design steps of the cross-section, we perform a preliminary experimental analysis on how we organize the cabin and what dimensions we need to accommodate passengers comfortably within the cross-section. We decide to let two passengers sit side-by-side without the presence of an aisle. Instead, we decide to have 5 cm of clearance between the two passengers. We assume that the people are sitting with their knees slightly bent to reduce the necessary height of the cabin. We chose a seat that has a width of 45 cm and an armrest for each person measuring 6 cm. Finally, we assume a clearance of 2 cm between the armrest and the inner cabin wall. We choose the armrest height to be 50 cm above the floor, a shoulder height of 100 cm, and a headroom height of 135 cm.

*Step 1* With the dimensions mentioned above and using $w_{\text{aisle}} = 5$ cm, we can use (6.6) to find that $w_{\text{cabin}} = 123$ cm. The floor width is computed with (6.7) and is $w_{\text{floor}} = 107$ cm. The headroom width is computed with (6.8) and amounts to $w_{\text{headroom}} = 62$ cm. The armrest height, shoulder height, and headroom height have dimensions $h_{\text{armrest}} = 59$ cm, $h_{\text{shoulder}} = 100$ cm, and $h_{\text{headroom}} = 135$ cm, respectively.

*Step 2* Based on $w_{\text{floor}}$ and $h_{\text{headroom}}$ as well as the dimensions of our paper, we select a scale for our drawing. Here, we assume the available width and height on our paper are $w_{\text{paper}} = 18$ cm and $h_{\text{paper}} = 28$ cm, respectively. We decide to use a 1:10 scale to ensure an easy conversion of the dimensions while having a large enough drawing to show design detail. With this scale, the dimensions are drawn in the illustration below.

*Step 3* Based on the dimensions of step 1, we draw the two-dimensional space allocation using rectangular blocks (see below).

[1] in case of a double-bubble fuselage, choose two circles