



Chapter 1

INTRODUCTION

Today, liquid propellant rocket engines are commonly applied in the space segment all over the world. The idea of liquid propellant rockets was conceived over 100 years ago by pioneers as Konstantin Ziolkowski and Hermann Oberth. The American professor, physicist and inventor Robert H. Goddard is credited with developing and building the world's first liquid propellant rocket. He started testing the first liquid-fueled rocket engine in 1923 and after further developments and experiments, he launched the first liquid propellant rocket with gasoline and liquid oxygen as propellants in 1926 in Auburn, Massachusetts. In Europe, Johannes Winkler successfully launched the first liquid-fueled rocket in 1931 on the parade-ground near Dessau-Großkühnau in Germany, using the propellant combination liquid oxygen and liquid methane. In 1942, the first large, flight controlled and stabilized rocket Aggregat 4, later called V2, lifted off in Peenemünde, Germany, using an alcohol-water mixture and liquid oxygen as propellants [94, 98].

The technology of liquid propellant rocket engines is much more refined today. Various propellants were developed and tested and the liquid propellant combination with the highest specific impulse is liquid oxygen (LOX) and liquid hydrogen (LH₂). It is the most famous cryogenic propellant combination for launcher applications and focus of this study. LOX was the earliest, cheapest, safest and eventually the preferred oxidizer for large launch vehicles. Its main drawback is that it is moderately cryogenic, which means that it has to be stored at very low temperatures in order to keep it liquefied. LH₂ was identified by all the leading rocket visionaries as the theoretically ideal rocket fuel. However, its big drawbacks are that it is highly cryogenic and has a very low density, requiring large tanks.

In the mid-1950s, the USA mastered hydrogen technology and developed the world's first cryogenic launcher upper stage, the Centaur. Relevant cryogenic launcher stages which were consequently developed in the USA were the second and third stage of the Saturn V launcher, which were using LOX and LH₂ as propellant combination and enabled a manned landing



on the Moon. Furthermore, the NASA Space Transportation System, better known as the Space Shuttle, used LOX and LH₂, stored in the external tank, for the operation of the orbiter main engines. For the first three flights of the Indian launcher GSLV, the 12KRB cryogenic third stage from a Russian manufacturer was used. This stage is based on the KVRB upper stage, which is currently under development for the Russian launchers Proton and Angara. In 2010, the first flight test of the indigenous designed and built Indian cryogenic upper stage failed, however in January 2014 it was launched successfully. Further cryogenic stages, which already came into operation are the first stage of the Russian Energiya launcher, the first and second stage of the Japanese H-II launcher and the third stage of the Chinese launcher Long March 3 (also called CZ-3). For the new generation of the Long March rocket family Long March 5 to 7 a cryogenic upper stage is currently under development.

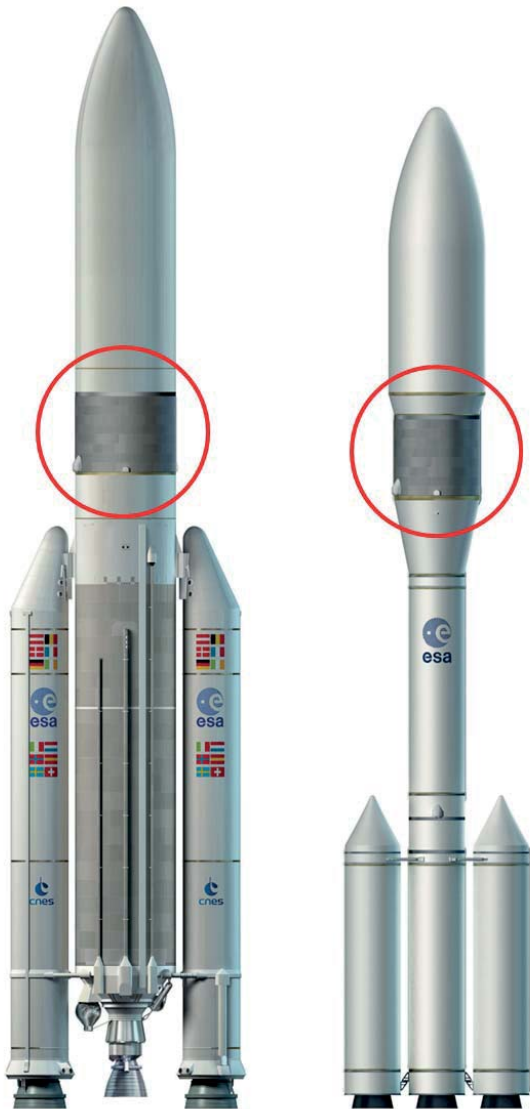


Fig. 1.1 Proposal for an Adapted Ariane 5 ME (left) and proposal for Ariane 6 (right) [41]. The red circles indicate the cryogenic upper stages.

The first development of new technologies for a cryogenic launcher stage in Europe began in 1967 under the name of LH₂ Experimental Program, coordinated by the European Launcher Development Organization (ELDO). It was experimental work as preparation for the development of the new Europa 3 launcher with its LOX/LH₂ second stage. However in 1972, the European Space Conference decided to cease of work on the Europa 3 launcher in favor of the French proposal L3S (Lanceur 3^{ième} Génération Substitution). This launcher concept envisaged a LOX/LH₂ third stage, called H6, which was much smaller than the Europa 3 upper stage concept. In 1973, the L3S concept was improved and the launcher name was changed to Ariane [102]. Six years later, on the 24th of December 1979, the first European launcher Ariane 1 lifted off the spaceport in Kourou in French Guiana. The Ari-

ane 1, used the first HM-7 engine using LOX and LH2. This upper stage concept was applied with some modifications for the Ariane 2, Ariane 3 and Ariane 4 launchers [61]. The current Ariane 5 launcher can use either the cryogenic upper stage ESC-A (Étage Supérieur Cryotechnique Type A) with LOX and LH2 as propellants, or the re-ignitable upper stage EPS (Étage à Propergols Stockables) with the storable propellants NTO and MMH. The main stage EPC (Étage Principal Cryotechnique) of the Ariane 5 launcher also uses the cryogenic propellants LOX and LH2.

Since 2008 ESA member states have been developing an improved type of the current launcher Ariane 5, named Adapted Ariane 5 ME (Mid-life Evolution), schematically depicted in Figure 1.1. This launcher is intended to have an increased payload capacity which should be enabled, amongst others, by a new cryogenic upper stage. This stage should be propelled with approximately 28 tons of LOX and LH2 and use the Vinci engine, which is currently under development. In November 2012, the ESA ministerial council committed to the continuation of the Adapted Ariane 5 ME program for another two years. They also initiated detailed definition studies of another new launcher concept, called Ariane 6 (schematically depicted in Figure 1.1). Further decision on the future of both launcher concepts shall be taken at the end of 2014. Today's baseline is that both, the Adapted Ariane 5 ME and the Ariane 6 concept, shall each include a newly developed cryogenic upper stage, based on the restartable cryogenic engine Vinci (positions marked in Figure 1.1 by the red circles) [87]. This baseline is significant for Germany and especially for the Bremen site, which is a prominent player in upper stage research and development.

Against the background of the new development of European cryogenic upper stages for the new launcher generation, interest in the advancement of cryogenic fluid management technologies has been significantly revived. Included in cryogenic fluid management technologies is the propellant feed system, the task of which is to feed the liquid propellant from the propellant tank to the rocket engine thrust chamber at pre-defined mass flow rates and required conditions, such as temperature and pressure. Two distinct types of propellant feed systems can be defined: the pressure-fed and the pump-fed systems. For a pressure-fed propellant feed systems, the engine thrust is directly proportional to the tank ullage pressure. Therefore, relatively high tank pressures of $1.3 \cdot 10^3$ kPa up to $9 \cdot 10^3$ kPa are required, as the propellant tank pressure needs to be higher than the pressure in the engine combustion chamber. In a pump-fed propellant feed system, lower tank pressures

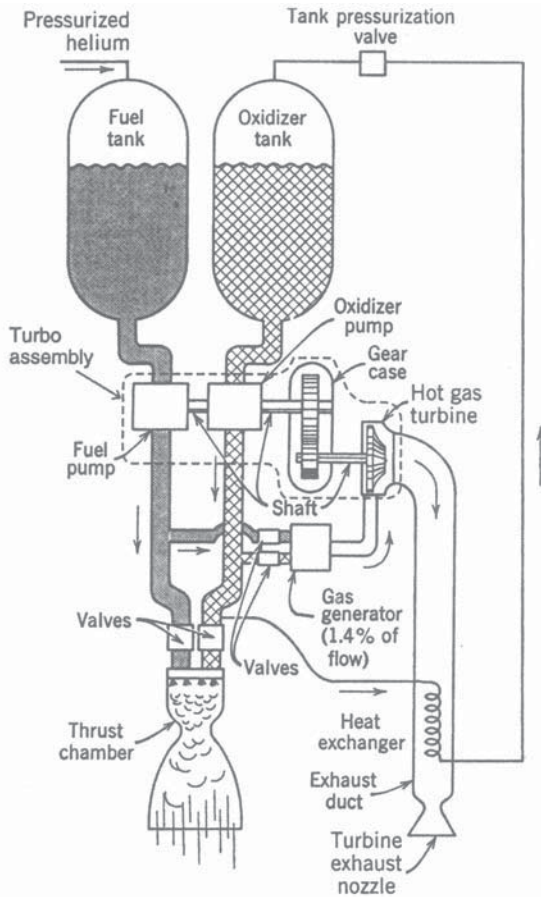


Fig. 1.2 Flow diagram of a liquid propellant rocket engine with stored-inert-gas and evaporated-propellant pressurization systems [94].

Commonly, three different tank pressurization systems are distinguished: the combustion-products, the evaporated-propellant and the stored-inert-gas system.

For the combustion-products system, the pressurant gas is obtained by combustion of propellants. E.g., a gas generator produces fuel-rich exhaust gas for the operation of the engine turbopump and at the outlet of the turbine, gas for the tank pressurization is tapped off, cooled down and fed back into the propellant tank. This system has been used only for few applications, as the pressurant gas is often chemically incompatible with the propellant, has a too high temperature or condensable elements.

The evaporated-propellant pressurization system appears in two versions: the simplest is the self-pressurization by the propellant's own boil-off vapor. This system in general requires a propellant with a high vapor pressure and the resulting tank pressure is very dependant on the mission profile. The system is very reliable, but requires high pressurant gas masses due to low pressurant gas temperatures and therefore high fluid densities. The second and most common form of the evaporated-propellant pressurization system is active-pressurization with

(between $0.07 \cdot 10^3$ kPa and $0.34 \cdot 10^3$ kPa) are used, as turbopumps are applied to raise the propellant pressure to a level above tank pressure, which is suitable for the injection into the engine combustion chamber [95].

One component of the pump-fed propellant feed system is the tank pressurization system, which forms the focus of this study. The purpose of the pressurization system is to control and maintain the required pressure in the propellant tanks at any time of the mission. The defined tank pressure history is bounded by propellant and tank structural requirements, the thrust profile and the net positive suction head (NPSH). The NPSH is the minimum turbopump suction head which is required to avoid cavitation throughout the operating cycles.



evaporated propellant. A part of the liquid propellant is tapped off at the pump discharge area and routed through a heat exchanger. Within this heat exchanger, the propellant is vaporized and then led as pressurant gas back into its own propellant tank [72]. Such system is schematically depicted in Figure 1.2 for the oxidizer propellant tank.

The third, widely used system for pressurizing a propellant tank is the stored-inert-gas system, as schematically depicted in Figure 1.2 for the fuel tank, where helium is applied as a pressurant gas. For the stored-inert-gas system, the applied pressurant gas is stored in external vessels, preferably at low temperature, resulting in high storage density. Pressurizing with an inert gas results in the fact, that during and after pressurization, the tank ullage includes both, the inert pressurant gas and a quantity of evaporated propellant.

This study focuses on both, active-pressurization with an evaporated-propellant system and a stored-inert-gas pressurization system, as they are intended to be applied for the future European cryogenic upper stage concepts.



Chapter 2

THEORETICAL BACKGROUND

This chapter provides an overview of the theoretical background of this study, such as the relevant fundamental equations of fluid dynamic and thermodynamics, heat transfer and phase change mechanisms as well as thermal stratification. It also includes the derivation of an analytical equation for the pressure drop and the scaling concept with relevant characteristic numbers.

Figure 2.1 displays a typical propellant tank, which is cylindrical with an ellipsoidal shaped bottom. It is partly filled with liquid propellant and pressurant gas is injected through an inlet in the tank lid. The z -axis points in the opposite direction to the gravity vector $\mathbf{g} = (0,0,-g)^T$ and the origin of the coordinate system is set to the bottom of the tank. Liquid and ullage is separated by a free surface and the normal vector, perpendicular to it, is set as \mathbf{n} , pointing in the positive z direction.

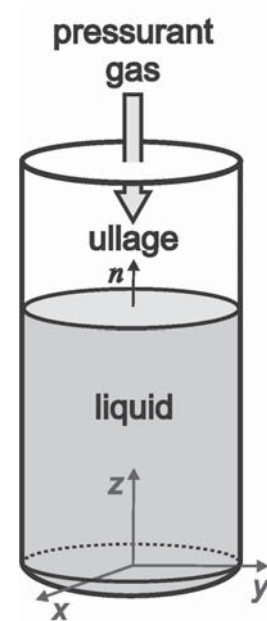


Fig. 2.1 Schematic of the considered cylindrical propellant tank with ellipsoidal shaped bottom, partly filled with liquid propellant. The pressurant gas is injected through an inlet in the lid into the tank ullage.

2.1 Ideal Gas

The thermodynamic properties of an ideal gas can be described by the ideal gas law

$$p V = n \bar{R} T \quad (2.1)$$

with p as the pressure, V the volume, n is the amount of substance of the gas, \bar{R} the universal gas constant and T is the temperature.

In a mixture of ideal gases, each gas has a partial pressure. The partial pressure p_i of one component is determined as

$$p_i = p \chi_i \quad (2.2)$$

with χ_i as the mole fraction of that gas component. The mole fraction is defined as

$$\chi_i = \frac{n_i}{n} \quad (2.3)$$

where n_i is the amount of substance of the gas component i and n is the total amount of substance. The mass fraction of an ideal gas mixture is defined as

$$\psi_i = \frac{m_i}{m} \quad (2.4)$$

where m_i is the mass of one gas component and m the total gas mass.

2.2 Conservation Equations

In order to describe a typical fluid dynamics problem, the mode of change over time of the system's physical variables needs to be determined. This is achieved by the Reynolds' transport theorem from which the basic conservation equations, the conservation of mass, the conservation of momentum and the conservation of energy are derived. In the following sections, these conservation equations are introduced together with the Reynolds' transport theorem, based on White [103] and Bird [13]. In accordance with the conventional definition, the presented governing equations describe a one-phase system with liquid, which is considered as a continuum with a free surface.

2.2.1 Reynolds' Transport Theorem

The Reynolds' transport theorem is used to characterize the dynamics of a physical variable and its change over time. It states that the total change over time of any fluid property G in a volume V that moves with the flow is equal to the change over time in the current control volume CV plus the flux of G passing through the instantaneous surface area A of the control volume. For \mathbf{n} being the outward normal unit vector on the control surface, the Reynolds' transport theorem can be written in the compact form as

$$\frac{d}{dt}(G_{sys}) = \frac{d}{dt} \left(\int_{CV} \gamma \rho dV \right) + \int_{CS} \gamma \rho (\mathbf{v} \cdot \mathbf{n}) dA \quad (2.5)$$

with CV being an arbitrary fixed control volume, CS the control surface, t the time, ρ the density $\gamma = dG/dm$ the intensive value (i.e. the amount of the arbitrary mass-specific parameter G in any small portion of the fluid) and with \mathbf{v} being the fluid velocity.

2.2.2 Conservation of Mass

The equation for the conservation of mass can be derived from the Reynolds' transport theorem (Equation 2.5). This conservation equation states that the mass of a closed system m_{sys} remains constant over time. For the integral form of the equation for conservation of mass, the variable G becomes the mass m . It follows $\gamma = dm/dm = 1$. The equation for a deformable control volume is therefore

$$\left(\frac{dm}{dt}\right)_{sys} = 0 = \frac{d}{dt} \left(\int_{CV} \rho dV \right) + \int_{CS} \rho(\mathbf{v}_r \cdot \mathbf{n}) dA \quad (2.6)$$

where the relative velocity \mathbf{v}_r is defined as $\mathbf{v}_r = \mathbf{v}(\mathbf{r}, t) - \mathbf{v}_{CV}(t)$ if the control volume is moving uniformly at velocity $\mathbf{v}_{CV}(t)$. The vector \mathbf{r} is the position vector and t is the time. As previously stated, CS is the control surface of the mass flux. For a fixed control volume, it follows that

$$\int_{CV} \frac{\partial \rho}{\partial t} dV + \int_{CS} \rho(\mathbf{v} \cdot \mathbf{n}) dA = 0. \quad (2.7)$$

If the control volume only has an i number of one-dimensional inlets and outlets, it can be written as:

$$\int_{CV} \frac{\partial \rho}{\partial t} dV + \sum_i (\rho_i A_i v_i)_{out} - \sum_i (\rho_i A_i v_i)_{in} = 0 \quad (2.8)$$

For incompressible flows, where $\frac{\partial \rho}{\partial t} \approx 0$, Equation 2.7 becomes

$$\int_{CS} \rho(\mathbf{v} \cdot \mathbf{n}) dA = 0. \quad (2.9)$$

The differential form of mass conservation for an infinitesimal fixed control volume (dx, dy, dz) follows from Equation 2.7. The considered volume element is so small that the volume integral reduces to a differential term.

$$\int_{CV} \frac{\partial \rho}{\partial t} dV \approx \frac{\partial \rho}{\partial t} dx dy dz \quad (2.10)$$

Equation 2.10 is the equation of mass conservation, which describes the rate of change of density at a fixed point resulting from the changes in the mass velocity vector $\rho\mathbf{v}$. It follows



that the compact form for the differential equation of mass conservation, written in the vector symbolism, is

$$\frac{\partial \rho}{\partial t} + \nabla \cdot (\rho \mathbf{v}) = 0 \quad (2.11)$$

where $\nabla \cdot (\rho \mathbf{v})$ is the divergence of the mass flux $\rho \mathbf{v}$. For incompressible flows this results in

$$\nabla \cdot \mathbf{v} = 0. \quad (2.12)$$

2.2.3 Conservation of Momentum

The law of conservation of momentum states that the total momentum of a system remains unchanged if the system is closed. If the surroundings exert a net force \mathbf{F} on the system, Newton's second law states that its mass will begin to accelerate.

$$\mathbf{F} = m\mathbf{a} = m \frac{d\mathbf{v}}{dt} = \frac{d}{dt}(m\mathbf{v}) \quad (2.13)$$

Consequently, it follows from the Reynolds' transport theorem (Equation 2.5), with $\mathbf{G} = m\mathbf{v}$ being the linear momentum and $\boldsymbol{\gamma} = d\mathbf{G}/dm = \mathbf{v}$

$$\frac{d}{dt}(m\mathbf{v})_{sys} = \sum \mathbf{F} = \frac{d}{dt} \left(\int_{CV} \mathbf{v} \rho dV \right) + \int_{CS} \mathbf{v} \rho (\mathbf{v}_r \cdot \mathbf{n}) dA \quad (2.14)$$

where the term \mathbf{v} is the fluid velocity relative to an inertial coordinate system and the term $\sum \mathbf{F}$ is the vector sum of all forces acting on the control volume. If the control volume only has an i number of one-dimensional inlets and outlets, Equation 2.14 reduces to the following form

$$\sum \mathbf{F} = \frac{d}{dt} \left(\int_{CV} \mathbf{v} \rho dV \right) + \sum (\dot{m}_i \mathbf{v}_i)_{out} - \sum (\dot{m}_i \mathbf{v}_i)_{in} \quad (2.15)$$

with \dot{m}_i as the mass flow rates.

The differential form of the equation of linear momentum is derived from Equation 2.15 with the assumption that one element is so small that the volume integral reduces to a derivative term.

$$\frac{\partial}{\partial t} \left(\int_{CV} \mathbf{v} \rho dV \right) \approx \frac{\partial}{\partial t} (\rho \mathbf{v}) dx dy dz \quad (2.16)$$

By considering gravity \mathbf{g} as the only body force and $\boldsymbol{\tau}_{ij}$ as the viscous-stress tensor acting on an element, the basic differential momentum equation for an infinitesimal element follows

$$\rho \left(\frac{\partial \mathbf{v}}{\partial t} + (\mathbf{v} \cdot \nabla) \mathbf{v} \right) = -\nabla p - (\nabla \cdot \boldsymbol{\tau}) + \rho \mathbf{g} \quad (2.17)$$