Preliminary design of a liquid propellant engine for a reusable sounding rocket

Y I Jenie, A C Asyary, and R E Poetro

Flight Physics Research Group, Faculty of Mechanical and Aerospace Engineering, Institut Teknologi Bandung. Jl. Ganeca no 10, Bandung, Jawa Barat, Indonesia

Abstract. Sounding rocket (or Roket Sonda in Indonesian), is a type of rocket that carries instruments to conduct some scientific experiments in suborbital altitudes. Since the use of sounding rockets is constantly increased, there has been an idea to make it reusable. The development of a reusable sounding rockets should includes at least six key technologies, which from those, the development of a proper liquid propellant engine, which thrust can be intermittently controlled, is mandatory. This paper aims to contribute in Indonesia’s own reusable sounding rocket (RSR) development, by presenting an example of preliminary calculation for its liquid-propellant engine. To achieve the intended launch/flight profile to a targeted 120 km maximum altitude, and return safely to the ground, several design parameter is chosen, including the total mass of propellant for the powered ascent and descent that need to be at least 8700 kg, with the mass rate of 60 kg/s, producing 189 kN force. The force is produced by four nozzles to also provide control, where the exit area can be determined to be 0.164 m². Finally, from these values, a rough estimation of the geometry of the reusable sounding rocket can be determined.

1. Introduction
Sounding rocket (or Roket Sonda in Indonesian), is a type of rocket that carries instruments to conduct some scientific experiments in suborbital altitudes, typically between 50 to 150 kilometers [1][2]. The experiments itself can be conducted during the rocket flight phase (ascent phase) or after the instruments are jettisoned and descent slowly with a parachute [1][2]. Since 2006, Indonesian Space Agency (LAPAN), has been routinely launching its own sounding rockets, either for technological dissemination, or for student’s competition [2][3]. While these are mainly small-scaled rockets with simple engines, LAPAN has also been active in researching and developing larger rocket with more advanced liquid propellant engines [4][5].

Since the use of sounding rockets is constantly increased, there has been an idea to make it reusable [1][6]. This idea aims to create a more economical environment for scientific purposed launches, by developing more comprehensive, but long lasting, rocket systems. The reusable feature also ensures the return of the payload, which nowadays consists of increasingly expensive scientific instruments. As introduced in [6], the development of a reusable sounding rockets should includes at least six key technologies, which are (1) aerodynamic design, (2) landing gears, (3) reusable engine, (4) fuel/oxidizer management, (5) reusable insulation for cryogenic tank, and (6) health management system. These imply the importance of a liquid propellant engine, which thrust can be intermittently controlled, instead of the usual solid one that most sounding rocket uses.

Liquid propellant engine, however, has not been the number one choice for sounding rockets, especially in Indonesia, since it is far more complex and costly to develop, compared to solids. Several
research for liquid-propellant engine in Indonesia has been conducted, mostly led by LAPAN, including [4] and [5]. These, however, are more focused on the single engine development to achieve a certain burning time, and have not considered the rocket design and mission. Other references, such as [6][7] discussed the used of liquid-propellant engine for a reusable sounding rocket, which also includes several demonstration of its capability. This approach of designing the engines, which is directly being related to the rocket mission, is more preferable for a faster and on-point engine development.

This paper aims to contribute in Indonesia’s own reusable sounding rocket development, by presenting an example of preliminary calculation for its liquid-propellant engine. This engine is designed for repeated operation that at least mirrors the performances of RVT#4, one of the rocket being launch for the development of reusable sounding rocket in ISAS [6] by JAXA. The calculation includes the propellant weight and volume; the nozzles exit area; flow characteristics; and other parameters in the engine systems. It will be shown later on in this paper that the calculation results have a high importance in the overall rocket sizing and configuration, which includes the dimensions of the body, fins, and nozzles.

This paper is organized as follows. After this introduction, the Section 2 will present the general design requirement and objective for the liquid-propellant engine, together with the design and typical mission of several existing reusable sounding rocket. Afterwards, the third section will elaborate a complete preliminary calculation of the engine and the overall results. Section 4 will discuss the possible rocket size and configuration using the results from previous chapter as the requirements. Finally, Section 5 will concludes this paper and gives several suggestion for future works.

2. Reusable sounding rocket overview

Between 1998 to 2003, reusable rocket has been extensively studied and experimented by JAXA in their Reusable Vehicle Testing Project [6]. The experiment includes four types of single stage rocket, designated by RVT #1 – RVT #4, where all of them are powered by liquid-propellant engines. From those four, the first three where launched three times, while RVT #4 was able to make its fifth launch. The success of the project gave a boost to the current reusable sounding rocket development project, hopefully to be fully operating sometime in this decade. In the meantime, several practical reusable rocket have been operating, including, perhaps the most famous ones, the first stage of Falcon 9 rocket (2015), and the booster stages of Falcon Heavy rocket (2018) [8][9][10]. While the intended mission of both Falcon 9 and Falcon Heavy are much more complex, their reusable stage’s mission profile is essentially the same as that of a typical sounding rocket.

![Figure 1. Example of existing reusable rockets](image)

2.1. Typical mission profile

The mission profile for a reusable sounding rocket is represented in figure 1, which depicts six phases of operation: (1) lift off, (2) powered ascent, (3) coasting (unpowered ascent), (4) flip maneuver, (5) free fall, (6) powered descent, and (6) landing. The highlighted lines indicates engine burn period, which are used for ascending, flip maneuver, and descending. The mission is extended, where instead of just coasting (and inevitably falling), an aerodynamic maneuver (flipping) is conducted to change...
the engine force line into a preferable direction. A controlled descent then can be performed, to slowly land the rocket at a designated area. Experiments, therefore, can be conducted longer, during the ascent as well as during the powered descent.

The mission profile drives the engine requirement for this reusable rocket. Firstly, a liquid propellant type of engine is needed for the intermittent operation, which cannot be conducted by a single burn-solid propellant engine. Secondly, to provide an efficient control by thrust vectoring, multiple engines (or thrust line) might be mandatory in conducting the flip manoeuvre. While aerodynamic control surfaces might also be sufficient to do the task, thrust vectoring is more commonly preferred for high load (g) operation. Nevertheless, aerodynamic fins are still considered for the rocket stability.

![Figure 2. Typical Reusable Sounding Rocket (RSR) launch/flight profile.](image)

2.2. Design requirements and objectives
The mission of the rocket, which, in this research, should at least mirrors the ISAS project’s RVT#4 [6] by JAXA, drives the design of the liquid engine. The baseline specification for the designed RSR in this research, compared to the RVT#4 specification is listed as follows [6]: “no spec.” means that the base values will be computed in this paper.

| Specifications                  | RVT #4                | RSR Design Point          |
|--------------------------------|-----------------------|---------------------------|
| Summit altitude                | 100 - 150 km         | > 120 km                  |
| Body length                    | 13.5 meter           | no spec.                  |
| Body Diameter                  | 3 meter              | no spec.                  |
| Take Off Mass                  | 10750 kg             | < 12000 kg                |
| Payload weight                 | 100kg                | > 100 kg                  |
| Dry Mass                       | 3814 kg              | < 3300 kg                 |
| Engine thrust                  | 164 kN               | no spec.                  |
| Engine ISP                     | 320 sec              | < 320 sec                 |
| Propellant Type                | liquid oxygen–liquid hydrogen |

The goal of this research is to define a preliminary specification of the liquid engine that can support the RSR design point in Table 1. The specification includes the mixture ratio, the combustion temperature, the propellant volume and flow-rate, nozzle throat and exit area, and the thrust required
to be produced by the engine. Furthermore, as already explained, the rocket will be using four engines, to provide a thrust vectoring control.

Engine specification, especially the nozzle throat and exit area, will be important for the rocket body design. These two parameters will determined the diameter of the body, which need to be large enough to fit four nozzles. The diameter, together with the propellant volume, will in turn determine the length of the body and other configurations.

3. Liquid propellant engine design
The goal of this chapter is to define the maximum thrust needed to achieve the flight profile in figure 2, and to determine the required volume (and weight) of the propellant. The calculation will considered two main powered phase, which are the powered ascent and the powered descent. The flip manoeuvre, while also requires propellant, is neglected here, since it will only accounted for a small ration of the total volume of propellant.

3.1. Powered and unpowered ascent requirements
Perhaps the easiest way to calculate the thrust and propellant needed, this paper start with calculation on the unpowered ascent phase, after the rocket reaches an initial coasting velocity, or its velocity on the end of its powered ascent, \( V_{pa} \) (see figure 3). By neglecting the aerodynamic and the change of gravitational acceleration, for a preliminary calculation of the height in the unpowered ascent \( (h_u-h_{pa}) \), related to the initial velocity can be determined by:

\[
(h_u - h_{pa}) = \frac{V_{pa}^2}{2g_u}
\]

(1)

To reach the designed maximum altitude (apogee), the rocket can choose a combination of altitude and speed, the \( h_{pa} \) and \( V_{ns} \), of which the unpowered ascent started with. The lower \( h_{pa} \), requires higher \( V_{pa} \), and vice versa. Figure 3 present the calculation to reach 150 km (the designed apogee), for the RSR in this research, derived from equation (1). Notice that his is derived with an assumption that the rocket is heading straight upward at all time, and the gravity acceleration is considered constant. The dashed line in figure 3 marks the 50% altitude from, the targeted 150 km apogee. Commonly, the unpowered ascent altitude gain preferably is higher than the powered one, and therefore, in a sounding rocket design, the \( h_{pa} \) is selected lower than the 50% line.

![Figure 3. RSR unpowered/powered ascent and the altitude-speed combination.](image-url)
The problem from here is shifted to how the rocket can reach the initial unpowered ascent (free coasting) velocity $V_{pa}$, from zero velocity at launch. This is the powered ascent phase, in which the rocket burns the propellant for the first time. A modified Tsiolkowsky’s equation with the additional gravity pull, assumed to work only on the rocket longitudinal axis, is defined in equation (2). Notice that equation (3) neglects the aerodynamic effect, due to its difficulty to be represented in a simple equation. The effect will be reconsidered latter on when numerical method is used, in a more detailed design phase.

$$
\Delta V = V_{pa} = I_{SP}g_o \ln \frac{M_{oa}}{M_{oa} - M_{fa}} - g_o t_{pa}
$$

Equation (2) provides a relationship between the propellant mass and how fast it needs to burn ($t$), to achieve a specific $\Delta V$. In this case, the $\Delta V$ is the initial speed $V_{pa}$ in the coasting phase. From equation (3), the height of the powered ascent ($h_{pa}$) can be determined by integration, as presented in equation (4).

$$
h_{pa} = I_{SP}g_o \left( \frac{M_{oa} - M_{fa}}{M_{fa}} \ln \frac{M_{oa} - M_{fa}}{M_{oa}} + 1 \right) t_{pa} - \frac{g_o}{2} t_{pa}^2
$$

Using equation (3), one can plot the combination of propellant mass ($M_{fa}$) and the elapsed time ($t_{pa}$), to reach a certain speed, $V_{pa}$. While using equation (4), a similar plot $M_{fa} - t_{pa}$ plot can be produced, which shows how can the rocket reaches a certain altitude, $h_{pa}$. These two plots are presented simultaneously in figure 4, from the calculation of the combination of $M_{fa} - t_{pa}$ to reach a certain altitudes, and certain velocities. These altitudes and velocities are taken from the requirement to reach the apogee of 120 km, as presented before in figure 3.

![Figure 4. RSR ascent phase propellant mass requirements.](image)

Intersections in the figure (with diamond marking) mark the combination of propellant mass and time that can give a specific altitude with a specific velocity, required for the rocket to complete its apogee of exactly 120 km with the unpowered ascent. If one draws a line connecting these intersections, then a boundary of which combination of propellant mass and elapsed time that resides on the upper side is guaranteed to reach the designed apogee.
Therefore, figure 4 can be used to decide which combination of $h_{pa}$ and $V_{pa}$ should the RSR have. In this research the point marked with a star is selected as the design point, i.e., the required propellant mass, $M_{fa}$, is 6000 kg, that need to burn in $t_{pa} = 100$ seconds. This combination will set the $h_{pa}$ to 47.26 km, with the speed of $V_{pa} = 1.19$ km/s, and reaches the apogee of 120.01 km. Note that these result is calculated with assumption of engine’s I_{SP} is 320 second, and with constant gravitational acceleration of 9.80665 m/s$^2$.

Based on the design points explained before, the propellant weight should not exceed 7,900 kg, and hence, the selected propellant mass fulfils this, at a propellant weight for ascending will spend 76% of the total propellant. The mass rate required can also be calculated by dividing the required propellant mass with the time needed, which will be 60 kg/second. From these values, together with the assumed I_{sp} of 320, based on equation (6) the thrust that is needed for the powered ascent should be approximately 189 kN.

3.2. Unpowered and powered descent requirements

The unpowered descent (free fall) and powered descent profile determines the additional propellant requirements. The result, however, will not necessarily affected the rocket required thrust force, since it is can be less than it is in the ascent phase. The calculation also assumed no aerodynamics factor, which in reality will affect the descent phase significantly. Hopefully, when aerodynamic is taken as a factor, the propellant requirement for powered descent will be less.

![Figure 5. RSR descent phase.](image)

The calculation is similar to the ascent phase, and without any aerodynamics, the speed/altitude profile is exactly the same as in figure 3. Similar plot as in figure 4 can also be produced for the descent case, as shown in figure 6. Here, there are two differences in the calculation. First one is the initial mass, which will be 6,000 kg, from the total initial 12,000 kg minus the propellant used for ascent, 6,000 kg. The second difference is the apogee height, which should be the calculated apogee from the ascent phase, i.e. 120.01 km.

In this research, it is selected that for powered descent, the propellant mass for powered descent should be 2,700 kg, which should be burned within 45 second. Notice the same mass rate as the ascent phase, that is, 60 kg/s. The force in this phase will also be 189 kN.
3.3. Nozzles area requirement

From previous calculations, the required thrust the rocket engine need to produce is 189 kN, with burning time of 100 second. Since there will be four nozzles in the rocket, each engine nozzle should be able to produce 47.25 kN. With additional assumption of optimum operation at altitude where the pressure is 14.7 psia (1 atm), and the chamber pressure of 1000 psia, the theoretical combustion chamber performances of the liquid-propellant (liquid oxygen–liquid hydrogen) engine can be determined. These calculation is based on the calculation method in [11][12], which resulting parameters are explained in the following paragraphs of this section, and are summarized in table 2.

The theoretical combustion chamber performances includes the mixture ratio of 3.40 for the fuel/oxidizer, the mean molecular mass of exhaust products, M, of 0.0089 kg/mol, the combustion chamber temperature, T1, of 2959 K, or 2685.9°C, and specific heat ratio, k, of 1.26. For further calculation, it is also assumed that the actual specific impulse is 97% of theoretical (0.97 x 320 sec) and that the thrust coefficient, C_F, is 98% of the ideal value.

The exhaust velocity, \( v_e \), for an optimum nozzle can be determined afterward using equation (4) below, with the parameters as described in figure 7. It is assumed that the atmosphere is having the
ideal gas properties, in which R is the ideal gas constant, 8.314 J/K·mol. From this, the exhaust velocity is 4.232.58 meter/second.

\[
v_2 = \sqrt{2k \over k-1} RT \left[ 1 - \left( \frac{p_2}{p_1} \right)^{(k-1)/k} \right]
\] (4)

The optimum area ratio can be found from equation (5), where \( p_2 \) is assumed to be the same as \( p_3 \). As the result, the optimum area ratio for the nozzle design is 42.

\[
\frac{A_1}{A_y} = \frac{V_y u_y}{V_2 u_2} = \frac{k+1}{2} \left( \frac{p_y}{p_1} \right)^{(k-1)/k} \sqrt{\frac{k+1}{k-1}} \left[ 1 - \left( \frac{p_2}{p_1} \right)^{(k-1)/k} \right]
\] (5)

The theoretical or ideal thrust coefficient can be found from equation (6), as well as in the graph in [111]. Here, \( p_2 \) assumed equal to \( p_3 \). As the result, the pressure ratio in this nozzle design is 33, and the ideal thrust coefficient \( C_F \) is 1.76. The actual thrust coefficient is usually slightly less, which can be assumed to be 98% from the ideal value. Hence, for the nozzle design, the actual \( C_F \) is 1.72.

\[
C_F = \frac{F}{p_1 A_1} = \frac{u_2^2 A_2}{p_1 A_1 V_2} + \frac{p_2 A_1}{p_1 A_2} - \frac{p_3 A_2}{p_1 A_1}
= \frac{2k^2}{k-1} \left( \frac{2}{k+1} \right)^{(k+1)/(k-1)} \left[ 1 - \left( \frac{p_2}{p_1} \right)^{(k-1)/k} \right] + \frac{p_2 - p_3 A_2}{p_1 A_1}
\] (6)

The throat area required can be derived from equation (7), which is calculated to be \( 3.97 \times 10^{-3} \) m\(^2\). Finally, with the calculated optimum area ratio of 42, the nozzle exit area is calculated with equation (8), with the result of \( 1.64 \times 10^{-1} \) m\(^2\).

\[
A_y = \frac{F}{C_F p_1}
\] (7)

\[
A_2 = A_1 \times 42
\] (8)

4. Rocket configuration consideration
Overall, the result for this nozzle design and propellant requirements, elaborated in the previous section, are tabulated in the following table 2. This section will use these result to produce a preliminary configuration of the RSR, especially its dimensions.

| Parameters                                                  | Value       |
|-------------------------------------------------------------|-------------|
| Propellant Mass required                                     | 8700 kg     |
| Mass rate                                                   | 60 kg/s     |
| Operation Time (Powered Ascent and Descent)                 | 145 second  |
| Number of Nozzles                                           | 4           |

Table 2. Propellant and nozzles design parameters and result.
| **Parameter**                  | **Value**          |
|-------------------------------|--------------------|
| Thrust Required               | 47.25 kN           |
| Propellant                    | liquid oxygen–liquid hydrogen |
| Mixture Ratio                 | 3.4                |
| Hydrogen Mass                 | 6722.7 kg          |
| Hydrogen Volume               | 2.68 m³            |
| Oxygen Mass                   | 1977.3 kg          |
| Oxygen Volume                 | 0.05 m³            |
| Combustion Chamber Pressure   | 1000 psia          |
| Combustion Chamber Temperature| 2959 K             |
| Exhaust Velocity              | 4,232.58 m/s       |
| Pressure Ratio                | 33                 |
| Thrust Coefficient            | 1.72               |
| Optimum Area Ratio            | 42                 |
| Nozzle Throat Area            | $3.97 \times 10^{-3}$ m² |
| Nozzle Exit Area              | $1.64 \times 10^{-1}$ m² |

### 4.1. Rocket base area

Similar to a solid propellant rocket, the base area of a liquid propellant rocket is usually determined by the requirements from the nozzles dimensions. In previous section, the nozzle requirement have been established, that is, four nozzles where each of them has the throat area of $3.97 \times 10^{-3}$ m², and the exit area of $1.64 \times 10^{-1}$ m². These mean that the diameters of those area should be $3.55 \times 10^{-2}$ m, and $2.28 \times 10^{-1}$ m, respectively. Therefore, assuming a round base, the base area of the RSR can be designed as in the following figure 8.

![Figure 8. Design of the reusable sounding rocket base area.](image)

As can be observed in figure 8, to make the nozzles fit, a diameter of 2.38 meters is required. Note that in this research, a gap of 0.115 m (half of the nozzles exit radius) is added, taking account the need of each nozzles to be directed (vectored). The exact gap requirements, however, will not be discussed in this preliminary design phase. With the diameter, the base area of the RSR in this design phase is $4.5$ m².

### 4.2. Rocket length and rough dimension

Based on the results tabulated in table 2, the total volume for the propellant (Liquid Hydrogen-Liquid Oxygen) can be rounded to 2.75 m³. With the determined base area before, and by neglecting the wall
thickness, for cylindrical tanks configuration, the required length of the rocket engine is at least 7.57 meter, rounded up to 7.6 meter. Hence, if it is strictly cylindrical, the required tank will consisted of a 6.3 m in height liquid hydrogen tank, and 1.3 m in height of the liquid oxygen tank.

![Diagram of rocket configuration](image)

**Figure 9.** Preliminary design result of the reusable sounding rocket.

Figure 9 shows a sketch of such rocket configurations. The volume under the tanks is preserved for the rocket engine systems, approximately 1 m of height. This value is just preserved for the sake of example. A detailed design is required to determine the dimension of this part. The part above the tanks is preserved for the payload, which dimension can be assumed to consist of the nose cone of the rocket. Note that for the diameter and length (except for the payloads), an extra 10% length is added, to take into account several extra system that may be required in the manufacturing process.

5. Conclusion
In this research, a preliminary design process of the liquid-propellant engine, for a reusable sounding rocket (RSR), has been conducted. To reach the designed maximum altitude of 120 km, several choices have been taken. This research choose a propellant mass of 6,000 kg propellant to ascent a rocket, that has initial weight of 12,000 kg, with propellant burn occurs for 100 second. Since it is a reusable rocket, a second burn is also calculated for its powered descent to the ground, of which phase requires 2700 kg propellant to be burn for 45 second. In both powered phase, the force from the rocket is 189 kN.

Since the RSR need to have a controlled descent, four nozzles is chosen for control, where each of them need to produce 47.25 kN. With liquid oxygen-hydrogen mixture, it is found that each nozzles throat area need to be at least $3.97 \times 10^{-3}$ m$^2$, while the nozzle exit area should be at least $1.64 \times 10^{-1}$ m$^2$. These two results drive the requirement for the rocket base area, which is calculated to be at least 2.38 m in diameter. From here, together with the propellant volume requirement, the oxidizer/fuel tank for this RSR need to be at least 7.57 meters in height, assuming a strictly cylindrical tank.

While several design points have been derived, this process is still in preliminary design phase. There are still many factor to be calculated and decided, for example, whether the assumed specific impulse (ISP) can be achieved or not. Furthermore, the effects of aerodynamics need to be immediately analysed, since it will be the main factor in determining the geometry of the fins and the nose cone. Lastly, weight and balance for the rocket is also mandatory to be studied, since it will affected how well the RSR can return safely to the ground.

**Acknowledgment**
This research is made possible by the P3MI grant in the Flight Physics Research Group, by the Faculty of Mechanical and Aerospace Engineering, Institut Teknologi Bandung.
References

[1] ______, NASA Sounding Rockets User Handbook Sounding Rockets Program Office Sub-orbital and Special Orbital Projects Directorate, 2015, NASA Goddard Space Flight Center, Wallops Flight Facility, Wallops Island, VA 23337

[2] ______, LAPAN Rencana Strategis Lembaga Penerbangan dan Antariksa Nasional 2015-2019, 2016, LAPAN Pusat, Rawamangun, Jakarta Timur

[3] Sembiring, T., Penelitian Prestasi Terbang Roket Sonda Satu Tingkat RX-320, 2008, Jurnal Teknologi Dirgantara Vol 6, No.2 Desember 2008, pp 83-91

[4] Hakim, A N., Rancang Bangun Enjin Roket Cair dengan Gaya Dorong 1000 Kgf Menggunakan Propelan Asam Nitrat – Kerosen, 2015, Jurnal Teknologi Dirgantara Vol. 13 No. 1 Juni 2015: pp 71-86

[5] Priamadi, E., Hakim, A N., Bura, R O., Desain Nosel Roket Cair RCX250 Menggunakan Metode Parabolik dengan Modifikasi Sudut Ekspansi, 2011, Jurnal Teknologi Dirgantara Vol. 9 No. 1 Juni 2011: pp 8-17

[6] Nonaka, S., Ogawa, H., Narua, Y., Inatani, Y., System Design and Technical Demonstration for Reusable Sounding Rocket, 2011, Proc. 20th ESA Symposium on European Rocket and Balloon Programmes and Related Research, Hyere, France pp 137-142

[7] Sippel, M., Stappert, S., Bussler, L., Dumont, E., Systematic Assessment of Reusable First-Stage Return Options, 2017, DLR-SART, the 68th International Astronautical Congress, Adelaide, Australia

[8] Lee, S., Roh, T., A Program for Design and Analysis of Liquid Rocket Engine System, 2013, Joint Propulsion Conferences, 49 AIAA.ASME.SAE.ASEE Conference, San Jose CA

[9] Tinoco, J. K., Firmo, R., Yu, C., Moallemi, M., Castro, C., & Waller, T., Commercial Space Transportation: A Simulation and Analysis of Operations Impacts on the United States National Airspace System and Airline Stakeholders, 2018, Embry-Riddle Aeronautical University, retrieved from https://commons.erau.edu/presentation/960

[10] Reed, Benjamin "Benji", International Space Station Providers, 2018. The Space Congress® Proceedings. 9. Embry-Riddle Aeronautical University, retrieved from https://commons.erau.edu/space-congress-proceedings/proceedings-2018-45th/presentations/9

[11] Sutton, G. P., Biblarz, O., Rocket Propulsion Elements, 2017, Ninth Edition John Wiley and Sons Inc., Hoboken, New Jersey

[12] Huzel, D.K., Huang, D.H., Modern Engineering for Design of Liquid-propellant Rocket Engines, 1992, the American Institute of Aeronautics and Astronautics, Washington, DC.