Design of the thrust chamber: dimensional analysis of the combustion chamber and the nozzle of rocket engine using LOX/LCH4 propellants

Hatem Houhou1,*, Hemza Layachi1 and Abd Elmajid Boudjemai2

1Algerian space agency, ASAL Algeria
2Satellite development centre, CDS Oran Algeria

*E-mail: hhouhou@asal.dz

Abstract. This paper presents a preliminary dimensional study of small thrust chamber using LOX/LCH4 as propellants. In this study, the mixture ratio of the propellants, the temperature and the pressure at the combustion chamber are used to determine the geometric parameters of the nozzle and the combustion chamber such as the volume, surfaces, lengths, diameters, etc. Our work is based on the most important required output of the propulsion engine which is the thrust force. The results of our work provides the flow areas at the exit of the nozzle and at the throat for different values of the thrust force (10-20-30 KN) and different mixture ratio of the propellants (2.7-2.8-2.9). Moreover, the volume of the combustion chamber is calculated for different proposed characteristic length (80-160-240 cm). Our results offer decisive guidelines for the design of these key parts of the thrust chamber which is required for the design of the whole rocket engine as future work.

1. Introduction

Space launch systems are composed of a great number of mechanisms assembled in order of subsystems. The performance of the vehicle depends on the specific performance of each subsystem [1, 2], which depend on the design sets and parameters [3, 4]. Liquid-propellant rocket engines as subsystem, have been researched and studied since the late of 1930s, it can be varied depending to the details of the missions [5]. Its function is to provide thrust force to lift the vehicle which is resultant of chemical reaction of propellants. The liquid methane is classified as a no-toxic green propellant with high storage temperature (-161.6°C) comparing to the other fuels used in this field (example: liquid hydrogen -252.9°C) which mean that its storage tank need lower cost for the thermal insulation technologies [6]. Moreover, the tanks could be smaller since the density of the liquid methane is high [7], once more this is a saving of money and weight, that’s explain our chose of liquid methane as a propellant for this study.

The combustion chamber is a key part of the thrust chamber where the chemical reaction take place to generate the gas with high pressure and high temperature used to produce the thrust force. To determine the energy required for this thrust force, we need to take in consideration the mass flow rates of the propellants, their mixture ratio and the specific impulse are defined as decisive parameters for the design set.

Another main part of the thrust chamber is the nozzle, whose role is to create high exhaust velocity. In the design of this two parts, the parameters to determine are the volume of the combustion
chamber, the flow areas at the nozzle exit and at the throat. A useful parameter relative to chamber volume and the nozzle sonic throat area is the characteristic length, which is the length that a chamber of the same volume would have if it were a straight tube and had no converging nozzle section [8]. When designing higher rockets, it’s better to compare data in deciding volume and shape instead of using characteristic chamber length, but for smaller rockets with fewer data available it’s practical. Standards of characteristic chamber length lies between 0.8-3.0 m [9]. Again it’s important to choose a value appropriate for one’s propellant. A high value would in theory cause higher reaction and thus a better performance, however in reality this would cause loss of power.

In order to calculate the geometric parameters related to the design of the combustion chamber and the nozzle (volume of the combustion chamber, the flow area at the nozzle exit and at the throat), different desired engine specifications such as thrust force, specific impulse and characteristic length are combined with different pressure and temperature resultant of burns LOX/LCH4 with same mixture ratio.

2. Case of study
This study is based on the proposed engine specifications shown in the (table 1).

| Parameters             | Specification value |
|------------------------|---------------------|
| Thrust                 | 10-20-30 (KN)       |
| Specific impulse       | 250 (s)             |
| Mixture ratio          | 2.7 - 2.8 - 2.9     |
| Characteristic length  | 80 - 160 - 240 (cm) |

Using the adiabatic flame temperature chart ‘Figure 1’, versus combustion chamber pressure for LOX/LCH4, we extract different combine of (Temperature (\(T_c\))/Pressure (\(P_c\))) to be used in next section.

![Figure 1. The adiabatic flame temperature chart][10].

3. Mathematical model
In this section, the mathematical equations used to calculate the desired parameters of our study are presented, including the volume of the combustion chamber, the nozzle and the throat exit areas.

3.1. Mass flow rate of propellants
The total mass flow of the propellants (\(m_{tot}\)) is related to the thrust force (\(F\)), the specific impulse (\(I_{sp}\)) and the gravity constant (\(g\)), described as follows:

\[
m_{tot} = \frac{F}{I_{sp} \times g}
\]
The mass flow of the oxidizer (\(m_{\text{ox}}\)) and the fuel (\(m_{\text{fu}}\)) is calculated using the mixture ratio (MR) and the total mass flow as shown in the following equations:

\[
m_{\text{ox}} = \frac{\dot{m}_{\text{tot}}}{1 + \frac{1}{\text{MR}}}
\]

\[
m_{\text{fu}} = \dot{m}_{\text{tot}} - m_{\text{ox}}
\]

3.2. Nozzle

At the nozzle throat, the pressure \((P_t)\) and the temperature \((T_t)\) are given as:

\[
P_t = P_e \left(1 + \frac{\gamma - 1}{2}\right)^{-\gamma/(\gamma - 1)}
\]

\[
T_t = \frac{T_e}{1 + \frac{\gamma - 1}{2}}
\]

Mach number \((M_a)\) is the ratio of the gas velocity to the local speed of sound. The Mach number at the nozzle exit is given by the perfect gas expansion expression:

\[
M_a = \left(\frac{2}{\gamma - 1}\right) \left[\left(\frac{P_e}{P_a}\right)^{(\gamma - 1)/\gamma} - 1\right]^{1/2}
\]

where \(\gamma\) is the specific heat ratio and \(P_a\) is the atmospheric pressure.

The nozzle throat area \((A_t)\) can be found if the total propellant flow rate is known and operating conditions have been selected. Assuming perfect gas law theory, we have:

\[
A_t = \frac{\dot{m}_{\text{tot}}}{P_e} \left(\frac{RT_t}{\gamma M_a}\right)^{1/2}
\]

assumed that the gas flow through the nozzle is an isentropic expansion, the expansion ratio \((\varepsilon)\) is defined as follows:

\[
\varepsilon = \frac{A_e}{A_t} = \frac{\left(\frac{P_t}{P_e}\right)^{(\gamma - 1)/\gamma} \frac{2}{\gamma + 1}}{\frac{\gamma + 1}{\gamma - 1} \left[1 - \left(\frac{P_a}{P_e}\right)^{\gamma/(\gamma - 1)}\right]^{1/2}}
\]

According to the equation (8), the nozzle exit area \((A_e)\) is determinate as:

\[
A_e = A_t \varepsilon = A_t \times \frac{\left(\frac{P_t}{P_e}\right)^{(\gamma - 1)/\gamma} \frac{2}{\gamma + 1}}{\frac{\gamma + 1}{\gamma - 1} \left[1 - \left(\frac{P_a}{P_e}\right)^{\gamma/(\gamma - 1)}\right]^{1/2}}
\]
3.3. The combustion chamber

The theoretically required volume of the combustion chamber \( V_c \) have relation with the mass flow rate of the propellants, the density of the produced combustion gases and the stay time, or combustion residence time needs for efficient combustion, expressed by the following equation:

\[
V_c = m_{\text{tot}} V_{av} t_s
\]  

(10)

where \( V_{av} \) is the average specific volume and \( t_s \) is the propellant stay-time.

Moreover, the volume of the combustion chamber can be calculated using the definition of the characteristic length \( L_x \):

\[
V_c = A_t L_x
\]  

(11)

4. Result and discussion

Using the deferent extracted combine \( (T_c/P_c) \) from the adiabatic flame temperature chart ‘Figure 1’ and the proposed engine specification (table 1), the throat and the nozzle exit flow areas were calculated for deferent mixture ratio and deferent thrust force.

‘Figure 2’ and ‘Figure 3’ shows that as the thrust force is high as the flow areas are large which mean that the propulsion engine will be big. This result can be used as tools to decide the number of the used engines instead of using only one big to get the same required thrust force.

![Figure 2. The throat flow area for deferent proposed thrust forces and mixture ratios.](image)

![Figure 3. The nozzle exits flow area for deferent proposed thrust forces and mixture ratios.](image)

The calculated throat flow area was combined with the proposed characteristic lengths of the combustion chamber to calculate its volume using the equation (11). ‘Figure 4’, ‘Figure 5’ and ‘Figure 6’ displays the volume of the combustion chamber for 80cm, 160cm and 240cm of characteristic length respectively. From this figures we note that the characteristic length is one of the decisive parameters to define the size of the combustion chamber, these lasts have proportional relationship (as the characteristic length is large, the volume the combustion chamber should be big enough) this is explained by the equation (10), since the throat area is proportion to the mass flow of the propellants and the specific volume, the characteristic length is function of stay-time needed for an efficient combustion.
Figure 4. The volume of the combustion chamber for different proposed thrust forces and mixture ratios ($L_s = 0.8$ m).

Figure 5. The volume of the combustion chamber for different proposed thrust forces and mixture ratios ($L_s = 1.6$ m).

Figure 6. The volume of the combustion chamber for different proposed thrust forces and mixture ratios ($L_s = 2.4$ m).

5. Conclusion
This study was based on a different proposed engine specification combined with different thermodynamic parameters related to the combustion of LOX/LCH4 to determine the geometric parameters of the nozzle and the combustion chamber. The flow areas at the nozzle (throat and nozzle exit) designed to exhaust the gases resultant from the combustion and to satisfy the needed thrust, it is necessary to give the suitable dimensioning for this chamber, in this paper the different dimensions to respect in order to obtain that needed thrust were analyzed. Our results show that for high thrust force, the flow areas are large which mean that the propulsion engine to ensure that is big, this is such a problem for the manufacturing procedure but for that, using more than one small engine to get the desired thrust force is an alternative solution. Also the relation between the volume of the combustion chamber and the characteristic length is helpful for the efficiency of the combustion by deciding the ideal stay-time of the produced combustion gases.

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