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## **Modeling of a blow-down propulsion system**

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### **Lockheed Martini Group**

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## Notation

|             |                        |                     |                            |
|-------------|------------------------|---------------------|----------------------------|
| <b>LRE</b>  | Liquid Rocket Engine   | $O/F$               | Oxidizer to Fuel ratio [-] |
| <b>LOX</b>  | Liquid Oxygen          | $\varepsilon$       | Expansion ratio [-]        |
| <b>RP-1</b> | RP-1 fuel              | $\varepsilon_{con}$ | Contraction ratio [-]      |
| <b>AM</b>   | Additive Manufacturing | $L^*$               | Characteristic length [m]  |

# 1 Introduction and literature overview

In this work a preliminary design of a 1 kN semi-cryogenic LRE (LOX/RP-1) equipped with a blow-down feeding system is discussed. In particular, a first literature analysis was done in order to review previous studies regarding this particular architecture. Recent developments in additive manufacture (AM) techniques were analyzed to obtain some knowledge regarding processes and precision of this new frontier. Moreover, due the reduced size of this system, some criticalities regarding boundary layer and erosion losses were researched. The second part of the paper aim at designing the engine with some imposed initial conditions and some assumptions. The whole dimensioning of the system, including the tanks and feeding lines, is carried on including the dynamics of the system. The final sizing will accomplish the maximization of the total impulse, with the initial and final constraints. An off-design analysis is then performed to quantify the performances with nozzle losses and AM uncertainties. Finally, a feasibility analysis of nozzle fuel cooling is discussed.

## 1.1 Blow-down heritage

The blow-down architecture is the simplest feeding technique for LRE since it does not require additional pressurizing gas tanks with failure-prone pressure regulator valves nor complex turbomachinery. The scheme includes only two liquid propellant tanks filled with helium or nitrogen, eventually separated by a membrane. The major downsides of this simplicity is relative to the non-stationarity of the tank pressures that induce chamber pressure drop, decrease of propellant mass flow rate and as a consequence O/F ratio variation. This chain of events degrades performances overtime and must be carefully evaluated since combustion efficiency relies upon viable domains of injection pressure and correct mass flow ratio. The interest on blow-down is although justified with respect to well-known pressure regulated feed system since this last can also manifest some criticalities in terms of long-term reliability. In particular, propulsion systems play crucial roles for mission success, such as long interplanetary trips, and they must ensure failure-free lifetime. This is a major concern when focusing on pressure regulated feeding lines in which a pressure regulator valve is present. This kind of elements can be quite complex and hence add a weakness for the whole system<sup>[1]</sup>. Considering these facts, a blow-down type architecture could be of interest since it decrease system complexity. Moreover, different feasibility analysis for blow-down units are present in the literature in which also an external re-pressurization tank is considered<sup>[2]</sup>. This is an upgrade that allows to recover performance of the feeding pressure and hence combustion properties. Although the valve complexity is removed since a pyro valve can be used to discharge the gas with a single shot application, the eventual re-pressurization can be a crucial point as the sudden change in pressure could induce unwanted instabilities. Other configurations could foresee the use of a Venturi valve to maintain constant mass flow rate by cavitating the liquid and choking the flow on the feeding line. However, neither extra tank nor Venturi valves will be considered in this work in order to meet the requirements presented in (reference a section successiva).

The whole evaluation of the dynamics of the examined propulsion system was not based upon previous works, instead a self-made model was developed.

## 1.2 Additive manufacturing state of art

## 1.3 Analysis of losses

# 2 Modeling of propulsion system: DRY-1

The workflow for the sizing of DRY-1 is introduced by Figure 1 and it is divided into three stages:

- **Input data:** the problem is set up.
- **Nominal sizing:** the system is sized according to initial conditions and general assumptions.
- **Dynamics:** an iterative process is set up to model the blow-down dynamics and finalize the sizing.

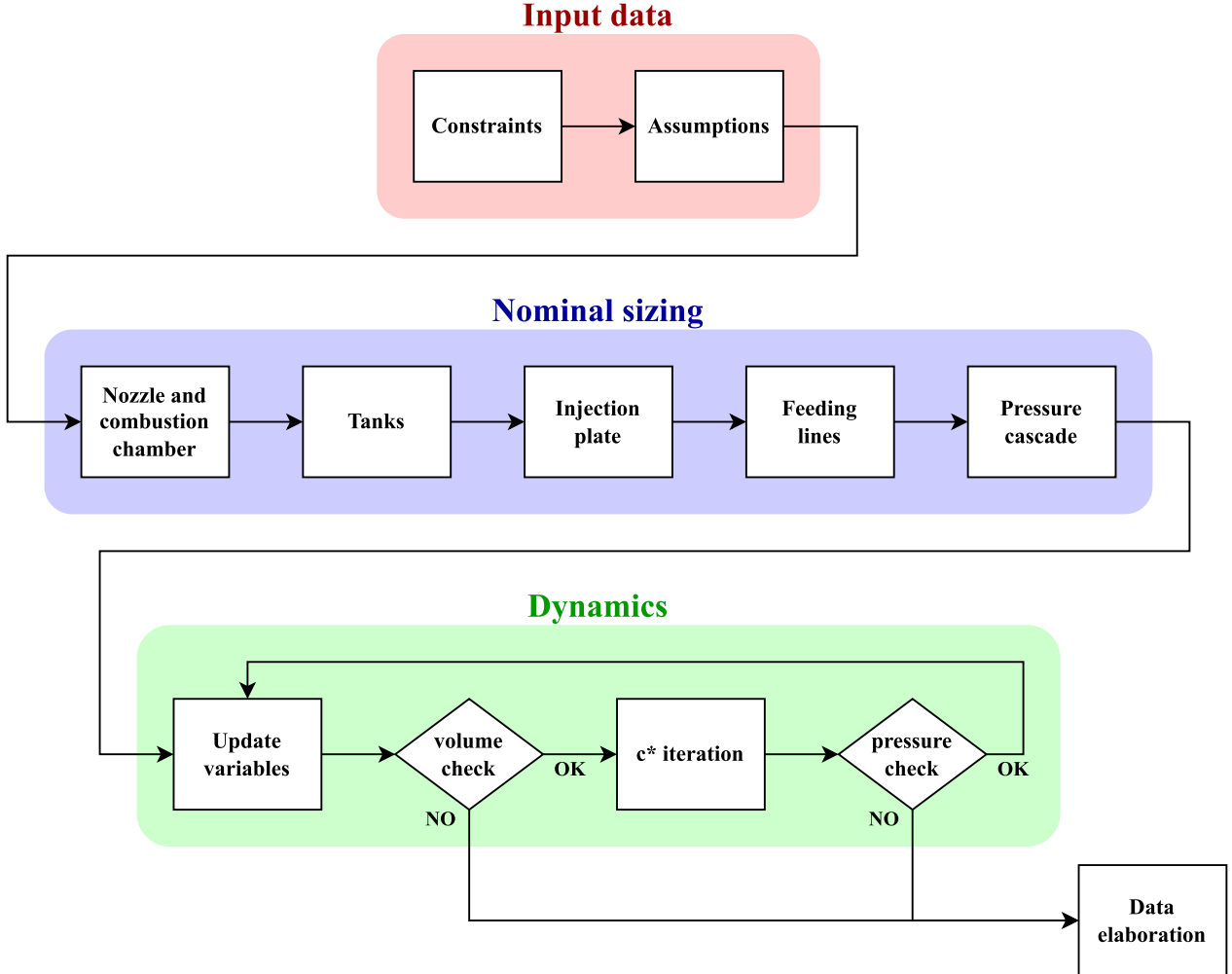


Figure 1: Flowchart of the sizing

## 2.1 Input data

The input data defined different kind of requirements, related to operability environment, engine performance, size constraints, chemistry, architecture and manufacturing techniques. Regardless of the development of the engine design, the system shall respect the following pinpoints:

- **Environment:** vacuum for the whole operation.
- **Thrust  $\mathcal{T}$ :** initial magnitude of 1 kN, no lower boundary.
- **Chamber Pressure  $P_c$ :** initial value of 50 bar, always above 20 bar throughout the whole mission.
- **Allocated Space:** tanks, combustion chamber and convergent nozzle occupancy is exactly 80% of the volume occupied by a cylinder of 1 meter diameter and 2 meter length. No bounds on the extension of the divergent.
- **Propellants:** semi-cryogenic couple of LOX and RP-1.
- **Architecture:** blow-down type.
- **Manufacturing:** all the system is produced in AM, no restriction on material nor techniques.

The nominal sizing refers to the design of the overall system considering the imposed initial constraints as static conditions. This design choice was dictated by the dynamics of the blow-down system, which imposes

the maximum flow rate at the beginning of the mission, leading to the oversizing of the engine throughout the rest of the mission.

Various hypothesis were necessary to develop the system, this values are reported in Table 1.

| $O/F$ [-] | $\varepsilon$ [-] | $\varepsilon_{con}$ [-] | $L^*$ [m] |
|-----------|-------------------|-------------------------|-----------|
| 2.42      | 300               | 10                      | 1.143     |

Table 1: Hypothesis from literature and previous design

The choices of  $L^*$  and  $O/F$  were only dictated by the propellant couple<sup>[3]</sup>, while  $\varepsilon$  was chosen as the characteristic of the engine refers to an in-space application<sup>[4]</sup>. Regarding the value of  $\varepsilon_{con}$  a mean value between 5 and 15 was taken. Smaller values entails longer combustion chamber and small cross sectional area, with large pressure drops. Larger values refers to bigger chamber cross sectional area, with limited length for the combustion. From the literature the suggestion for the choice of this value is to refer to previous successful engines design, considering the same application<sup>[5]</sup>. Therefore, a 400N Bi-Propellant apogee motor was taken as reference and revealed a value of  $\varepsilon_{con} \approx 10$ <sup>[4]</sup>.

## 2.2 Nominal sizing

After defining the main input data, the workflow is carried out as shown in Figure 1. All the combustion simulations were performed with Nasa-CEA software, implemented in Matlab (CEAM). In particular the "rocket problem" was considered, imposing frozen equilibrium, infinite combustion chamber ( $M_c = 0$ ), initial injecting temperatures of the propellant equal to the storage temperatures. The chamber pressure was set as  $P_c = 50$  bar and the mixture ratio as  $O/F = 2.42$ . Latter refinement of this last value will be performed. The output values used from the simulation are represented (vacuum value is considered for the  $c_T$ ):

| $c^*$ [m/s] | $c_T$ [-] | $T_c$ [K] | $\gamma_c$ [-] |
|-------------|-----------|-----------|----------------|
| 1851        | 1.935     | 3708      | 1.1405         |

Table 2: First run on CEAM

From this results, the propellant mass flow rate and the throat area can be calculated:

$$\dot{m}_p = \frac{\mathcal{T}}{c_T c^*} \quad A_t = \frac{\dot{m}_p c^*}{P_c} \quad (1)$$

From the geometry assumption of Table 1, the nozzle exit area and the combustion chamber geometry can be retrieved:

$$A_e = \varepsilon A_t \quad (2)$$

$$A_c = \varepsilon_{con} A_t \quad L_c = \frac{L^*}{\varepsilon_{con}} \quad (3)$$

Nozzle geometry model will be delved in section 4. The results for the previous calculation are the following:

| $\dot{m}_p$ [kg/s] | $d_t$ [cm] | $d_e$ [cm] | $d_c$ [cm] | $L_c$ [cm] |
|--------------------|------------|------------|------------|------------|
| 0.279              | 1.15       | 19.86      | 3.63       | 11.43      |

Table 3: Preliminary DRY-1 geometry

A check on the compliance of the chamber Mach number is done ( $M_c < 0.3$ ):

$$\frac{1}{\varepsilon_{con}} = M_c \left[ \frac{1 + \frac{\gamma-1}{2}}{1 + \frac{\gamma-1}{2} M_c^2} \right]^{\frac{\gamma+1}{2(\gamma-1)}} \xrightarrow{fsolve} M_c = 0.059 \quad (4)$$

From the geometry of the motor and the allocated space constraints, the total height of the tanks can be calculated. The volume around the thrust chamber (combustion chamber + convergent) can be assessed:

$$V_{tc} = \frac{\pi}{4} \left[ L_c d_c^2 + \frac{L_{con}}{3} (d_c^2 + d_t^2 + d_c d_t) \right] \quad (5)$$

Also, the volume of the cylinder that covers the length of the thrust chamber and with the total diameter of 1 meter ( $d_{tot}$ ) can be computed. From there, the empty volume around the thrust chamber can be found as a difference:

$$V_{lost} = V_{tc} - \frac{\pi}{4} (L_c + L_{con}) d_{tot}^2 \quad (6)$$

This value must be 20% of the total cylinder volume, as cited in [subsection 2.1](#). As the computed value was lower, additional volume had to be removed from the tanks in order to meet the requirement. The height dedicated to the tanks is calculated as follow:

$$h_{tk} = h - \left[ L_c + L_{con} + \frac{4}{\pi d_{tot}^2} (0.2 V_{tot} - V_{lost}) \right] \quad (7)$$

The total volume allocated to the tanks is hence fully defined. In order to calculate the masses of pressurizer and propellants, some assumption have to be made:

- **Adiabatic expansion of the pressurizing gas.**
- **Blow-down ratios can be tuned.**
- **Mean value of the oxidizer to fuel ratio.**

A system of equations can be set up:

In order to solve (autoref), the pressurizer and the initial temperature of the propellants have to be set. Different pressurizing gas are available, mainly nitrogen or helium are the most common choices. The main differences for the two are the storage temperature, the molar mass and the specific heat ratio. The latter parameter influences the adiabatic discharge, higher values implies faster pressure discharge. The molar mass affects the amount of gas to be embarked at a given pressure. The storage temperature is a matter of compatibility with the propellant. Nitrogen gas was chosen to pressurize RP-1 since no cryogenic conditions were present also it guarantees lower discharge, with the downside of increasing the mass of the system. On the other side, LOX required a cryogenic compatibility that can be ensured by helium. Even though an efficient insulating bladder is employed, the design choice was dictated by a more conservative approach. The employment of nitrogen also with LOX was discarded since storage temperature and pressure are not compatible with its properties. (reference NIST)

| $T_{pr, fu} [K]$ | $T_{pr, ox} [K]$ | $\gamma_{N_2} [-]$ | $\gamma_{He} [-]$ |
|------------------|------------------|--------------------|-------------------|
| 300              | 90               | 1.40               | 1.67              |

**Table 4:** Initial values for pressurizer gases and specific heat ratio values

The masses and volumes of oxidizer, fuel and pressurizing gases are computed.

Finally, the feeding lines can be modelled. Considering figure (...), the length of the pipes can be retrieved as a difference:

$$L_{fd, fu} = h - L_c - h_{tk, fu} \quad (8)$$

$$L_{fd, ox} = h - L_c - h_{tk} \quad (9)$$

To completely determine the DRY-1 geometry, the injection plate must be modeled. The pressure drop across the injector  $\Delta P_{inj}$  have to be assumed as a percentage of the initial chamber pressure. Acceptable range of this fraction goes from 5% to 30%. A value of 20% is chosen for both oxidizer and fuel lines. The fuel and oxidizer injector area can be computed assuming reasonable values for the discharge coefficient  $C_d$ . A reasonable assumption has been made according to the superficial roughness quality of AM.

$$\dot{m}_{fu} = \frac{1}{1+O/F} \dot{m}_p \quad \dot{m}_{ox} = \frac{O/F}{1+O/F} \dot{m}_p \quad (10)$$

$$A_{inj,tot} = \frac{\dot{m}_p}{C_d \sqrt{2\Delta P_{inj} \rho_p}} \quad (11)$$

### 2.3 System dynamics

## 3 Results analysis

### 4 Nozzle losses

### 5 Additive manufacturing influences

### 6 Cooling analysis



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