Università degli Studi di Napoli
Federico II
Scuola Politecnica e delle Scienze di Base

Corso di Laurea Magistrale in Ingegneria Aerospaziale

VEGA'S P80 FIRST STAGE

PERFORMANCE RECONSTRUCTION OF A SOLID ROCKET MOTOR

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M53/1488

Outline

- Overview
- Requirements
- Sizing
  - Performance Parameters
  - Grain
  - Blueprint
  - Combustion Chamber
- Deliverables

1. Overview

- VEGA
- P80 (FIRST ROCKET STAGE)
- OBJECTIVE

Overview - VEGA

- Vega (Vettore Europeo di Generazione Avanzata) is an expendable launch system in use by Arianespace jointly developed by the Italian Space Agency (ASI) and the European Space Agency (ESA).
- Vega joined the family of launch vehicles at Europe's Spaceport in French Guiana in 2012.

VEGA at the National Museum of Science and Technology
(courtesy of AVIO Group)

Overview - VEGA

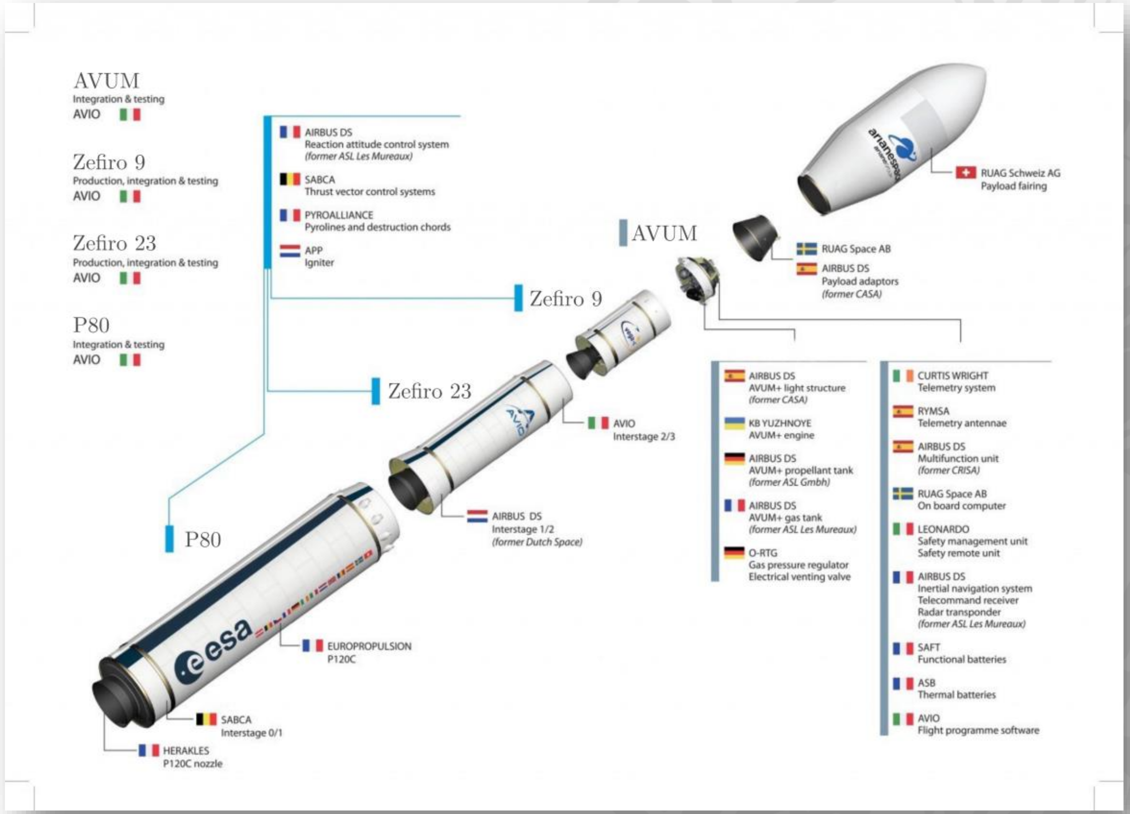
- VEGA is a single-body launcher with three solid rocket stages and the upper module is liquid rocket:

Solid

- I stage: P80
- II stage: Zefiro 23
- III stage: Zefiro 9

Liquid

- IV stage: AVUM



The multi-national contributions to Vega

(courtesy of ESA – European Space Agency)

Overview – P80

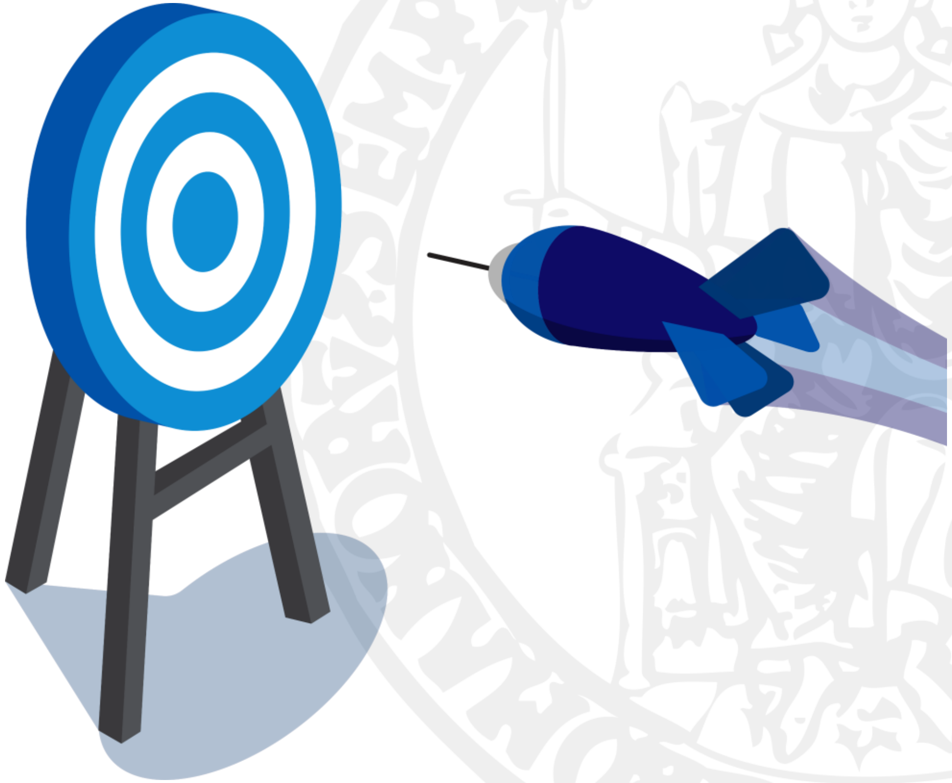
- P80 is solid-fuel first-stage rocket motor used on Vega rocket.
- It is the world's largest and most powerful one-piece solid-fuel rocket engine.

Departure of the P80 first stage from the Booster Integration Facility

(courtesy of Arianespace)

Overview – Objective

- The main objective of this investigation is to obtain a performance reconstruction of the P80 rocket and to propose a preliminary sizing for the P80.



2. Requirements

- Main characteristics
- Propellant specs

Requirements - Main characteristics

P80 Rocket Motor Characteristics [2,3,4,6]

Length **L** (m)	7,50
Motor diameter $\mathbf{d_m}$ (m)	3,00
Nozzle diameter $\mathbf{d_n}$ (m)	1,98
Case thickness $\mathbf{d_{th}}$ (mm)	5
Nozzle expansion ratio **ε**	16
Maximum pressure $\mathbf{P_{max}}$ (bar)	95,00
Propellant mass $\mathbf{M_p}$(kg)	80180
Specific impulse $\mathbf{I_{sp}}$ (s)	280
Thrust **S** (kN)	3037

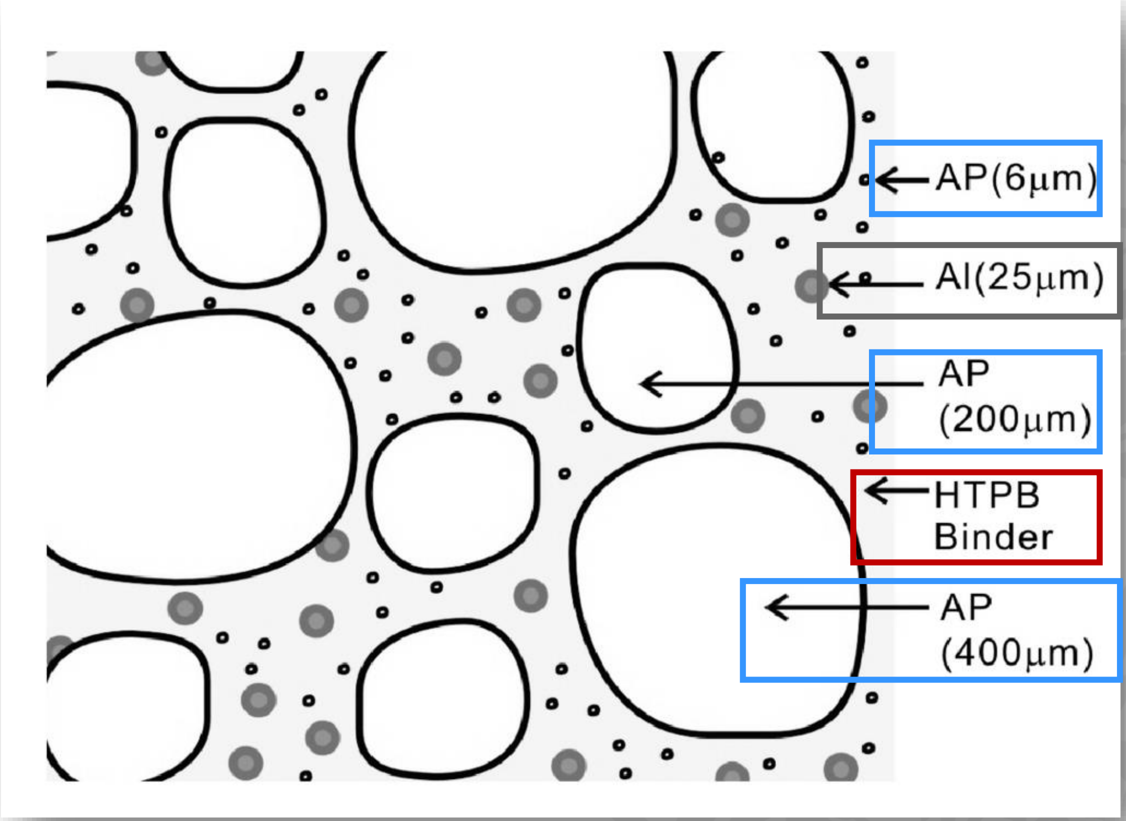
Requirements – Propellant Specs

HTPB-based 1912 propellant [1]

Oxidizer (%)	69
Aluminium (%)	19
HTPB (%)	12
Chamber Temperature $\mathbf{T_c}$ (K)	3550
Density $\boldsymbol{\rho_p}$ (kg/m^3)	1810
Burning rate $\dot{r}$ (in/s)	0,5
n	0,39

Requirements – Propellant Specs

- HTPB 1912 is composite propellant, consisting of an oxidizer and a reductant mixed.
- In more detail, the oxidizer is in the form of crystals which are salts either chlorine-based (AP) or nitrogen-based ;
- On the other hand, as far as reducing agents are concerned, the most used polymer is HTPB, a polymer whose single monomer is butadiene, a plastic organic matter essentially based on carbon and hydrogen.
- Aluminum (Al) particles, added as they allow for a much higher rate of regression.



Schematic diagrams of composition distribution in a HTPB-based propellant [5]

3. SIZING

- PERFORMANCE PARAMETERS
- GRAIN
- BLUEPRINT
- COMBUSTION CHAMBER

Sizing – Performance Parameters

- In order to evaluate the performance parameters, the RPA(*) software has been used.

Engine Definition
- Chamber Pressure

Propellant Specification
- HTPB 1912

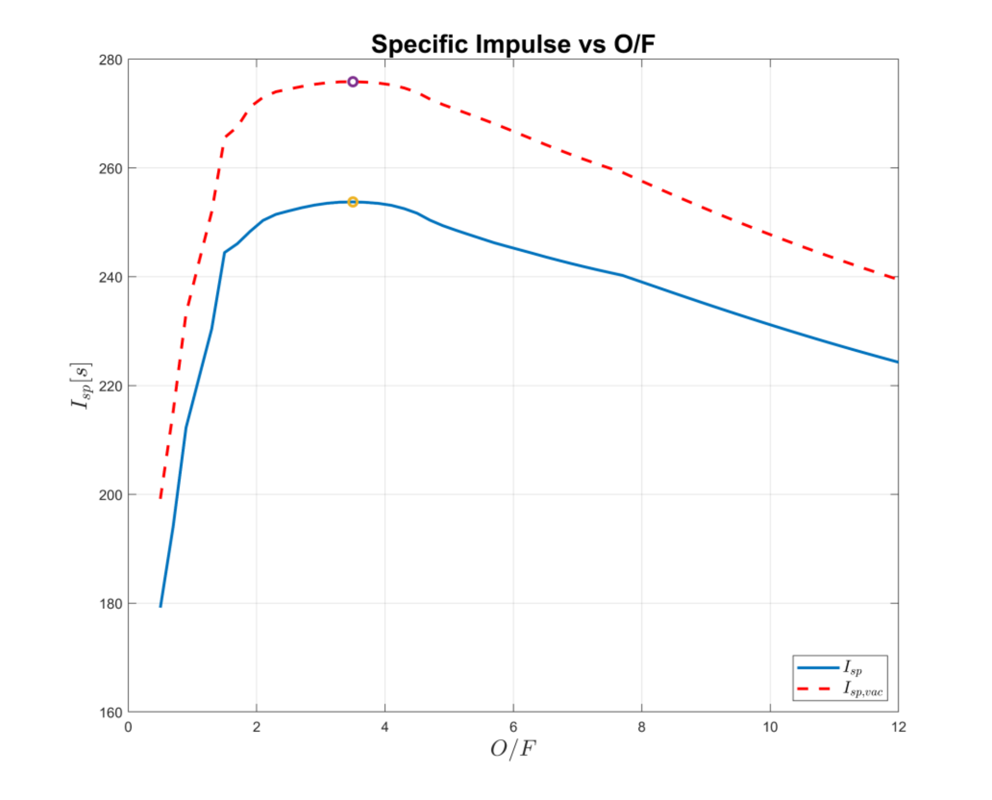
Nozzle Flow Model
- Expansion Area Ratio

Chamber Performance
- Ideal Performance

Nested Analysis
- Evaluation of the parameters vs OF

(*) Rocket Propulsion Analysis – Lite Edition

Sizing – Performance Parameters



The data were plotted using MATLAB

- To better visualize the data collected after the RPA analyses, the specific impulse has been plotted.
- Conclusions:
  - $I_{sp,max}$ is obtained in vacuum
  - $(O/F)_{id}$ is about 3,416

Sizing – Performance Parameters

- Using the collected data from the software and the formulas known, it is possible to draw a preliminary performance evaluation:

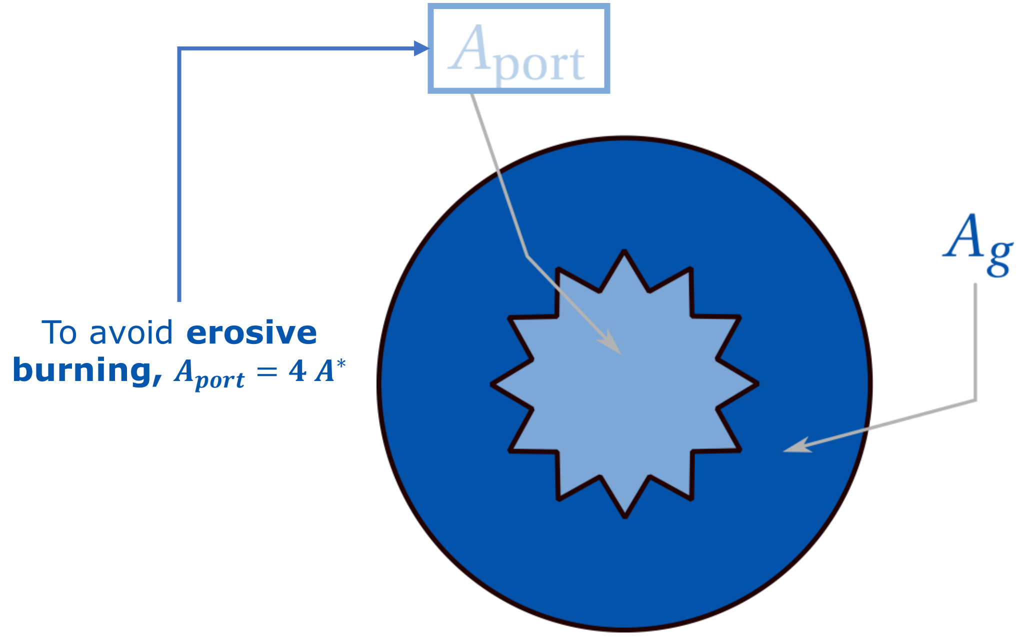
$$c^* = \frac{\sqrt{RT_c}}{f(\gamma)} = 1558.780\, m/s \qquad \leftarrow \qquad \text{Characteristic velocity}$$

$$c_f^0 = f(\gamma)\sqrt{\eta_{noz}} = 1.685 \qquad \leftarrow \qquad \text{Thrust coefficient}$$

$$c = c^*\, c_f^0 = 2626.544\, m/s \qquad \leftarrow \qquad \text{Effective exhaust velocity}$$

Sizing - Grain

- Let's calculate some of the main geometrical characteristic of the grain.



$$A_e = \left(\frac{d_n}{2}\right)^2 \pi = 3.078\,m^2$$

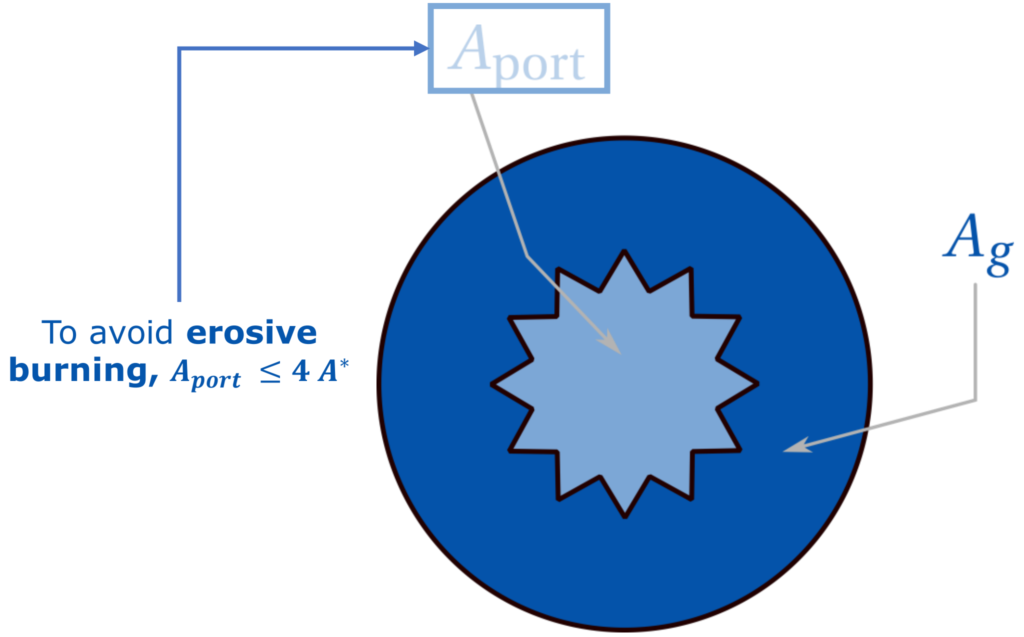
$$A^* = \frac{A_e}{\varepsilon} = 0.192\,m^2$$

$$A_g = \left(\frac{d_c}{2}\right)^2 \pi = 7.018\,m^2$$

$$d^* = \sqrt{\frac{4A^*}{\pi}} = 0.496\,m = 49.6\,cm$$

Sizing - Grain

➢ Let's calculate some of the main geometrical characteristic of the grain.



$$A_{\text{port}} = 4\,A^* = 0.772\,m^2$$

$$V_0 = A_g\,L_c \qquad V_{\text{port}} = A_{\text{port}}\,L_c$$

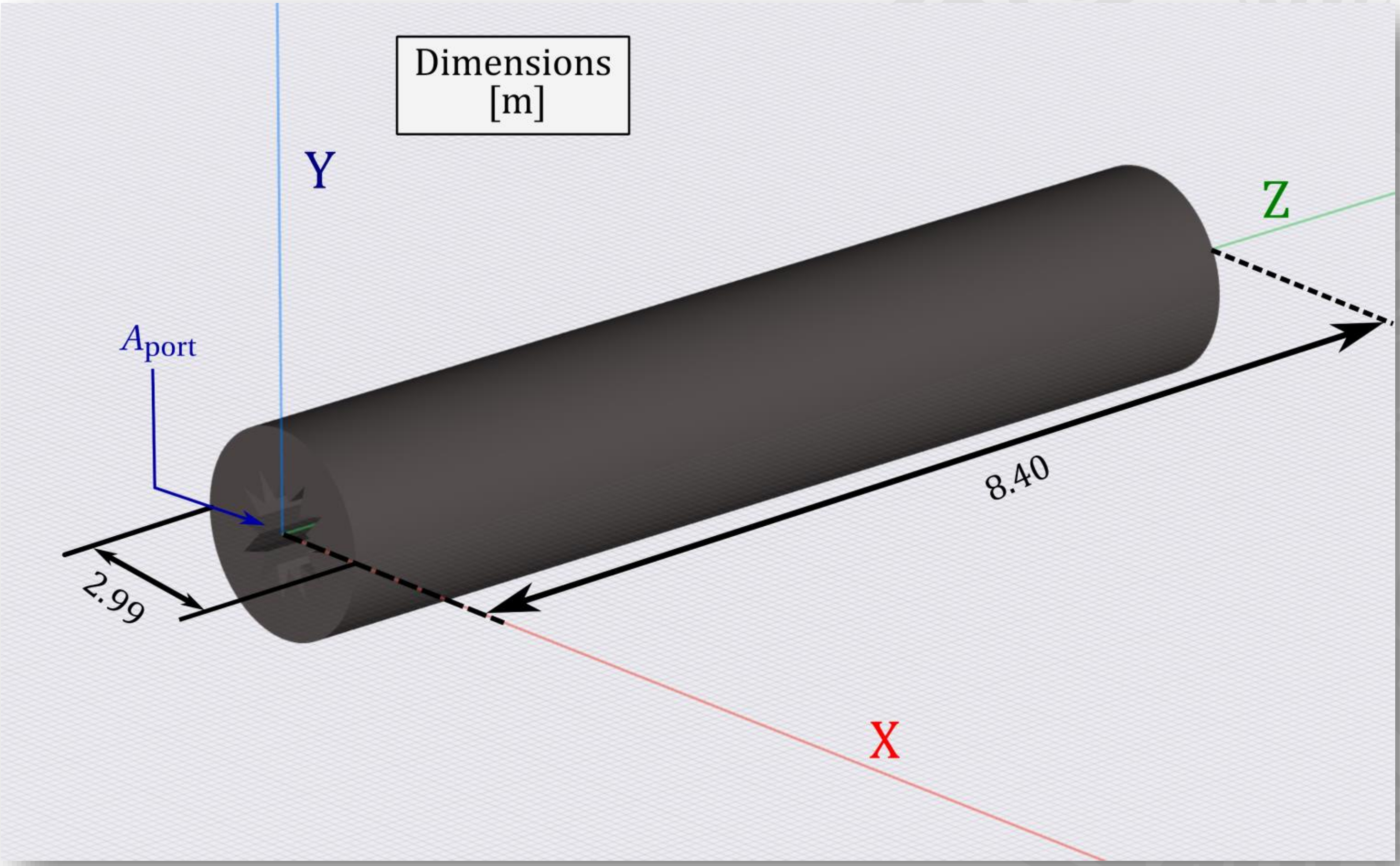
$$V_g \overset{\text{def}}{=} \frac{M_p}{\rho_p} = 44.298\,m^3$$

$$V_g = V_0 - V_{\text{port}}$$

$$\boxed{L_c = \frac{V_g}{A_g - A_{\text{port}}} = 7.09\,m}$$

Sizing - Grain

- Using the dimensions previously found, it was possible to draw a CAD model of the grain.



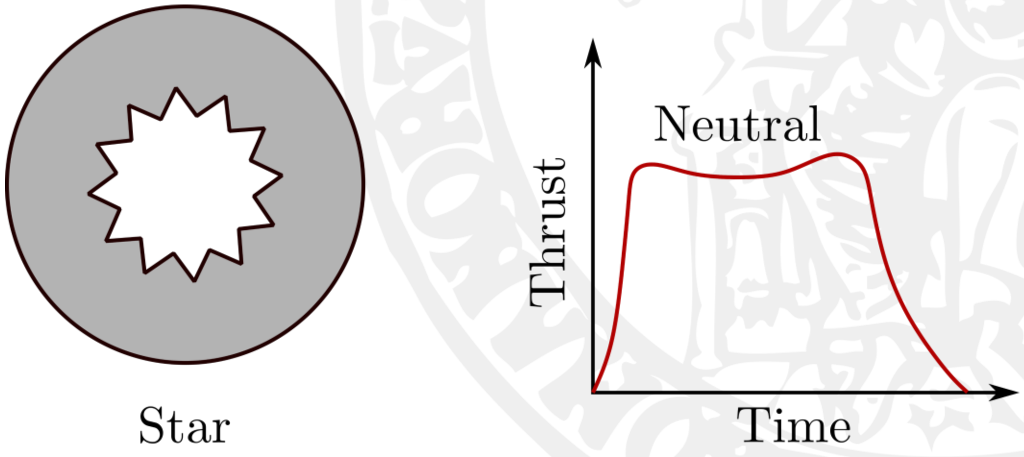
CAD drawn using Shapr3D

Sizing - Grain

- Assuming a case thickness of about 5 mm, the grain diameter will be $d_c = 2{,}990\ m$
- The star-shaped grain has been adopted in order to have an almost neutral thrust curve. In general, the grain geometry is such that the wetting area is about the 90% of the external grain circumference.

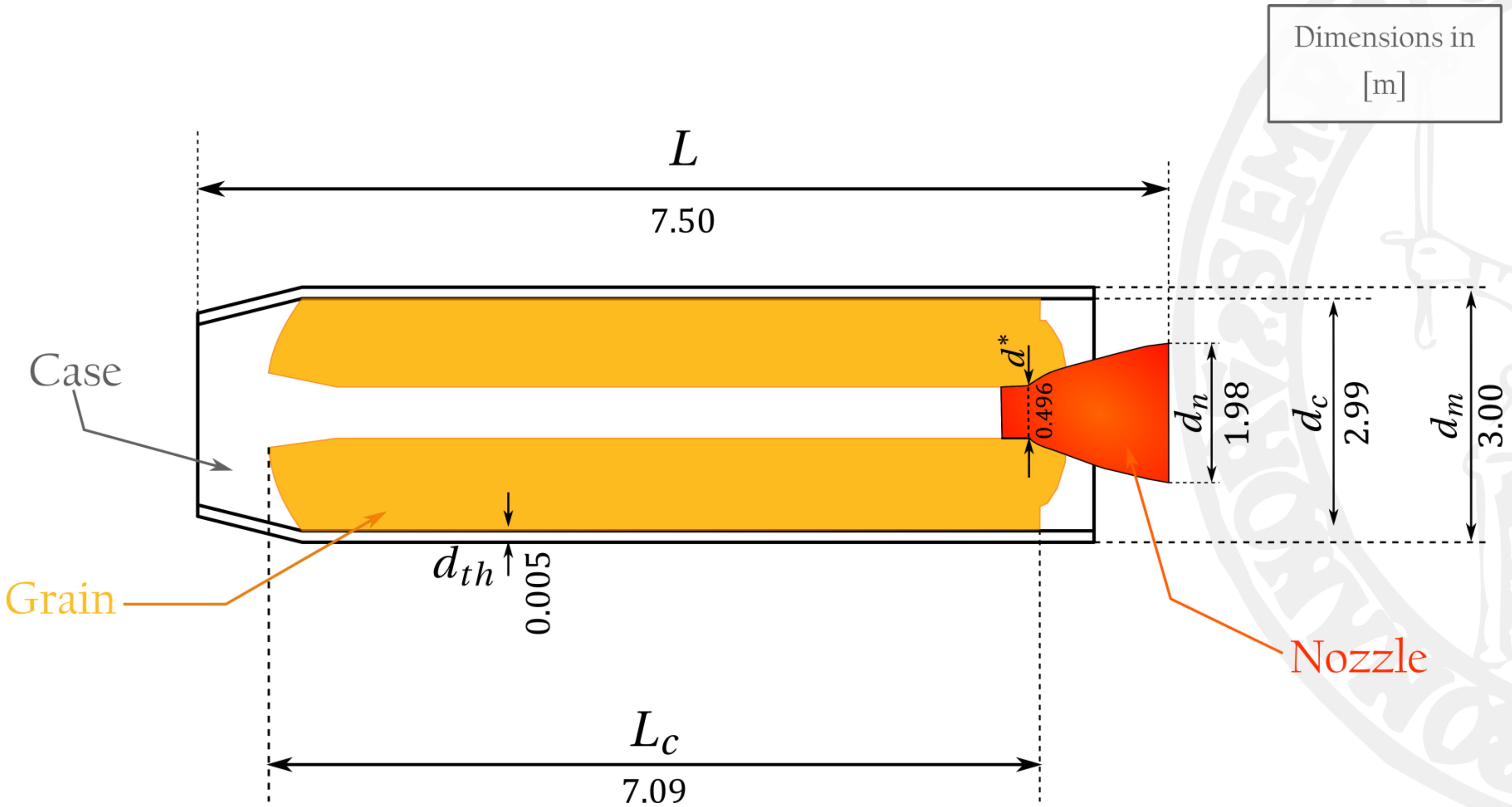
$$A_b = (\text{burn perimeter}) \times (\text{charge length}) =$$

$$= 0.9\,\pi\, d_c\, L_c = 59.909\ m^2$$



P80 has a 12-star shaped grain [3]

Sizing – Blueprint



P80's dimensions obtained. *(non to scale)*

Sizing – Combustion Chamber

- Looking at the tables previously shown, it is possible to obtain the chamber pressure (P_c).

$$A_e = 3.078\, m^2$$

$$A^* = 0.193\, m^2$$

$$A_b = 59.909\, m^2$$

The obtained pressure must be less than P_{max}

$$P_c = \left[\frac{1}{a\rho_p c^*}\right]^{\frac{1}{n-1}} \left[\frac{A^*}{A_b}\right]^{\frac{1}{n-1}} \simeq \boxed{90.456\ bar}$$

Summerfield Criteria is met

$$\boxed{\frac{P_e}{P_c} \simeq 0.011 \le \alpha = 0.25 - 0.35}$$

$$\boxed{P_c < P_{max}}$$

The throat area is <u>compatible</u> with the requirements imposed

Sizing – Combustion Chamber

- To account for various losses, the ideal thrust coefficient c_f can be reduced by a factor of 7%.
- Moreover, the thrust S, and the total impulse I_t, and the specific impulse I_{sp} can be obtained.
- Finally, it is possible to evaluate the propellant mass flow rate $\dot{m}_p$, the burn time t_b and the web thickness w can be evaluated.

$$c_{f,real} = 0.93\, c_f = 1.567$$

$$S = c_{f,real}\, P_c\, A^* = 2700.010\ kN$$

$$I_{sp} = \frac{c^*\, c_{f,real}}{g_0} = 248.848\ s$$

$$\dot{m}_p = \frac{S}{c} = 1027.872\ kg/s$$

$$t_b = 85\ s$$

$$I_{tot} = S\, t_b = 210.794 \times 10^3\ kN \cdot s$$

$$w = \dot{r}\, t_b = 1.080\ m$$

Bibliography and Sitography

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THANK YOU
FOR YOUR ATTENTION