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Department of Aerospace Science and Technology

Aerospace Propulsion

Atlas V 551
Final degree project

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Abstract

The project reports an analysis of the two-stage rocket launcher Atlas V model 551, designed and built by ULA (United Launch Alliance). In particular only the first stage, the Common Core Booster (CCB), is considered.
Atlas V 551 is the largest of the Atlas V Launcher Family (which rockets have flown since 2002 with nearly perfect success rate) flown to date featuring five Solid Rocket Boosters and a 5.4-meter Payload Fairing [1].
This rocket has been used for several important missions, such as the recent InSight Mission; in this operation it was lanced by NASA on May 5th 2018 in order to discover the internal structure of Mars, with the aim of understanding better the formation of the solar system's planets.
The project started using CEA (in frozen equilibrium), using as input the combustion chamber pressure, its temperature, the exit to throat area ratio and the expansion ratio. CEA gives back different values referred to chamber, throat and exit of the nozzle that were the starting point for the sizing of the rocket.
Firstly, the Common Core Booster's performance parameters are evaluated in terms of thrust coefficient, specific impulse and nominal thrust. All of them are calculated in three different situations: optimal expansion, sea level and vacuum.
Only the sea level thrust has been fixed, in order to make the calculations. Then, the combustion chamber is sized in terms of volume, section length and, moreover, some thermochemical parameters are calculated too. After that the nozzle has been sized in terms of convergent and divergent lengths.
In particular, λ, which is a quantity that belongs to a range of values, linked to the losses of thrust due to the perpendicular component of velocity, is fixed in order to estimate the angle of convergence. Then the divergence angle is calculated.
Also the short tube injectors and the feeding systems are calculated and the propellant tanks are sized, by estimating pressure for each tank.
In the end, two other combinations of fuel and oxidant are examined, in order to compare them with the one used in Atlas V; the examination concerns an economy and ecological points of view through the study of the reachable thrust and TSFC.

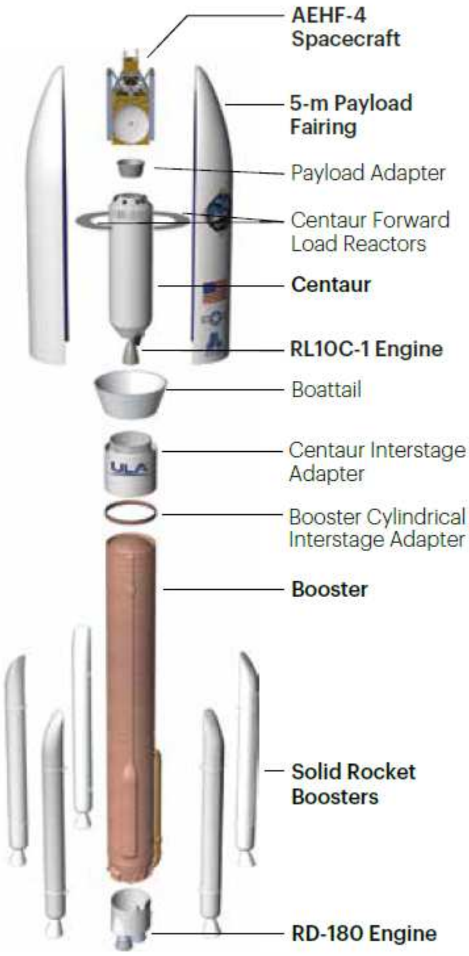


Figure 1.1: Configuration of the Atlas V 551

2
Notifications

In this project the following notification is used

Angles $[rad]$

δ	Angle after the impact (propellant-injectors)
β	Convergence angle
α	Divergence angle
ψ	Incidence angle

Non-dimensional quantities

Notation	Meaning
ϵ	Area ratio (exit to throat)
C_D	Discharge coefficient
$r_{L/D}$	Length to diameter ratio injectors hole
M	Mach number
n_h	Number of holes (injectors)
r	Oxidant to fuel ratio ox/f
c_T	Thrust coefficient
λ	Thrust correction factor
K_s	Safety coefficient
γ	Specific heat ratio

Physical quantities

Notation	Meaning	Unit of measurement
g_0	Acceleration due to gravity at sea level	$\frac{m}{s^2}$
A	Area	m^2
t_b	Burning time in combustion chamber	s
L^*	Characteristic length	m
c^*	Characteristic speed	$\frac{m}{s}$
ρ	Density	$\frac{kg}{m^3}$
D	Diameter	m
$\dot{m}$	Flow rate	$\frac{kg}{s}$
L	Length	m
m	Mass	kg

M_m	Molar mass	$\frac{kg}{mol}$
R	Molar weight specific gas constant	$\frac{J}{kgK}$
P	Pressure	bar
ΔP	Pressure losses due to the injection hole	Pa
a	Sound velocity	$\frac{m}{s}$
I_{sp}	Specific impulse	s
u	Speed	$\frac{m}{s}$
T	Temperature	K
F	Thrust	N
R_u	Universal gas constant	$\frac{JK}{mol}$
V	Volume	m^3
Q	Volumetric mass flow rate	$\frac{kg}{m^3}$

Subscripts

a	Atmospheric
c	Combustion chamber
e	Combustion chamber exit
t	Combustion chamber throat
$conv$	Convergent
div	Divergent
dyn	Dynamics
eff	Effective
$engine$	Engine (combustion chamber + nozzle)
$feed$	Feeding system
fin	Final
f	Fuel
h	Hole (injectors)
in	Initial
i	Injector entrance
lat	Lateral
lim	Limit (maximum)
ox	Oxidant
p	Propellant (oxidizer + fuel)
ra	Radius
0	Sea level
sh	Single hole (injectors)
vac	Vacuum

3
Data

Firstly the sea level thrust (F_0) is fixed, in order to evaluate the geometry of the rocket.
So from [1] and [3] the following data shown in the table are considered for the first stage.

For ρ_f and L^* different values have been evaluated; the table provides the set in which they commonly belong to, according to the engine and the propellant in object.

First stage CCB (liquid propellant)

Notation	Measure	Unit of measurement
F_0	3827	kN
F_{vac}	4152	kN
K_s	1.05	$-$
r	2.72	$-$
P_c	266.8	bar
T_c	3800	K
ϵ	36.87	$-$
t_b	253	s
ρ_f	$[0.81, 1.02]$	$\frac{kg}{m^3}$
ρ_{ox}	1141	$\frac{kg}{m^3}$
L^*	$[1.02, 1.27]$	m

4
Problem description

4.1 Description of the launch system

The following analysis of the first stage of Atlas V 551 occurs by calculating the most relevant performance parameters, by sizing the feeding lines and the pressurisation system.
The 551 configuration has two main stages, a Common Core Booster and a Centaur Upper Stage, which can make multiple burns to deliver payloads to a variety of orbits including Low Earth Orbit, Geostationary Transfer Orbit and Geostationary Orbit as well as Earth Escape Trajectories [1]. The engine used in the first stage is RD-180, while RL10-4-2 is the engine used in the Centaur Upper Stage from 2014. From 2014 they have used another type of engine in the first stage, which is RL-10C-1.
The first stage is 32.46 meters long and has a diameter of 3.81 meters. It can hold up to 284089 Kilograms of Rocket Propellant-1 and Liquid Oxygen, with an oxidizer to fuel ratio of 2.72 [1]; instead the upper stage Centaur has a diameter of 3.05 meters and is 12.68 meters long. It uses Liquid Hydrogen propellant and Liquid Oxygen as oxidizer.

4.2 Tasks and method of solution

The tasks are to size the Atlas V 551 and to determine the specifications of the engine.
The following procedure will be carried out:

- thermochemical characterization of the propellant combination, using the CEA, to evaluate the combustion chamber characteristics;
- determination of the engine geometric parameters, such as A_c, D_t, etc, by using the quantities found with CEA code (thermodynamic properties of the engine) and the initial data;
- with the expansion area ratio ϵ of the redesigned engine, the dimensions of the parts of the nozzle has been found, both the convergent and divergent ones;
- analysis of the injectors and sizing of the tank;
- pressurisation of the tanks with helium;
- a comparison between three different propellants in order to understand which is the best to maximize the thrust and to minimize the TSFC.

Throughout the whole analysis, two main software have been used to determine the thermochemical reactions and to size the different parts of the rocket, which are MATLAB and CEA. The codes are reported in the APPENDIX.

5
First stage sizing

5.1 Performance parameters calculations

The Common Core Booster has a RD-180 engine that features a dual combustion chamber and dual nozzle design; its fueling mixture is composed by kerosene (Rocket Propellant-1) and liquid oxygen as oxidizer. The engine is 3.56 meters long while its diameter is 3.15.
To size the combustion chamber the values of ϵ, M_e, M_m, γ, c^* have been obtained using CEA, initially setting the values of T_c, ϵ and P_c[1].
To acquire the following results the chemical reaction has been esteemed as theoretical, by assuming frozen composition.
The results are:

	Chamber	**Throat**	**Exit**	Unit of measurement
M_e	0	1	4.297	–
M_m	24.348	24.668	26.319	$\frac{1}{mol}$
γ	1.2028	1.2051	1.2527	–
P	266.70	150.27	0.55753	bar
T	3889.65	3529.42	1267.77	K
c^*	1775.0	1775.0	1775.0	$\frac{m}{s}$

The parameters that have been estimated with CEA are c_T, I_{sp} and T, each characterised by three different values that concern three dissimilar situations: the case of optimal expansion (opt), at sea level (SL) and vacuum (VAC). As previously stated, only F_0 has been fixed, in order to size the engine.
Then the following procedure has been implemented, by setting:

- the characteristics at the exit of the nozzle

$$a_e = \sqrt{\gamma_e R T_e}$$
$$u_e = M_e a_e$$
$$u_{lim} = \sqrt{\frac{2\gamma}{\gamma - 1} \cdot \frac{R_u}{M_m} T_c}$$
$$A_e = \frac{1}{2} \frac{F_0}{\rho_e u_e^2 + P_e - P_a}$$
$$D_e = \sqrt{\frac{4A_e}{\pi}}$$
$$\rho_e = \frac{P_e}{T_e \frac{R_u}{M_m}}$$

  where u_{lim} is the exit maximum velocity, assuming the previously declared combustion chamber thermochemical parameters; indeed, $u_{lim} > u_e$

to determine A_e the masses conservation has been used; moreover, the exit area has been esteemed by dividing in half F_0, since the first stage of the Atlas V ends with two nozzles. Indeed, the thrust used as initial data in this report is the sum of the two contributes provided by the two nozzles.
The results obtained are

a_e	736.4057	$\frac{m}{s}$
u_e	3164.3	$\frac{m}{s}$
u_{lim}	3628.8	$\frac{m}{s}$
A_e	1.5382	m^2
D_e	1.3995	m

- the geometric characteristics of the throat

$$A_t = \frac{A_e}{\epsilon}$$
$$D_t = \sqrt{\frac{4A_t}{\pi}}$$

the dimensions of which are

A_t	0.0417	m^2
D_t	0.2305	m

- the properties of the propellant

$$\dot{m}_p = \frac{P_c A_t}{c^*} \qquad m_p = \dot{m}_p t_b$$
$$\dot{m}_f = \frac{\dot{m}_p}{1+r} \qquad m_f = \dot{m}_f t_b K_s$$
$$\dot{m}_{ox} = \dot{m}_p - \dot{m}_f \qquad m_{ox} = \dot{m}_{ox} t_b K_s$$
$$V_f = \frac{m_f}{\rho_f} \qquad V_{ox} = \frac{m_{ox}}{\rho_{ox}}$$
$$\rho_{avg} = \frac{m_{ox} + m_f}{V_{ox} + V_f}$$

the results are

$\dot{m}_p$	626.8435	$\frac{kg}{s}$
m_p	$1.5859 \cdot 10^5$	kg
$\dot{m}_f$	168.5063	$\frac{kg}{s}$
$\dot{m}_{ox}$	458.3372	$\frac{kg}{s}$
m_f	$4.4764 \cdot 10^4$	kg
m_{ox}	$1.2176 \cdot 10^5$	kg
ρ_{avg}	$1.0765 \cdot 10^3$	$\frac{kg}{m^3}$

- the quality factors and the thrust at sea level and in optimal case

$$F_{opt} = u_e \dot{m}_p \qquad F_{vac} = u_e \dot{m}_p + A_e P_e$$
$$I_{sp,opt} = \frac{u_e}{g_0} \qquad I_{sp,vac} = \frac{F_{vac}}{g_0 \dot{m}_p}$$
$$c_{T,opt} = \frac{u_e}{c^*} \qquad c_{T,vac} = \frac{F_{vac}}{P_c A_t}$$
$$c_{T,0} = \frac{1}{2}\frac{F_0}{P_c A_t} \qquad I_{sp,0} = \frac{1}{2}\frac{F_0}{P_c A_t}$$

F_{opt}	$1.9835 \cdot 10^6$	N
$c_{T,opt}$	1.7827	–
$I_{sp,opt}$	322.5622	s
F_{vac}	$2.0693 \cdot 10^6$	N
$c_{T,vac}$	1.8598	–
$I_{sp,vac}$	336.5081	s
$c_{T,0}$	1.7198	–
$I_{sp,0}$	311.1718	s

5.2 Combustion chamber sizing

In this section the dimensions of the combustion chamber are determined, starting from the exhausted gases density ρ_c and M_c.
Since the value of M_c is not fixed, because it depends on the characteristics of the flow, a vector has been considered in order to contemplate all the possible values that M_c can generally achieve; therefore it swings between a minimum and a maximum, both related to theoretical situations. Hence L_c, D_c, A_c and u_c as functions of M_c are also not fixed, but represented as values that belong to sets.
In order to find V_c, the characteristic length (L^*) has been utilised. To determine it, the same reasoning adopted for M_c also applies to L^*, which values changes between 1.02 and 1.27. Through the definition of L^*, V_c then can be calculated as a function of A_t and L^*, as the following formulas show

$$\rho_c = \frac{P_c}{T_c \frac{R_u}{M_m}}$$
$$u_c = a_c M_c$$
$$A_c = \frac{\dot{m}_p}{\rho_c u_c}$$
$$D_c = \sqrt{\frac{4A_c}{\pi}}$$
$$V_c = A_t L^*$$
$$L_c = \frac{V_c}{A_c}$$

As stated, V_c and A_c have been assumed as values that belong to two different sets; hence to consider all the possible combinations for L_c, it has been assumed as a measure that belongs to a matrix of values, so it swings between a minimum ($L_{c,min}$) and a maximum ($L_{c,max}$). To easily represent all the cases the extremes have been reported:

$L_{c,min}$	0.3446	m
$L_{c,max}$	1.2871	m

Then the maximum and the minimum values of the lateral area of the combustion chamber (using $L_{c,max}$ and $L_{c,min}$ respectively) have been calculated as follows

$$A_{c,lat} = \pi D_c L_c$$

$A_{lat,max}$	1.6034	m^2
$A_{lat,min}$	0.2478	m^2

The values between which the Mach number swings both contemplate theoretical situations: they determine a volume too big or too small; in the first case it mainly requests a much higher weight and structural problems and in the second it may not inquire a combustion chamber long enough for chemical reactions to occur.
At the same time the desired Mach number tends towards its minimum possible value (0.2), in order to reduce V_c.

The following graphs represent how as M_c grows the velocity raises linearly while the geometric characteristics (A_c and D_c) decrease.

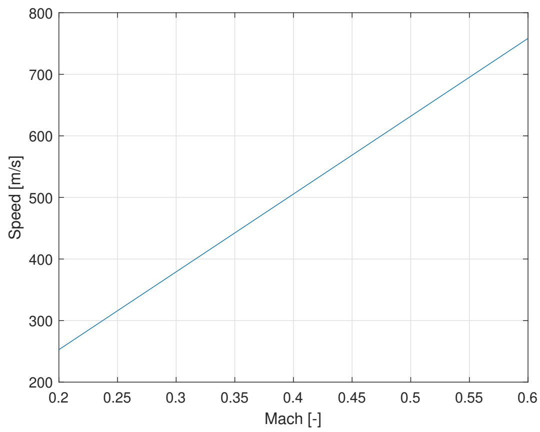


Figure 5.1: Speed in combustion chamber

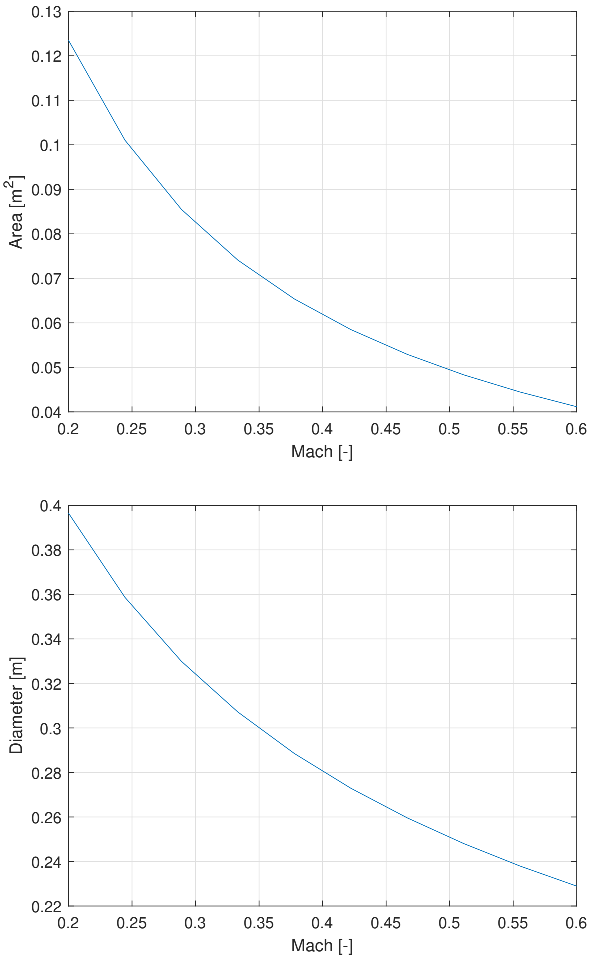


Figure 5.2: Area and diameter of the combustion chamber

5.3 Nozzle performances

The thrust produced by a nozzle is made up of two components which add to each other: pressure thrust and momentum thrust. Pressure thrust is the additional thrust gained due to the pressure difference between the nozzle exit plane (P_e) and the atmospheric pressure (P_a).

$$F = \dot{m}u_e + (P_e - P_a)A_e$$

where $\dot{m}$ is the mass flow rate.
To evaluate the "real" thrust the engine can develop for a simple and easily manufactured conical nozzle, a factor (λ) that provides the thrust correction has to be taken into consideration.
λ is found from the ration of thrust measurements with the corresponding ideal values and is multiplied onto the momentum flux term to adjust the ideal parameter. λ is used to predetermine performance ahead of testing, for preliminary designs and in health monitoring systems [7].
Also the nozzle angles variables α and β have been used, respectively the divergent and convergent ones.
The losses the engine can go through depends on a variety of non-ideal phenomena, such as friction, imperfect mixing and combustion, heat transfer, non-equilibrium, and two- and

three-dimensional effects [7].

The correction factor (λ) is always less than 1 for values of α greater than 0, since any deviation from perfectly axial flow will yield less thrust in the axial direction. A desirable result is λ approximatively equal to 1.
Hence to simplify the survey, in this section a simple conical nozzles is studied, according to the engine in object.

Since the values of α and β are generally not fixed but depend on a series of different factors, two different ways of analysis have been taken into consideration:

- the first one is about the determination of all possible results of the thrust.
  α and β belong to two dissimilar sets and the following formulas determine the range in which the thrust has to belong. To explicit the theoretical limits the maximum and the minimum allowable propulsive forces have been calculated.
  In order to do this, λ is not fixed but belongs to a set, as a function of α;

- the second one has been analysed in order to obtain a desired real thrust. λ now is fixed (assumed) and the curve trend of the thrust is within the maximum and minimum values calculated in the first item.
  In this item the desired effective ("real") thrust has been estimated by considering the 2% losses.

1. **Thrust range**
   Firstly, all possible values that the thrust can achieve have been analysed, by using the maximum ($D_{c,max}$) and the minimum ($D_{c,min}$) values of the combustion chamber diameter.
   α and β in this first part belong to two different sets, in order to find the maximum and the minimum lengths of the divergent and the convergent sections.
   Hence standard values have been contemplated for these angles, that swing between:

   $$\begin{array}{lll} \alpha & 12 \div 18 & deg \\ \beta & 30 \div 45 & deg \end{array}$$

   Then the following lengths have been found, using the vectors previously described:

   $$L_{div} = \frac{D_e - D_t}{2tan\alpha}$$
   $$L_{conv,max} = \frac{D_{c,max} - D_t}{2tan\beta}$$
   $$L_{conv,min} = \frac{D_{c,min} - D_t}{2tan\beta}$$

   and then the coefficient of losses and the effective thrusts (sea level and vacuum):

   $$\lambda = 0.5(1 + cos\alpha)$$
   $$F_{0,eff} = \lambda \dot{m}_p u_e + A_e(P_e - P_a)$$
   $$F_{vac,eff} = \lambda \dot{m}_p u_e + A_e P_e$$

   Fig. 5.3 shows all the results.

2. **fixed λ**
   λ is now assumed in order to permit 2% of losses and to establish fixed dimensions; hence in this part α has been calculated as a function of λ, while β and D_c have been assumed as follows as average values of their sets

λ	0.98	$-$
β	35	deg
D_c	0.3071	m

$$cos\alpha = 2\lambda - 1$$
$$L_{div} = \frac{D_e - D_t}{2tan\alpha}$$
$$L_{conv} = \frac{D_{c,max} - D_t}{2tan\beta}$$
$$L_{engine,max} = L_{c,max} + L_{conv} + L_{div}$$
$$L_{engine,min} = L_{c,min} + L_{conv} + L_{div}$$

With the chosen values the following results have been obtained

α	16.2602	deg
L_{div}	2.0040	m
L_{conv}	0.0548	m
$L_{engine,max}$	3.3458	m
$L_{engine,min}$	2.4033	m

The results have been plotted to compare them with the maximum and minimum solutions:

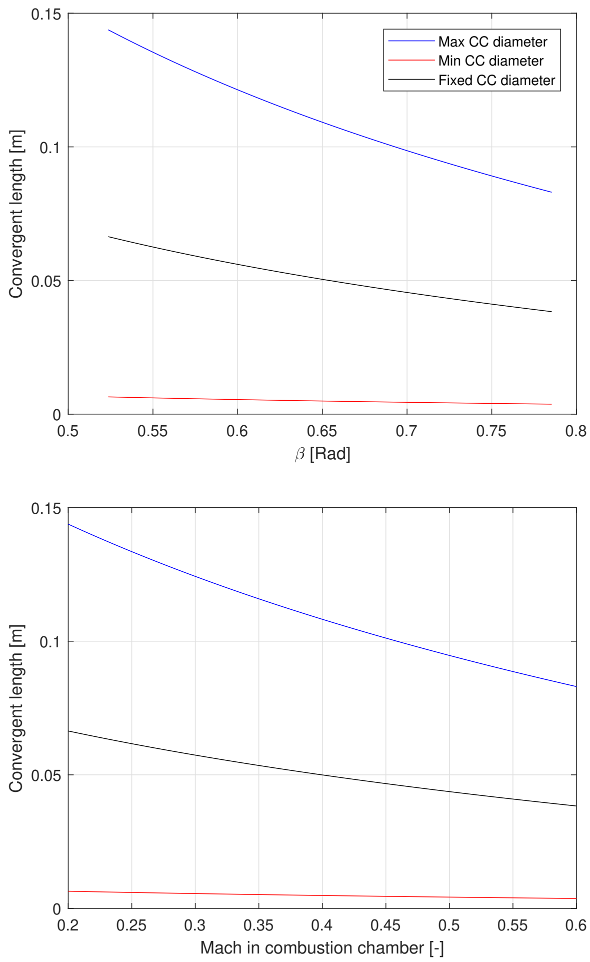


Figure 5.3: Convergent length due to β and to M_c

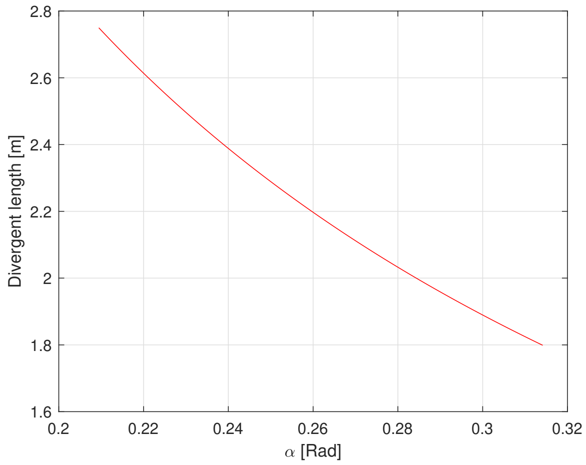


Figure 5.4: Divergent length due to α

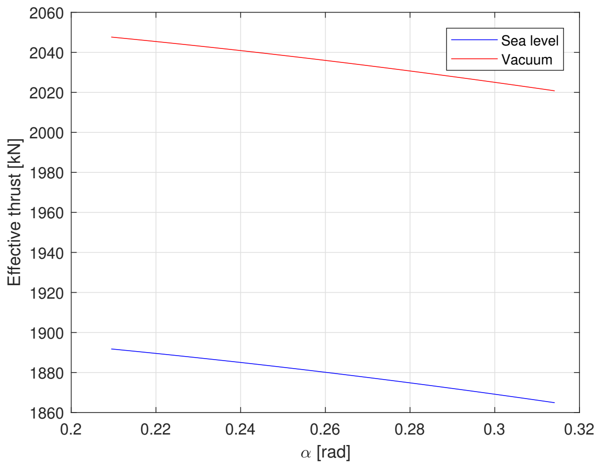


Figure 5.5: Effective thrust due to the losses in the divergent nozzle

5.4 Nozzle contour

By assuming the previously chosen values the engine has been plotted; the following graphs show approximately the combustion chamber, the divergent and the convergent parts of the nozzles.

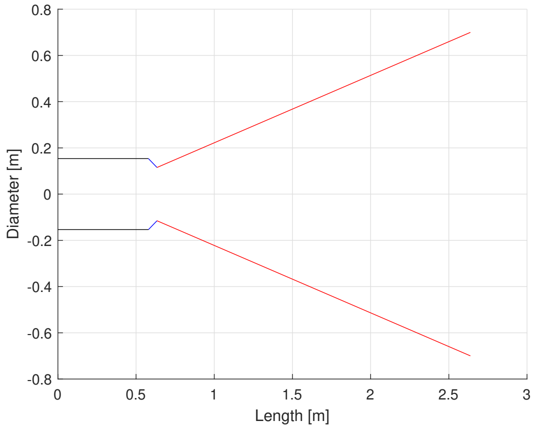


Figure 5.6: Representation of the engine (2D)

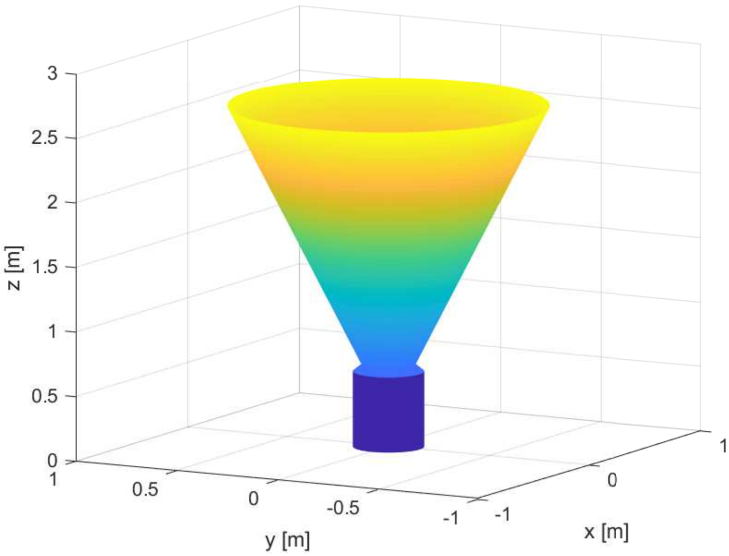


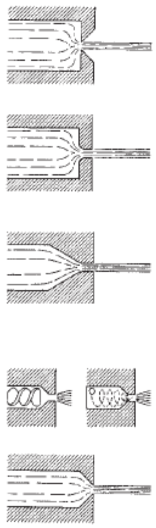
Figure 5.7: Representation of the engine (3D)

5.5 Injectors analysis

The quality of injectors in rockets influences the performances of nozzles. Indeed, a poor performance can cause a poor efficiency, by inducing unburnt propellant to leave the engine.
Injectors consist of small holes which aim jets of fuel and oxidiser so that they collide at a point in space a short distance away from the injector plate. This helps to break the flow up into small droplets that burn more easily [9].
The quality of this process can be improved by either using a superior propellant combustion, increasing the mass flow rate, positioning the angle of the hole or by reducing the size and increasing the number of orifices on the injector plate [8].

Injectors are schematized as a number of small holes arranged in patterns through which the fuel and oxidizer travel.
To analyse them the studied case consisted in a simple unlike doublet impinging-stream injection pattern. Moreover, the case of short tube with rounded entrance (length/diameter

<3.0) has been considered, as reported in Fig. 5.8.
A semi-cryogenic propellant (OX + RP-1) has been examined, hence δ is assumed null, in order to minimize the losses due to the undesired velocity components of the injected flow.



Orifice Type	Diagram	Diameter (mm)	Discharge Coefficient
Sharp-edged orifice		Above 2.5 Below 2.5	0.61 0.65 approx.
Short tube with rounded entrance $L/D > 3.0$		1.00 1.57 1.00 (with $L/D \sim 1.0$)	0.88 0.90 0.70
Short tube with conical entrance		0.50 1.00 1.57 2.54 3.18	0.7 0.82 0.76 0.84–0.80 0.84–0.78
Short tube with spiral effect		1.0–6.4	0.2–0.55
Sharp-edged cone		1.00 1.57	0.70–0.69 0.72

Figure 5.8: Injectors [7]

As shown in Fig. 5.8, the following characteristics have been assumed as:

$$
\begin{array}{lll}
C_D & 0.88 & - \\
D_h & 3.18 & mm \\
\rho_f & 810 & \frac{kg}{m^3}
\end{array}
$$

Moreover, the following definitions and assumptions have been used:

$$C_D = \frac{Q}{Q_{ideal}}$$

$$\Delta P = 15\% P_c$$

$$r_{L/D} = 3.0$$

With the following formulas the area of the holes through which the oxidant and the fuel travel has been determined. Also their number has been calculated, in order to obtain the area of the single hole and the velocities the fuel and the oxidant have through the tubes.

$$u_i = \sqrt{\frac{2\Delta P}{\rho_{avg}}}$$
$$A_{h,ox} = \frac{\dot{m}_{ox}}{C_D\sqrt{2\rho_{ox}\Delta P}}$$
$$A_{h,f} = \frac{\dot{m}_f}{C_D\sqrt{2\rho_f\Delta P}}$$
$$A_h = \pi\frac{D_h^2}{4}$$
$$n_{h,ox} = \frac{A_{h,ox}}{A_h}$$
$$n_{h,f} = \frac{A_{h,f}}{A_h}$$

Once $n_{h,ox}$ and $n_{h,f}$ are fixed, the minimum number of holes (n_h) is the smaller between them. Then the determination of the single hole area is carried out, in order to secondly calculate the velocities of the flows.

$$A_{sh,ox} = \frac{A_{h,ox}}{n_h} \qquad D_{sh,ox} = 4\frac{A_{sh,ox}}{\pi}$$
$$A_{sh,f} = \frac{A_{h,f}}{n_h} \qquad D_{sh,f} = 4\frac{A_{sh,f}}{\pi}$$
$$u_{ox} = C_D\sqrt{\frac{2\Delta P}{\rho_{ox}}} \qquad u_f = C_D\sqrt{\frac{2\Delta P}{\rho_f}}$$

ψ_{ox} has been assumed as 20 deg (0.3491 rad) and the value of ψ_f has been found as follows

$$\psi_f = asin\Big(\frac{\dot{m}_{ox}u_{ox}}{\dot{m}_f u_f}sin\psi_{ox}\Big)$$

The results are

u_i	70.39	$\frac{m}{s}$
$A_{h,ox}$	64.44	cm^2
$D_{sh,ox}$	0.473	cm
$A_{h,f}$	29.13	cm^2
$D_{sh,f}$	0.318	cm
u_{ox}	58.076	$\frac{m}{s}$
u_f	71.411	$\frac{m}{s}$
$n_{h,ox}$	(8.11) 9	–
$n_{h,f}$	(3.67) 4	–
ψ_f	49.16	rad

5.6 Feeding system and tank sizing

This rocket is bipropellant and uses a liquid hydrocarbon fuel (RP-1) and a liquid oxidant (O_2), stored separately; hence the expected volume of the tanks has to be relatively low for their reasonably high density.
The pressures related to the feeding system and propellant tanks have been calculated for the two flows (fuel and oxidant) by analysing different contributions: the pressure drop due

to the dynamic losses (ΔP_{dyn}), the one related to the combustion chamber (P_c), the losses of the feeding system (ΔP_{feed}) and the injectors (ΔP_{inj}).
The sum of these four contributes is the tanks pressure.

Having previously fixed P_c in the sections above, in order to explicit the remaining quantities two values have to be set: P_{feed} and u_{feed}, respectively the pressure of the feeding system and the velocity of the flow in object.
To set them, the overall tank pressure (fuel or oxidant) has been analysed as a function of P_{feed} and u_{feed}, in order to examine how much their values influence it.
As shown in Fig. 5.9 and 5.10, two graphs have been reported: the first one expresses the $P_{tank,ox}$ and $P_{tank,f}$ (respectively the pressures of the oxidant and fuel tanks) as functions of P_{feed}, the second one as functions of u_{feed}.
The tanks pressures have been calculated as follows

$$\begin{aligned}
\Delta P_{dyn,ox} &= \frac{1}{2}\rho_{ox} u_{feed}{}^2 \\
\Delta P_{dyn,f} &= \frac{1}{2}\rho_{f} u_{feed}{}^2 \\
\Delta P_{inj} &= 0.1 P_c \\
P_{tank,ox} &= P_c + \Delta P_{dyn,ox} + \Delta P_{feed} + \Delta P_{inj} \\
P_{tank,f} &= P_c + \Delta P_{dyn,f} + \Delta P_{feed} + \Delta P_{inj} \\
P_{tank,avg} &= \frac{P_{tank,ox} + P_{tank,f}}{2}
\end{aligned}$$

where P_{tank} has been expressed as a function of the P_c and the dynamic, the feeding system and injectors pressure losses respectively.

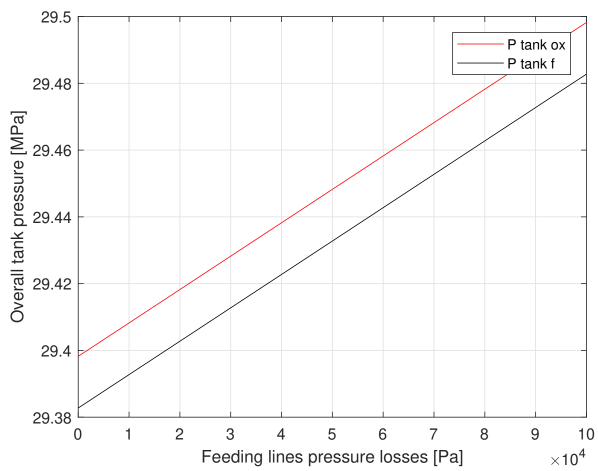


Figure 5.9: Overall tank pressure VS feeding lines pressure losses

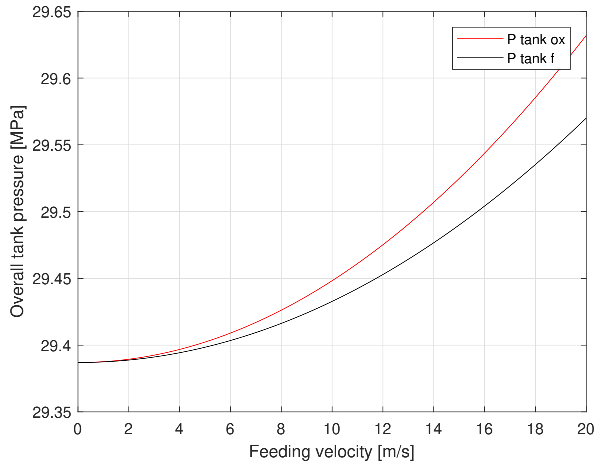


Figure 5.10: Overall tank pressure VS feeding velocity

Since the tanks overall pressures amount to about dozens MPa and the terms that are functions of P_{feed} and u_{feed} only to some kPa, as their values raise they do not cause an unbearable increase of pressure.
Indeed, both $P_{tank,ox}$ and $P_{tank,f}$ change lightly in the graphs as functions of these quantities, so P_{feed} and u_{feed} have been chosen within average values to avoid two limit cases; at the same time their modulus can not be null (that is a theoretical result of non-existent pressure losses) but also cannot be too high (because it develops unbearable pressure losses).

Hence they have been set as

$$\begin{array}{lll} u_{feed} & 10 & \frac{m}{s} \\ P_{feed} & 25 & kPa \end{array}$$

so the results are

$$\begin{array}{lll} P_{tank,ox} & 29.45 & MPa \\ P_{tank,f} & 29.43 & MPa \\ P_{tank,avg} & 29.44 & MPa \end{array}$$

5.7 Pressurisation system

The first stage propellants are held inside aluminium isogrid tanks; tank pressurization is computer-controlled and is accomplished with high-pressure helium that is stored in Helium Bottles on the Common Core Booster [1].
The pressurisation system requires the use of helium due to its inability to react with both oxidant and fuel.
Helium flows from the its spheres inside the LOX tank down to the engine compartment where it is heated up inside a heat exchanger connected to the hot gas flow from the Gas Generator to the LOX Boost Pump. The pressurized heated helium is then pressed into the LOX and Kerosene tanks to keep them at the proper pressure [1].

The helium coefficient of adiabatic expansion and other significant data for this section have been assumed as follows

γ_{He}	1.66	$-$
T_{tank}	290	K
$T_{He,in}$	350	K
ρ_f	915	$\frac{kg}{m^3}$
ρ_{He}	0.1785	$\frac{kg}{m^3}$

Some main quantities have been fixed through thermodynamics calculations, having previously set the final values of this system, as follows

$$T_{He,fin} = T_{tank}$$
$$P_{He,fin} = P_{tank,avg}$$
$$P_{He,in} = P_{He,fin}\left(\frac{T_{He,fin}}{T_{He,in}}\right)^{\frac{\gamma_{He}}{1-\gamma_{He}}}$$

$$V_{He,fin} = V_{ox}$$
$$V_{He,in} = \frac{T_{He,in}}{T_{He,fin}}\frac{P_{He,fin}}{P_{He,in}}V_{He,fin}$$
$$m_{He} = \rho_{He}V_{He,in}$$

$$ra_{He} = \left(\frac{3}{4\pi}V_{He,in}\right)^{\frac{1}{3}}$$

where the helium tank radius has been found by assuming spherical tanks.

The characteristics of the pressurisation system are

$P_{He,in}$	47.245	MPa
$V_{He,in}$	74.771	m^3
m_{He}	13.347	kg
ra_{He}	2.614	m

To obtain these results $T_{He,in}$ has been previously fixed as a common-used value. To verify its quality and to examine how the entire system can change as the temperature grows, $T_{He,in}$ has been secondly set as a value belonging to a set.
Hence, if it is not fixed but swings between two theoretical (a minimum and a maximum) values, the following graphs can be drawn, where all the quantities calculated beforehand have been represented as functions of the initial temperature of helium.
In Fig. 5.11 the He pressure without pressure losses coincides with P_c (which is a theoretical and unrealistic situation), although the minimum allowable He pressure is $P_{tank,avg}$.

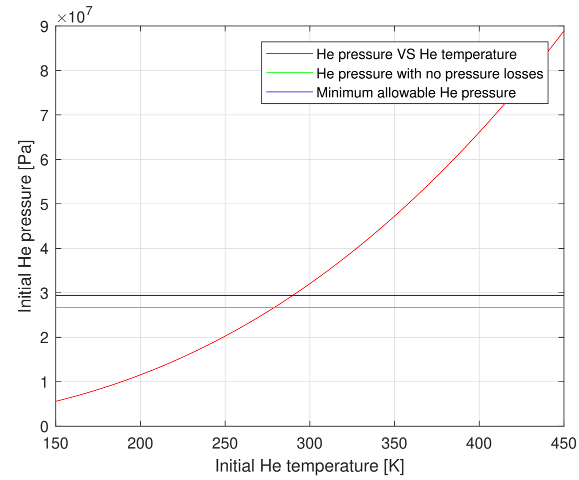


Figure 5.11: Initial He pressure VS initial He temperature

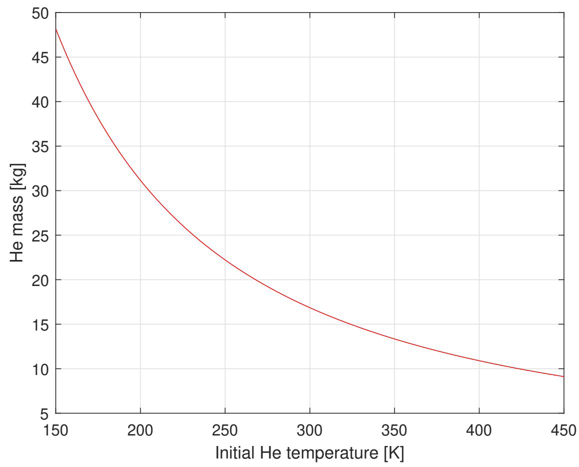


Figure 5.12: He mass VS Initial He temperature

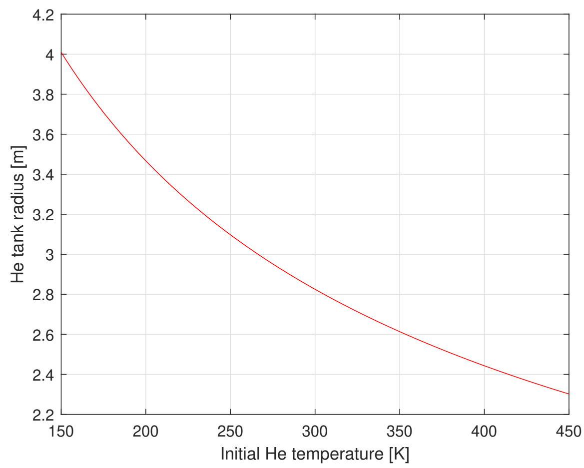


Figure 5.13: He tank radius VS Initial He temperature

6
Alternative F-OX compositions

In this section an analysis of the performances quality of Atlas V 551 first stage has been carried out; the examination has been evaluated in terms of the thrust and the TSFC (Thrust Specific Fuel Consumption), how the original propellant mixture and some other possible solutions effect them and which one eventually produces the better outcomes.
As previously reported, this engine uses Rocket Propellant-1 and Liquid Oxygen and the two other options examined are:

- H_2 (L) as fuel, O_2 (L) as oxidant;
- H_2 (L) as fuel, F_2 (L) as oxidant.

The geometry (A_c, A_e) and the pressure P_c have been kept constant in all cases, to be able to easily compare them by avoiding the analysis of three different nozzles and dimensions. CEA (which code has been reported in the appendix) has also been used to calculate thermodynamic parameters.
The following formulas use $H_2 - F_2$ as an example:

$$\dot{m}_{p,H2F2} = \frac{P_c A_t}{C^*}$$
$$F_{H2F2} = \dot{m}_{p,H2F2} u_e + A_e(P_e - P_a)$$
$$TSFC_{H2F2} = \frac{\dot{m}_{p,H2F2}}{F_{H2F2}}$$

The obtained results are:

F_{H2F2}	1856.4	kN
F_{RP1O2}	1913.5	kN
F_{H2O2}	1963.5	kN
$TSFC_{H2F2}$	0.8699	$\frac{kg}{h \cdot N}$
$TSFC_{RP1O2}$	1.1793	$\frac{kg}{h \cdot N}$
$TSFC_{H2O2}$	0.8677	$\frac{kg}{h \cdot N}$

$RP-1/O_2$: this propellant is the most used in the modern time. The thrust it can produce is remarkable, however the TSFC is not the most competitive one. Kerosene is one of the most common fuels used in rocket engines, but at the same time it is not the safest one: high quantity of kerosene causes health problems both for human and nature (it can lead to death). The advantage of this type of propellant is its cost. In USA it is around 7.46 \$/litre. Hence in the prospective to fill up the engine the final cost is not too elevated.

H_2/O_2 [12]: this combination is the most competitive one in terms of thrust and TSFC, but its cost is the higher: the amount is around 28.20 \$/litre (almost four times higher than the kerosene). However, scientists estimated that around 2027 the cost of liquid hydrogen will become cheaper than kerosene, considering that nowadays the "cost factor" is really important in order to realise a mission [10]. To underline that this type of combination is mainly used in the second stage is important, because both liquid hydrogen and oxygen are cryogenic. "To keep this combination from evaporating or boiling off, rockets fuelled with liquid hydrogen must be carefully insulated from all sources of heat, such as rocket engine exhaust and air friction during flight through the atmosphere. Once the vehicle reaches space, it must be protected from the radiant heat of the Sun. When liquid hydrogen absorbs heat, it expands rapidly; thus, venting is necessary to prevent the tank from exploding" [5]. Nevertheless this is the "greenest" environmentally friendly propellant and it contains more energy than the one that the kerosene has. The burning of hydrogen engines do not produce carbon dioxide and develops up to 80% less nitrogen emissions. Even though now the price is really high compared to the other propellants and there is also the cryogenic problem, in the future this combination can represent an important achievement to combine an eco-friendly and cheap mixture.

$\underline{H_2/F_2}$ [11]: the thrust performance with this combination is weaker than the other two, while TSFC is not. The cost of Fluorine (solid) is around 190/200 \$/grams. Fluorine is known as the most powerful oxidizing agent compared to all other elements but is extremely toxic; it is also hypergolic although it does not react with oxygen.
This element, in a huge amount, can lead humans to death and is very dangerous for the environment too. For example, fluorine was the key element to built the atomic bomb during World War II. So even thought it is a great oxidant, it is very dangerous.

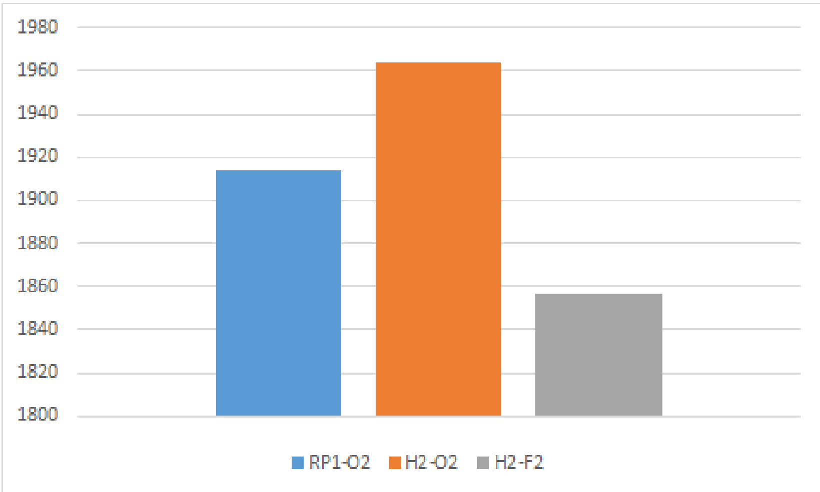


Figure 6.1: Thrust $[kN]$

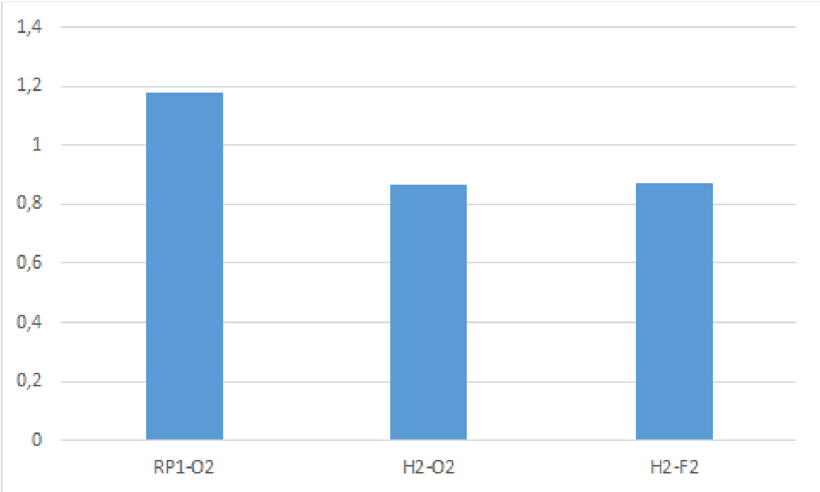


Figure 6.2: TSFC [$\frac{kg}{Nh}$]

A
Source Code

A.1 CEA

In this section the CEA codes of three different combinations of fuel and oxidant have been reported.
They concern, respectively:

1. H_2 as fuel and O_2 as oxidant;
2. $RP-1$ as fuel and O_2 as oxidant;
3. H_2 as fuel and F_2 as oxidant.

A.2 Matlab

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