Systems Analysis of a High Thrust, Low-Cost Rocket Engine

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ABSTRACT

This paper deals with the system-analytical aspects of a high thrust LOX-Kerosene engine to be used in expendable boosters. The attached reference space transportation system is a future Ariane 5 with cryogenic core and upper stage, but skipped solid rocket boosters.

The booster engine design is based on a gas generator cycle and features some of the low cost design parameters which were applied to the MC-1 engine. The investigations are focused on the feasibility and the effects of such an engine design when applied to booster propulsion of large launch vehicles such as ARIANE 5.

In its first part the paper reproduces numerically the internal (e.g. thrust chamber pressure, gg chamber pressure) and external (e.g. thrust, I_{sp}) performance characteristics of the MC-1 engine. Then engine cycle analyses are performed for a derivative of the MC-1 which is scaled up to meet the thrust requirements of large boosters.

A systems analysis is performed for a launch vehicle derived from ARIANE 5 but with two expendable LOX-Kerosene boosters instead of SRB's. The design payload capacity to GTO is in the 10 Mg class. The analysis includes trajectory simulation and optimization of the vehicle ascent. The relevant vehicle and rocket engine figures of performance, and mass are presented.

Subscripts, Abbreviations

EAP	Étage d'Accélération à Poudre
EPC	Étage Principal Cryotechnique (of Ariane 5)
ESC-B	Étage Supérieur Cryotechnique (of Ariane 5)
GLOW	Gross Lift-Off Mass
GTO	Geostationary Transfer Orbit
LRB	Liquid Rocket Booster
LOX	Liquid Oxygen
MECO	Main Engine Cut Off
RP(1)	Rocket Propellant (kerosene)
SRM	Solid Rocket Motor

1 INTRODUCTION

In addition to the driving factor of cost effectiveness, future launchers of the 21st century have to take into account the increasing demand on environmentally friendly propulsion systems. A big contributor to both cost and environmental issues is the first/booster stage propulsion of large launch vehicles, where high thrust levels are required and large amounts of propellant are expelled into the atmosphere. Large solid rocket boosters are a widespread choice for first/booster stage propulsion but often make use of propellant combinations that can be considered not environmental friendly. The use of a Lox-Kerosene propulsion system is a good and proven alternative choice allowing for high thrust levels, and among other advantages such as ease of operations, assures environmental friendliness. Regarding the cost effectiveness of liquid rocket engines a promising approach was made with the development of the MC-1 engine (formerly FASTRAC) by NASA MSFC. The MC-1 was the propulsion system foreseen to be used on the X-34 demonstrator vehicle. The engine design was driven by the desire to decrease engine complexity and cost.

2 ENGINE CYCLE ANALYSIS PROCEDURE

2.1 Program SEQ:

The computer program SEQ used for the cycle analysis presented in this paper, is based on a modular DLR (German Aerospace Center) code [2]. The modular aspect of the program allows for a quick rearrangement of the engine components, specifically the turbine and pumps assembly. After selection and suitable arrangement of the components in an input file, the program will calculate the fluids properties sequentially according to the respective components, which are passed by the fluid. Specific conditions can be linked with component data (i.e. the program is told to change the pump exit pressure to achieve a given chamber pressure).

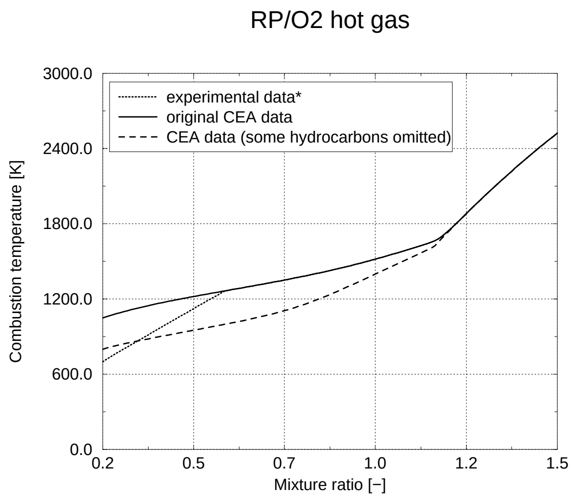
This results in a system of nonlinear equations (or rather dependencies), which is solved by an external numerical-mathematics subroutine[3].

2.2 Fuel Rich LOX-RP-Combustion Model

For combustion modeling the program SEQ generally relies on the CEA code developed by Gordon and McBride [1]. However for hydrocarbons in extreme fuel rich regimes (equivalence ratio around 10), the authors of this paper have noticed severe discrepancies between calculations made using the CEA code and tabulated data (e.g. [4]).

This is due to the fact, that CEA allows the cracking of long hydrocarbon chains into any theoretically possible smaller hydrocarbon and pure carbon and hydrogen molecules provided by the reactants database. In a highly fuel rich combustion, one has actually to expect a substantial amount of unburned chain molecules. CEA however will - if not forced to omit certain products - split the chains into ethane, methane and a big amount of graphite. This leads to higher calculated temperatures than those obtained experimentally. The correct combustion temperatures are of critical importance though, since they are the limiting factors for the pre-burner design. With current technology a maximum temperature of around 900 K is feasible as turbine entry- (i.e. pre-burner exhaust-) temperature.

In order to obtain realistic combustion temperatures, and corresponding fluid properties, the program SEQ runs two separate calculations of CEA in the extreme fuel rich regime. One is a standard program call, the other uses a severely reduced version of the thermodynamic properties database, which contains no graphite and no methane and ethane derivatives. Between these two sets, the properties used by SEQ are interpolated, in order to best reproduce the temperature data published in [4], see Figure 1. The properties of the unburned Rocket Propellant – 1 (kerosene) are obtained by using the method of LEE and KESLER, as described in [5]. For specific coefficients required by this method values taken from experiments with Jet A-1 at DLR [6] are used. This procedure is considered to be sufficiently accurate for systems analysis of kerosene burning rocket engines.



*[4]

Figure 1: Hydrocarbon properties in extreme fuel rich environment

2.3 Engine mass model

The engine mass is calculated by a combination of empirical and analytical methods. The latter are applied to the thrust chamber overwrap and ablative liner thickness using a preliminary design method. The procedure to determine the ablative liner thickness is described in chapter [4]. The liner thickness is assumed to be constant throughout the chamber and the throat section up to an expansion ratio of ε=2. From hereon the thickness is constantly decreasing towards the nozzle exit. The thickness is determined with an equation taken from [4]. The structural overwrap is assumed to utilize a high strength carbon fiber. Layer thickness is determined for the chamber loads assuming that only the fibres carry the loads.

The turbomachinery component masses, namely turbopumps, gas generator and injector are estimated using empirical methods developed at DLR [9,10].

2.4 Cooling calculation model

The cooling calculations for the investigated engine is based on a model described in [4] and consists of two parts. In the first part the film-coolant mass flow rate is calculated. Therefore the liner is assumed to be an non-ablative adiabatic wall. The maximum gas –side wall temperature is set to 2000 K which is the required value for the maximum allowable erosion rate of 0.051 mm/s. The gas –side heat transfer coefficient is given by the Bartz equation [4]:

$$h_g = \frac{41.8565}{d_t^{0.2}}\left(\frac{\mu^{0,2}cp}{Pr^{0,6}}\right)\left(\frac{p_c g}{c^*}\right)^{0.8}\left(\frac{d_t}{r}\right)^{0.1}\left(\frac{A_t}{A}\right)^{0.9}\sigma$$

where

h_g	=	Heat transfer coefficient	W/m-K
d_t	=	Throat diameter	m
μ	=	viscosity	Pa s
cp	=	specific heat at const. pressure	J/kg-K
Pr	=	Prandtl number	
P_c	=	chamber pressure	bar
g	=	gravitational constant	kg/m
c^*	=	characteristic velocity	m/s
r	=	throat radius of curvature	m
A_t	=	throat area	m^2
A	=	area along chamber axis	m

with the factor σ being:

$$\sigma = \frac{1}{\left(\frac{T_{wg}}{2T_c}\left(1+\frac{\kappa-1}{2}M^2\right)+\frac{1}{2}\right)^{0.68}\left(1+\frac{\kappa-1}{2}M^2\right)^{0.12}}$$

where

T_{wg}	=	gas-side wall temperature	K
T_c	=	nozzle stagnation temperature	K
M	=	local Mach number	

A MC-1 film-cooling analysis [7] was performed assuming a gaseous state of the fuel. Accordingly this is also assumed for this study. The required film-coolant mass flow is calculated with [4]:

$$G_C = \frac{h_g}{-\ln\left(\frac{T_{AW}-T_{WG}}{T_{AW}-T_{CO}}\right)c_p\eta_{CO}}$$

with:

G_C	=	mass flow rate per unit area	kg/m^2-s
c_p	=	specific heat at const. pressure of gaseous coolant	J/kg-K
T_{CO}	=	Initial coolant temperature	K

From G_C and the chamber surface the film-coolant mass flow rate can be calculated. Finally a calculation is performed to determine the required thickness of the ablative liner. Therefore the charred depth of the liner is calculated with [4]:

$$t_{char} = c\left(\frac{2kt_b}{R_r R_v C_p \rho}\ln\left(1+\frac{R_r R_v C_p\left(T_{AW}-T_d\right)}{L_p}\right)\right)^{0.5}\left(\frac{p_c}{100}\right)^{0.4}$$

with:

t_{char}	=	Char depth	m
c	=	Correction factor	-
R_r	=	Weight fraction of resin in the ablative material	-
R_v	=	Weight fraction of pyrolized resin vs. total resin content R_r	-
C_p	=	Heat capacity of pyrolized gases	J/kg-K
ρ	=	Density of liner	kg/m^3
k	=	Heat conductivity of char	W/m^2-K
t	=	Thrust chamber firing duration	s
L_p	=	Latent heat of combustion	J/kg
T_{AW}	=	Adiabatic wall temperature of the gas	K
T_d	=	Decomposition temperature of resin	K

3 MC-1 ANALYSIS

In a previous internal study the program SEQ has been used to simulate the MC-1 performance based on data published in [8]. The main purpose was to validate the calculation model, and to better understand the special design features of the US-engine.

For the simulation the following parameters of the MC-1 were taken as input values. The condition in front of the turbopump, namely pressure, temperature and mass flow. Furthermore the chamber pressure and mixture ratios of the main and the gas generator chamber. Table 1 shows a comparison of the main SEQ results and the MC-1 data.

For the MC-1 10% of the total fuel flow into the chamber is required for film cooling which corresponds to 2.65 kg/s. The above described method to determine the film-cooling flow rate is applied to the MC-1 and results in a coolant flow rate of 2.95 kg/s.

The results of the simulation show a good agreement with the main MC-1 performance parameters. This is also true for the results of the film-cooling calculation. The predicted coolant flow rate is slightly higher, thus giving a good margin for further calculations. As a consequence the procedure is seen to be applicable to the booster engine under study.

	SEQ	**MC-1 [8]**	
Vacuum thrust	283.26	284.41	N
Vacuum Isp	314.25	314.00	s
Total mass flow	91.92	91.90	kg/s
Engine MR	2.17	2.17	-
GG pressure	38.17	39.64	bar
GG temperature	844.41	888.89	K
GG mass flow	3.23	3.22	kg/s
Throat diameter	0.21	0.22	m

Table 1: MC-1 simulation results

4 SYSTEMS ANALYSIS OF LOW-COST ENGINE

4.1 Description of the Baseline Launch Vehicle

The regarded expendable space transportation system consists of two LRBs attached to the Ariane 5 core stage (EPC). This stage is powered by a single Vulcain 2 engine, which is currently under development. A new cryogenic upper stage (ESC-B) is already in the pre-development phase. It should include a new advanced expander cycle motor of 180 kN class (VINCI) by 2006. This study assumes a total propellant load of this stage around 27000 kg. The LRBs are foreseen to be mounted in the same way as the present EAPs, to avoid any major structural change of the core stage.

4.2 General LRB Engine design considerations

Some of the engine design considerations stem from the application of the engine as the propulsion device for a booster attached to an already existing system. This means that parameters like maximum dynamic pressure, maximum acceleration, separation conditions are predefined by the core system. Furthermore, because a high cost efficiency is required some engine characteristics such as high chamber pressure, high turbomachinery efficiency, deep throttling capability which are favorable from the performance point of view are sacrificed for the sake of simplicity and decreased technology demand. In the following some of the design considerations that led to the current design of the engine and the stage shall be presented.

The thrust level is selected based on a minimum launch vehicle acceleration of 1.25 g at lift-off, the maximum allowable axial acceleration of the core stage of 4.5 g. A major throttling capability of the engine is precluded. Initial performance calculations showed, that a vacuum thrust of around 4900 kN is required for each booster stage. This is about 3.6 times the thrust of the Vulcain 2 engine, and is felled to be too demanding to be realized in Europe with a single engine thrust chamber in the foreseeable future. Therefore, either a two chamber or a two engine solution can be considered. The turbomachinery requirements of a two-chamber or single-chamber engine are quite comparable but still require high turbopump and turbine performance. A two engine design requires an increase in thrust of 80 % compared to the Vulcain 2. Additionally the two engine designs increases reliability because of the added engine out capability. As a consequence the application of two engines with 2445 kN of thrust each was selected for the LRB. The engine utilizes a gas-generator cycle with a fuel-rich gas generator with an ablative thrust chamber and additional fuel film cooling. To our knowledge there is no ablative cooled liquid rocket engine that has been operated above 68 bars. Therefore a lower chamber pressure of 65 bar has been selected.

4.3 Preliminary Design of the Engine

The preliminary design of the LRB engine was performed based on the methods and assumptions as mentioned above. A schematic overview of the considered engine cycle is shown in Figure 3. For the cycle analysis the following assumptions were made regarding the efficiencies of the engine components. The turbopump efficiency is set to be 0.7 and for the turbine 0.4. A combustion efficiency of 0.975 was assumed. The design results in a vacuum thrust of 2445 kN , a mixture ratio of 2.4 for the thrust chamber and 0.3 for the gas generator. A nozzle expansion ratio of 15 was selected resulting in a sea-level thrust of 2118 kN. The values for the specific impulse are 301 s under vacuum conditions and 261 s at sea-level conditions. A summary of major engine performance parameters is listed in Table 2.

The film-cooling calculation showed that a coolant flow of around 14 kg/s is required. This is about 6% of the fuel flow entering the combustion chamber. For the determination of ablative liner thickness a total life burn duration of 300 seconds is assumed. This value accounts for the nominal burn time of the engine, the required acceptance test(s) and a safety factor. For the throat region the total liner thickness was calculated to be 20 mm.

The dimensions of the chamber/nozzle assembly were determined assuming a cylindrical thrust chamber with a characteristic length of 1.3 m. The selected nozzle contour is based on truncated ideal type. A contour plot of the assembly is depicted in Figure 2 and a detailed list of the geometry is listed in Table 3.

A preliminary estimation of the turbomachinery layout was made. It is proposed to utilize single stage radial pumps in a single-shaft arrangement with the turbine. The turbopump rotational velocity is estimated to be around 10000 rpm. A single stage supersonic turbine is used for reasons of cost efficiency. No detailed layout of the injector has been made yet. The combustion stability will have to be addressed in an actual engine sizing. A engine mass estimation has been made resulting in an overall mass of 3071 kg corresponding to a thrust to weight ratio of 81. Some important engine component masses are listed in Table 3.

Performance		
Sea-level Thrust	2118.14	kN
Vacuum Thrust	2444.41	kN
Sea-level Isp	260.83	s
Vacuum Isp	301.00	s
Thrust chamber pressure	65.00	bar
Thrust chamber temperature	3453.97	K
Thrust chamber MR	2.40	-
Film coolant mass flow	13.78	kg/s
Total mass flow	832.08	kg/s
Engine MR	2.17	-
C^*	1687.38	m/s
RP-1 pump power	4.06	MW
LOX pump power	6.21	MW
GG chamber pressure	56.88	bar
GG chamber temperature	886.62	K
GG mass flow	38.095	kg/s
GG MR	0.30	-

Table 2: Technical data of LRB engine

Dimension and masses		
Char. Chamber Length	1.30	m
Chamber Length	0.81	m
Nozzle Length	2.64	m
Total Length	3.45	m
Throat Radius	0.26	m
Exit diameter	2.02	m
Contraction Ratio	1.80	-
Expansion Ratio	15.00	-
Chamber/nozzle assembly	1147	kg
Turbomachinery	1671	kg
Misc.	254	kg
Total	3071	kg

Table 3: Dimensions and estimated component masses of the LRB engine

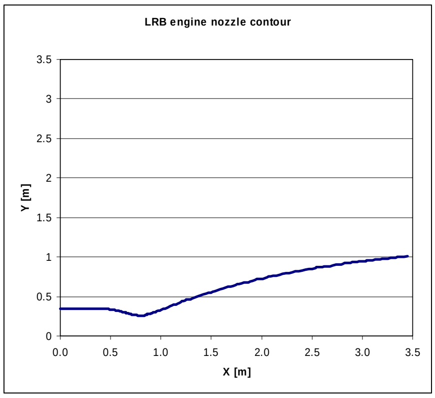


Figure 2: LRB nozzle contour data

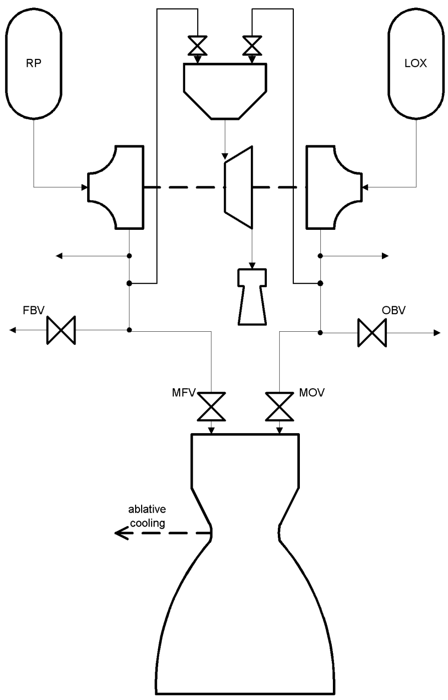


Figure 3: LRB engine cycle diagram

4.4 Liquid Rocket Booster stage

A preliminary design of the expendable liquid booster is performed in this study. The requirement to attach the LRBs to an unchanged Ariane 5 core determines the stage diameter of 3.74 m. The LRB is made of two integral tanks, separated by an intertank structure with the LOX tank in the forward position. The two attachment points with the core stage are located above, respectively below the tanks. The aft section of 4.1 m length, contains the stage attachment ring, the thrust structure, the engine assembly, and an aft-skirt. This section also houses the helium pressurization tanks, fuel supply and the TVC system. The launch vehicle is supported on the launch pad by the aft skirt of about 4.5 m diameter as with Ariane 5 EAP.

Liquid Rocket Booster	GTO mission
overall length	30.2 m
tank diameter	3.6 m
skirt diameter	4.5 m
	mass [kg]
mass of the structure: (including margins)	732
mass of the subsystem: (including margins)	4368
mass of the propulsion: (including margins)	6757
mass of thermal protection: (including margins)	339
stage mass empty (incl. marg.):	18696
residuals and reserves:	4000
ascent propellant:	204407
GLOW stage mass:	227103

Table 4: Dimensions and estimated component masses of one twin LRB with two engines

4.5 Launch vehicle performance GTO

For the Ariane 5/ LRB launch vehicle the performance into GTO (180 km x 35786 km, i = 7°) is determined. The total lift-off mass of the LV is 690 t and the T/W ratio at lift-off is 1.25. During the booster phase the AOA is held close to zero. The maximum dynamic pressure experienced by the vehicle is 42.5 kPa and the maximum acceleration is 4.45 g. The latter values are quite close to those of the reference system. Separation of the LRB takes place after 121 s in 48 km altitude at a velocity of 1.85 km/s. The resulting payload mass is 10690 kg. Some of the important parameters of the ascent trajectory are shown in Figure 4.

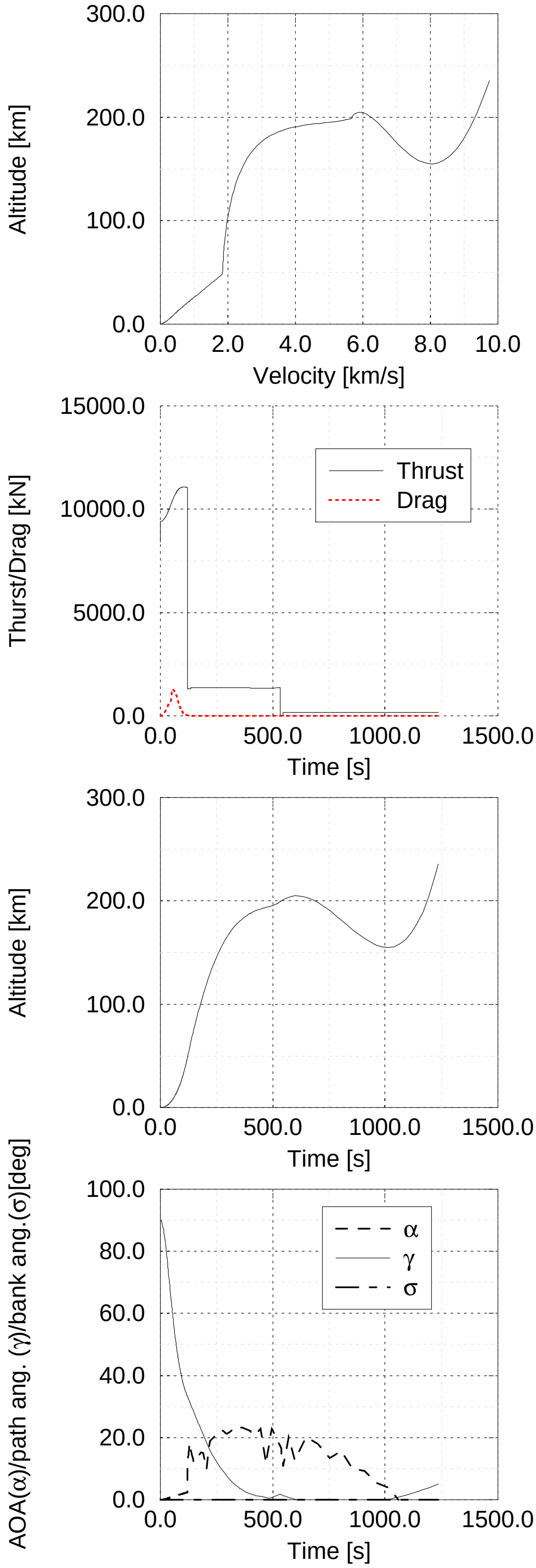


Figure 4 Performance parameters along the Ariane5 / LRB ascent trajectory

5 CONCLUSION

This study investigates expendable liquid rocket boosters which are attached to the expendable Ariane 5 core stage of an advanced future derivative. The boosters utilize environmental benign propellant combination of LOX and RP-1. A low cost, high thrust rocket engine is proposed for the boosters. The low-cost features are derived from those which were applied to the MC-1 engine, developed for the X-34 by MSFC.

The study shows that a high thrust of around 4900 kN is required for this application. Such a thrust level for a single chamber engine is treated to be out of reach for Europe in the foreseeable future. Therefore the use of two engines with 2445 kN thrust each was investigated in further detail. The resulting LRB stage has a total mass of 227 t and a propellant load of 208 t. The trajectory performance analysis showed, that the considered Ariane 5 derivative with the LRBs is able to deliver a payload of 10.7 t into GTO which is in the same class as the reference system.

The investigated engine concept has some design features that are cost efficient when compared to other first stage engines. It utilizes a gas generator cycle with a relatively low chamber pressure and low to moderate efficiencies on the turbomachinery side. The applied thrust chamber cooling technique is demonstrated and less demanding than regenerative cooling. But, such a technique has not been applied to an operational engine of this thrust level. Furthermore, a change in the development test procedure might be required. This is mostly due to the fact that the life time of the thrust chamber is defined by the ablative liner. Hence, several thrust chambers have to be used during the development phase.

The question whether the low cost manufacturing methods used for the MC-1 can be applied to the investigated engine can only be answered from an industrial point of view.

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Further updated information concerning the SART space transportation concepts is available at:
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