4th International Conference on Launcher Technology "Space Launcher Liquid Propulsion"

3-6 December 2002 – Liège (Belgium)

Comparative Study of Kerosene and Methane Propellant Engines for Reusable Liquid Booster Stages

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COMPARATIVE STUDY OF KEROSENE AND METHANE PROPELLANT ENGINES FOR REUSABLE LIQUID BOOSTER STAGES

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Kerosene and methane are two promising candidate propellants for a future reusable booster stage. This study assesses the merits of both propellants and compares their respective performance when used in a booster stage. The identification of the propellant properties is the starting point. An analysis of a staged combustion cycle engine for both propellants follows. The final assessment is based on the results of a performance analysis of a launch vehicle making use of these motors in reusable fly-back boosters.

Nomenclature

GLOW	Gross Lift-Off Weight
L/D	Lift/Drag Ratio
LFFB	Liquid Fly Back Booster
LH2	Liquid Hydrogen
LNG	Liquefied Natural Gas
LOX	Liquid Oxygen
MECO	Main Engine Cut Off
RCS	Reaction Control System
sfc	Specific Fuel Consumption
s/l	Sea-level
SLI	Space Launch Initiative
SRM	Solid Rocket Motor
TET	Turbine Entrance Temperature
T/W	Thrust/Weight Ratio
vac	Vacuum

1 INTRODUCTION

1.1 Background

A replacement of the solid booster stages EAP by liquid propellant boosters is envisioned in the future as an evolution of the European Ariane 5 launch vehicle. This step should provide for performance gains, reduced ecological impacts and possibly reduced costs. Potential propellant candidates include therefore LOX/LH2, LOX/Kerosene and LOX/Methane. Extensive experience with LOX/LH2 rocket motors was acquired in Europe through the development and utilization of the HM-7 and Vulcain engines and advantages as well as problems with this propellant combination are well understood. In contrast, no recent experience exists for hydrocarbon propellants in Europe. Practical experience is very limited and was mainly gained back in the 60's when for example Rolls Royce built a LOX-Kerosene motor for the Europa Rocket.

In view of upcoming programmatic decisions, several papers were published in recent years by European industry and research centers, evaluating the merits of kerosene and methane as propellants [1]-[4]. The focus of these studies was mostly on the propulsion system itself.

1.2 Objectives

Within the framework of this study, a system analysis is performed for two prospective liquid fly-back boosters for Ariane 5 using LOX/Kerosene respectively LOX/Methane propellant. The objective of the study is confined to the comparison of these two prospective propellants. We neither intend to make the case for liquid booster propulsion in general nor for (partialy) reusable launch vehicles.

In a first step, the thermodynamical and chemical properties of the propellants are compared. Problems of availability and supply are addressed as well. In a next step, suitable high-thrust engines are selected among existing motors or defined based on published data of prospective rocket motors if no adequate model is available. A staged combustion cycle is chosen for both motors as the effects of modeling uncertainties of the motor for this specific cycle have only very little effect on the launch vehicle performance calculations. A rocket motor cycle analysis is performed for both chosen motors.

Distinct reusable booster stages are designed for both propulsion options. A number of design parameters are kept constant to enable a meaningful comparison in the course of the study. They include the vacuum thrust of the motors and the length of the booster. The latter constraint is necessary to limit the effects on the interface of the Ariane 5 core stage.

Computer models of the booster are created for both concepts, respecting the distinct characteristics of the propellants. Mass and inertia properties of the booster are estimated based on structural load assumptions in conjunction with the chosen geometry. Theoretical-empirical methods permit individual estimation of aerodynamic coefficients of each launcher configuration. These data sets serve as input for a trajectory optimization to calculate the achievable payload performance and for return flight simulations. The booster configurations are refined through iterative design loops.

The comparative assessment of the suitability of the propellants for proposed future reusable boosters will be based on these results. An analysis of sensitivity complements the result.

2 PROPELLANT PROPERTIES

Kerosene, a blend of different hydrocarbons, is a common propellant for launcher applications. It has been widely used in the US (RP-1) and in Russia (T(S)-1) since the early age of liquid propulsion. Its high propellant density enables a compact design of turbomachinery and minimal stage sizes.

Methane or LNG are alternative propellants, that are being considered only recently. Advantages in comparison to kerosene include amongst other things higher specific impulse, lower pressure drop in cooling channels, superior cooling properties, higher coking limits and less soot deposition [5], [6]. The latter two are especially important in the context of reusability. Methane is however a "soft" cryogenic propellant with a storage temperature of about 111 K. This temperature is in proximity to LOX and can enable under favorable circumstances a simplified architecture. There exists no flight experience for the LOX/methane propellant combination so far. However, extensive research and ground testing has been performed in Russia and Japan in recent years [6], [7].

Some basic thermodynamical properties of the two propellant options are summarized in the following table.

		RP-1	T(S)-1	Methane
Boiling point (1 bar)	K	450-547	466-547	112
Freezing Point	K	224	226	91
Density @ 16 °C	kg/m³	809	836	0.72
Density (liquid) @ boiling point	kg/m³			422.5
Kinematic Viscosity (liquid)	mm²/s	3.02 @ 274 K	4.01 @ 274 K	0.28 @ 111 K
Critical Temp.	K	662	658	190
Critical Pressure	Pa	2 171 848	1 820 000	4 599 200
Specific Heat Capacity	J/(kg K)	2093	1980	3480
Specific Energy	MJ/kg	43.34	43.13	50
Volume specific energy (liquid)	MJ/m³	34 934.14	35 887.02	21 125.00
Coking limit	K	560	?	950
Handling properties		Storable	Storable	Cryogenic
		$CH_{1,952}$	$CH_{1,946}$	CH_4
Molecular mass	kg/kmol	172	167	16.043

Table 1 Thermodynamical Properties [4],[8]-[10]

The theoretical performance values under standard conditions of both propellants in combination with LOX are given in Table 2.

According to [6], acquisition cost of methane is about three times smaller than for kerosene and long-term availability is forecasted to be considerably higher. Considering current overall launch costs and launch rate, this seems however to be only of moderate importance. Likewise, the easier ground handling and storage of kerosene as compared to the cryogenic methane is of minor significance.

Combustion with LOX		RP-1	Methane
sea level, optimum expansion, chamber pressure 6.89 MPa			
max. Isp	s	300.1	309.6
mixture ratio	-	2.58	3.21
chamber temperature	K	3676	3533
bulk density	kg/m³	1030	820
characteristic velocity	m/s	1799	1857
vacuum expansion, ε=40, chamber pressure 6.89 MPa			
max. Isp	s	358.2	368.9
mixture ratio	-	2.77	3.45
chamber temperature	K	3701	3563
bulk density	kg/m³	1030	830
characteristic velocity	m/s	1783	1838

Table 2 Theoretical Performance [11]

3 ENGINE PERFORMANCE ANALYSIS

A liquid fly-back booster for the Ariane 5 core vehicle must provide in the order of 4000 kN of thrust. A RD-180 class engine fulfills this requirement. The selection of the RD-180 kerosene engine in this study should not be misunderstood as a commitment to this motor. It rather stands for a performance target for a suitable engine class.

The two propellants differ in a number of operational aspects. Some key aspects with regard to rocket motors are coking, sooting and corrosion.

Several publications such as [18] suggest that coking is of no particular concern when using methane as propellant. Furthermore, sooting does not represent a problem either [3] and thus all engine cycles are principally feasible. Copper corrosion due to sulfur impurities is addressed in [20]. A technical solution to obtain high purity methane is workable according to the authors. These above mentioned statements favor a methane motor especially in view of reusability aspects.

On the other hand, "no maintenance, flushing or degreasing" was required between test runs for the Aerojet AJ26-58 LOX/RP-1 motor according to [20]. The kerosene RD-170 engine is qualified for 10 flights [21] and all proposed U.S. hydrocarbon SLI motors use kerosene propellant [22].

3.1 Kerosene engine

The NPO Energomash / Pratt & Whitney RD-180 is one of the most advanced liquid rocket engines with the highest chamber pressure ever flown. It is in operation with Lockheed Martin Atlas 3 and since recently with Atlas 5. It is a staged combustion cycle engine with oxygen-rich preburner. A single turbine drives the single-shaft main oxidizer and propellant turbopumps. Boost pumps are used for both fuel and oxidizer to improve turbomachine efficiency. The rocket motor cycle was modeled using the SEQ program of DLR.

The high chamber pressure of the RD-180 motor can only be achieved using powerful turbomachinery. As the turbine entrance temperature has to stay within workable limits (776 K), a high preburner pressure is required. 53.7 MPa is the necessary value according to SEQ simulations. Considering the substantial pressure loss at the preburner injection, a discharge pressure of 80.9 MPa is to be furnished by the high pressure fuel turbo pump. High efficiency levels have to be reached for each turbomachine element to enable this powerful rocket motor.

The obtained fluid properties are in good accordance with published data for the RD-180 engine [12] and the closely related RD-170 / RD-191 engines [6].

3.2 Methane Engine

The conceptual 'SE-12' methane engine of DLR-SART uses the same engine cycle as the RD-180. Previous studies (e.g. [3]) have shown that such a single shaft turbopump arrangement is feasible for a methane engine.

In contrast, simulations have shown that a fuel-rich staged combustion cycle is not feasible under realistic assumptions for an engine with such high combustion chamber pressure. The fuel pump exit pressure would have to be well beyond 100 MPa to keep the TET at a reasonable value.

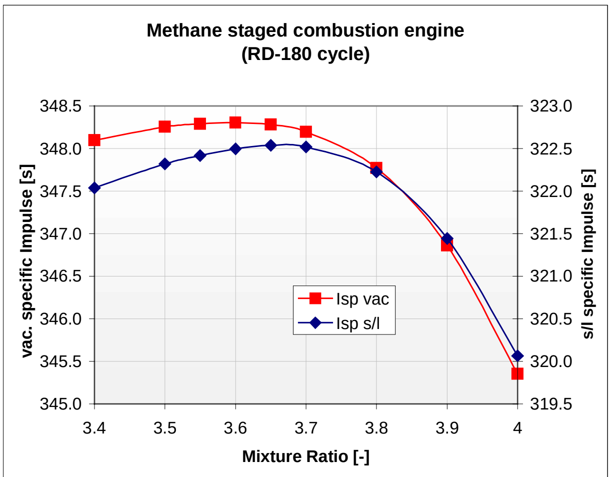

Figure 1 Mixture Ratio vs. Specific Impulse for 'SE-12' Methane Engine

Key parameters like turbine entrance temperature, combustion chamber pressure, expansion ratio and vacuum thrust of the SE-12 motor are kept constant in comparison to the RD-180. This should provide for similarity with respect to reusability issues on the one hand and performance on the other hand. The pressure loss in the cooling channels of the combustion chamber were assumed to be smaller than for the RD-180 engine, as commonly suggested (e.g. [5]). Some other parameters as for example pressure losses in general and efficiency factors are unchanged due to a lack of detailed information. These uncertainties entail an incertitude concerning the turbomachinery sizing and hence the motor mass. Its influence on the performance calculation of the launcher is however limited to the effects of a different lift-off mass, since staged combustion cycle engine performance only depends on main combustion chamber conditions and expansion ratio.

The mixture ratio for the selected chamber conditions was determined by a parametric variation. The near optimum value of RM = 3.6 was eventually chosen. This value corresponds to the maximum engine specific impulse in vacuum. A higher mixture ratio leads to lighter boosters due to the higher density of liquid oxygen. This potential performance benefit is sapped however by the sharply decreasing engine Isp. The influence of the mixture ratio on payload performance is shown in Figure 2.

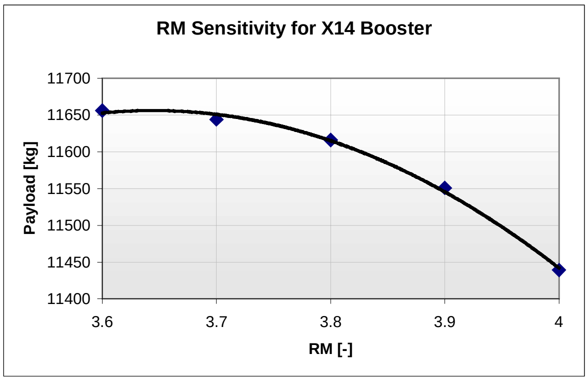

Figure 2 Influence of 'SE-12' mixture ratio on payload

The engine mass was estimated based on values given in [6]. This rule of thumb method provides obviously only a rough estimation. It is however evident that the mass of the SE-12 engine has to be above the RD-180 mass for the following reasons:

- Heavier turbomachinery (higher pump exit pressures and higher power levels)
- Heavier Preburner (higher operating pressure)
- Heavier combustion chamber (longer characteristic chamber length l* [11])

A Single thrust chamber lay-out was selected in contrast to the dual thrust chamber configuration of the RD-180 engine.

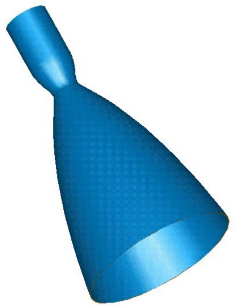

Figure 3 SE-12 Nozzle contour as generated by NCC

Previous DLR studies have shown that truncated ideal nozzles lead to a lower divergence angle while keeping a similar mass as a thrust optimized parabolic nozzle [13]. Furthermore it seems to be the case, that parabolic nozzles generate higher side loads due to different flow separation behavior [14]. Thus an ideal contour is chosen for the 'SE-12' engine. The drawback of this contour is an increased length in comparison to the parabolic nozzle. While this does not increase structural weight, as the nozzle is not encapsulated as required for upper stage engines, it may lead to increased thermal loads on the nozzle when it is subjected to the free air streams during the descent phase of the fly back booster. Therefore the body flap has to be adapted to the nozzle size. Figure 3 shows the engine combustion chamber and nozzle contour, as generated by the tool NCC [15].

Combustion with LOX		RD-180	SE-12
Propellant	-	Kerosene	Methane
Specific Impulse, s/l	s	311.3	322.5
Specific Impulse, vac	s	337.8	348.3
Thrust, s/l	kN	3841	3844
Thrust, vac	KN	4152	4152
TET	K	776	776
Mixture Ratio	-	2.72	3.6
Chamber Temperature	K	3736	3587
Chamber Pressure	MPa	25.63	25.63
Nozzle Area Ratio	-	36.4	36.4
Engine Length	m	3.58	4.64
Engine Diameter	mm	3000	1981 [1)]
Engine Mass [2)]	kg	5 393	6387

[1)] exit diameter
[2)] incl. thrust frame

Table 3 Key Engine Data

The key thermodynamical states of the fluids at various points within the cycle are shown in Figure 6 and Figure 7 for the two rocket motors. Obviously, the methane engine requires increased turbomachine power due to the lower density of the propellant. This can be achieved only through a higher preburner pressure, as the turbine entrance temperature is kept constant in light of reusability considerations.

Table 3 summarizes the key properties of the competing motors that serve as input for the performance calculations.

4 BOOSTER DESIGN

This study employs a basic design of a reusable booster stage as already described in [16]: A cylindrical fuselage contains two separate main tanks (one oxidizer in front and a fuel in the rear), two vertical stabilizers on top of the fuselage in V-configuration, and the rocket engine at the aft protected on its lower side by a body-flap. The investigated configuration is updated according to recent findings within the German ASTRA study to address problems of trimmability for re-entry and return flight [17].

The oxidizer tank is of integral structure, the fuel tank is a separate fabrication, mounted on top of the wing-carry-through. The booster is equipped in the aft section with two military air-breathing engines without afterburner for the return flight. Both boosters use kerosene propellant for the return flight, which is stored in wing tanks and a small trim tank in the nose section. This trim tank is necessary to provide trimming for hypersonic reentry. The required size of the trim tank is about 1 m^3 for the X11 kerosene and 2 m^3 for the X14 methane booster. Both tanks fit well within the available nose volume. The utilization of methane for return flight propulsion of the methane booster was deemed unpromising due to the additional cryogenic tank volume and mass. This assumptions has to be confirmed by a separate trade study however.

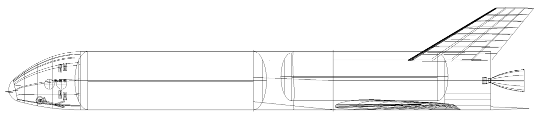

Figure 4 Reusable Kerosene Booster Design X11 Side View

The complete lay-out uses near term technology and avoids comprehensive development programs. The simple geometry intentionally sacrifices some aerodynamic efficiency to structural strength and hence weight reduction.

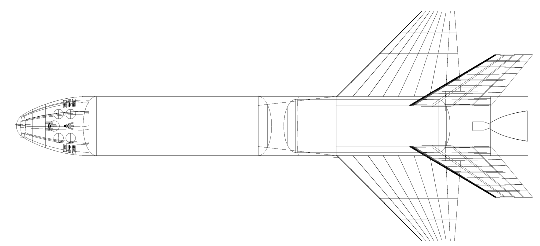

Figure 5 Reusable Methane Booster Design X14 Top View

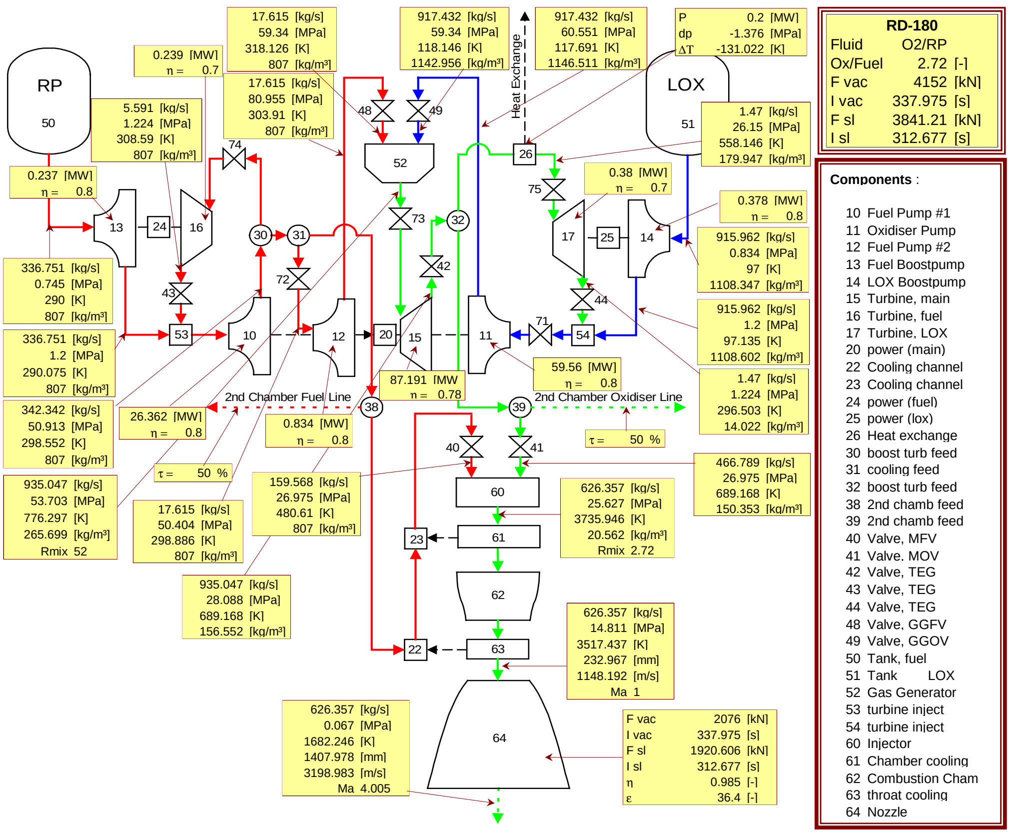

Figure 6 SEQ RD-180 motor cycle simulation (only one thrust chamber is shown)

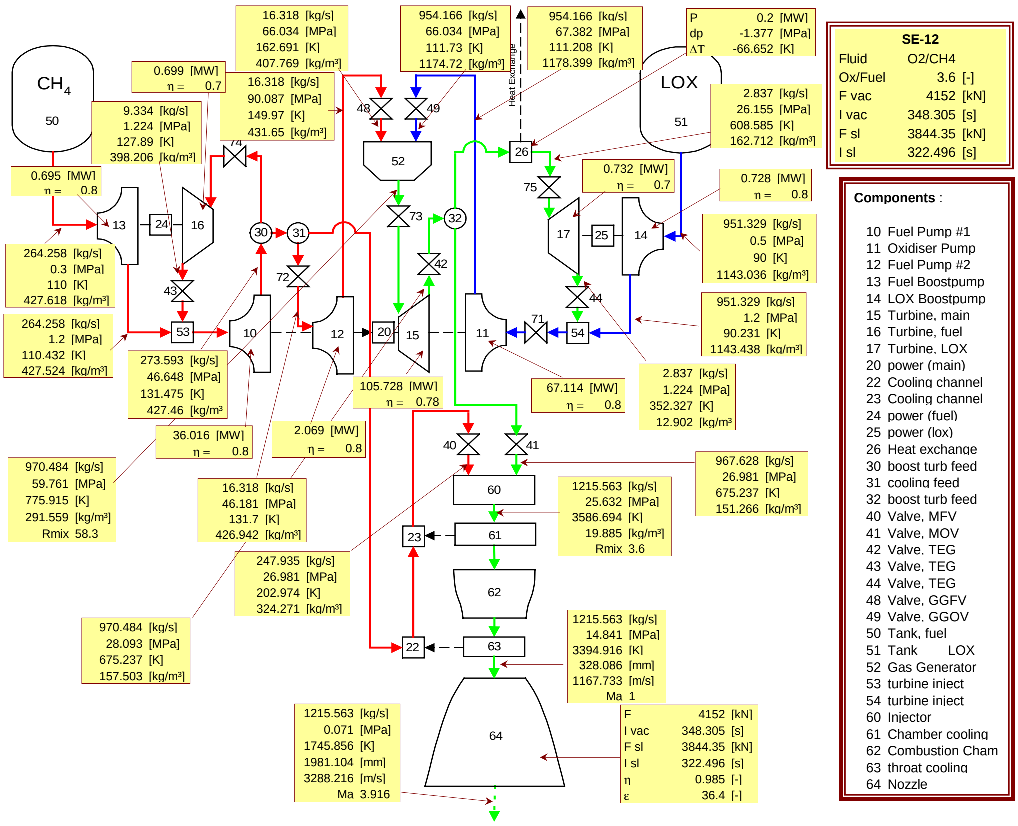

Figure 7 SEQ 'SE-12' motor cycle simulation

5 PERFORMANCE CALCULATIONS

The assessed partially reusable space transportation system consists of two booster stages, which are symmetrically attached to the expendable Ariane 5 core stage (EPC). An upgraded future technology level using a single advanced derivative of the Vulcain engine with increased vacuum thrust, a propellant loading of 185000 kg with slightly 'subcooled' propellants is assumed. A new cryogenic upper stage (ESC-B) is already in the development phase. A closed expander cycle motor of the 180 kN class (VINCI) is foreseen by 2006. This study assumes a total propellant load of the upper stage of about 27000 kg.

The common mission of commercial Ariane 5 flights will continue to be operated from Kourou to a 180 km x 35786 km GTO with an inclination of 7 degrees. This orbit data and a double satellite launch including the satellite support structure SPELTRA is assumed as a basis for this research analysis.

The two boosters were refined through an iterative design process to obtain consistent data sets for ascent and return flight.

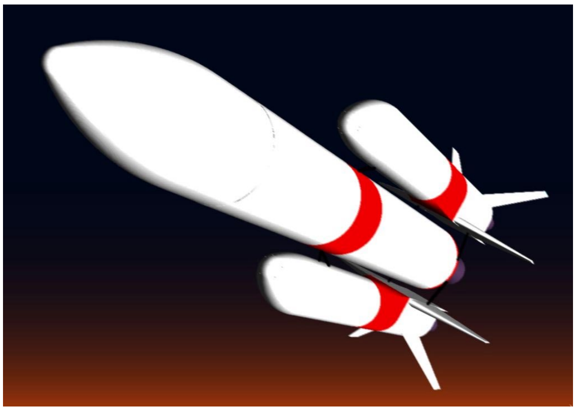

Figure 8 Launch Vehicle Configuration

5.1 Ascent flight optimization

The overall ascent trajectory of Ariane 5 with LFBB is similar to the generic GTO flight path of Ariane 5 with SRM. After vertical lift-off the vehicle turns during a pitch maneuver, and heads eastward to its low inclined transfer orbit. This trajectory has to respect certain constraints, which are close to those of Ariane 5+ ascent.

The separation conditions of the boosters are not explicitly defined. The only criterion is to achieve maximum payload to GTO within the design constraints of the booster and core stage. The total impulse of the booster stages arrive at quite similar values, as the core and upper stages are kept constant. Throttling of the Liquid Fly-back Boosters is not performed since the Ariane 5 acceleration limit is not reached. Flight performance is highly sensitive to the angle of attack history during booster operation. In the performed ascent trajectory optimization it is restricted to remain below 2 degrees, and is further reduced in the region of elevated dynamic pressure to stay below 0.2 deg., to meet structural requirements.

5.2 Descent and Return Flight

The initial re-entry and return flight mass of each LFBB is slightly below the MECO-value, because the aft part of the stage attachment is jettisoned, and the solid propellant of the separation motors is burned out. During the ballistic phase of the trajectory the remaining oxygen in the tank and in the fuel lines will be drained.

Aerodynamic data sets of the booster's return flight configuration have been generated with respect to flap deflection. Calculations showed that both boosters have robust margins for the positioning of the wing. Lift-, drag-, and pitching moment coefficients are used in combination with a calculation of center of gravity movement to perform a flight dynamics and control simulation. The trimmed hypersonic maximum lift-to-drag ratio reaches about 2.0. In the low subsonic and cruise flight regime trimmed L/D is slightly above 5.0. Hypersonic trimming is performed by all aerodynamic flaps including the bodyflap and supported by the RCS. A stable condition is achievable for angles of attack of up to 40 degrees. Therefore 40 deg. is used as the upper limit during return flight for both considered LFBBs.

Due to the flight path angle of about 28 degrees at booster separation, both LFBBs climb in a ballistic trajectory to an altitude of slightly above 100 km. Falling back, the booster reaches a maximum velocity of about 1.87 km/s (about Mach 5.7) at around 50 km altitude, since atmospheric drag is low. Although the angle of attack is held at the 40 deg. limit, a steep trajectory is performed due to the restricted dynamic pressure and lift force, with a path angle γ diving as low as about -33°.

When entering the denser layers of the atmosphere the aerodynamic forces rapidly increase, finally stabilizing the LFBB altitude, and achieving maximum deceleration at an altitude of around 20 km. The simulation is performed under a closed control loop, which keeps the trajectory within normal load boundaries, as far as control surface efficiency is available. An optimal trajectory is found by parametric variation of the initial banking maneuver. The return of the LFBB should start as early as possible, but is obviously not allowed to violate any restrictions. The banking is automatically controlled to a flight direction with minimum distance to the launch site. After turning the vehicle, the gliding flight is continued to an altitude of optimum cruise condition.

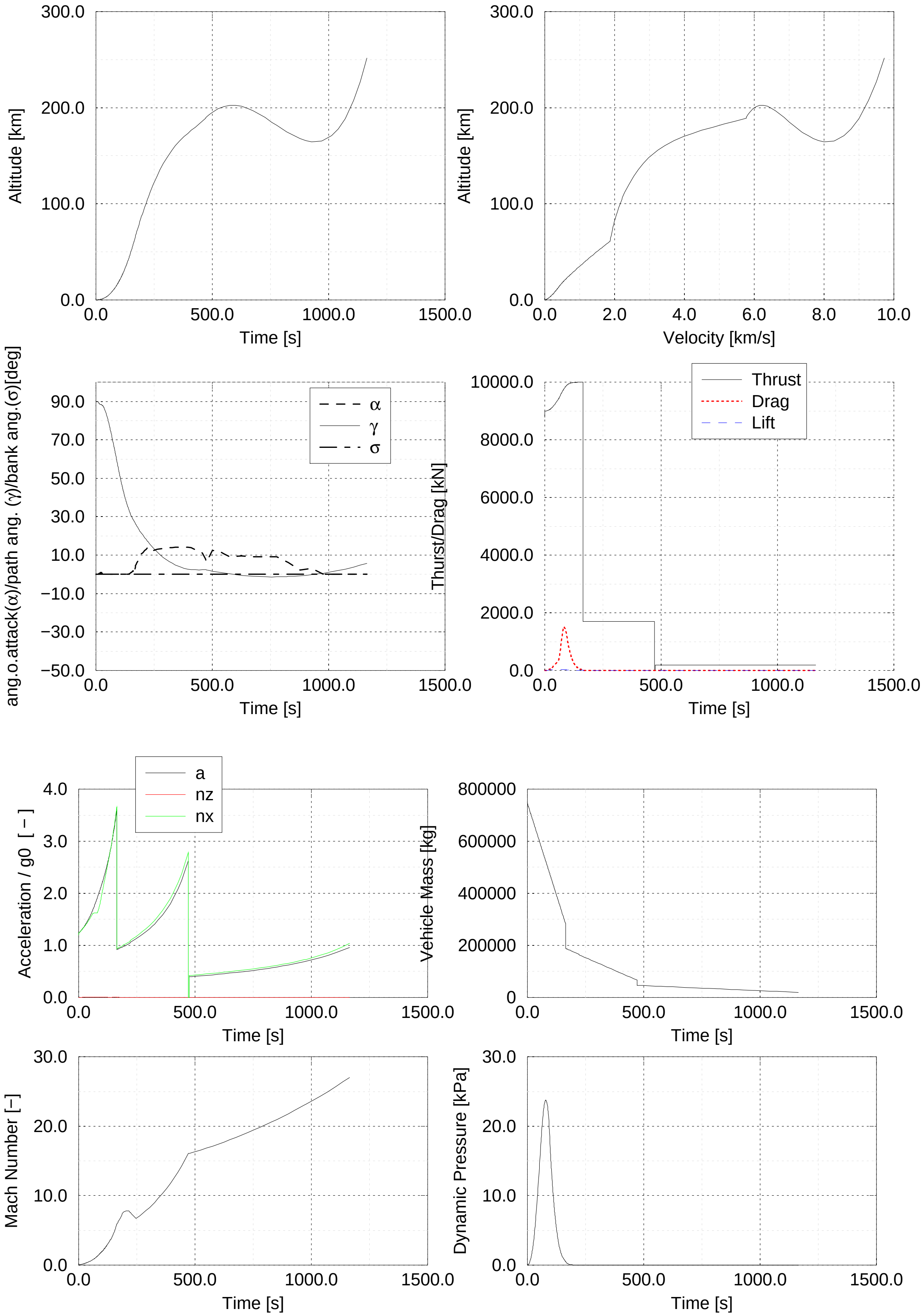

Figure 9 Trajectory Data for ascent flight to GTO for the X14 equipped launch configuration

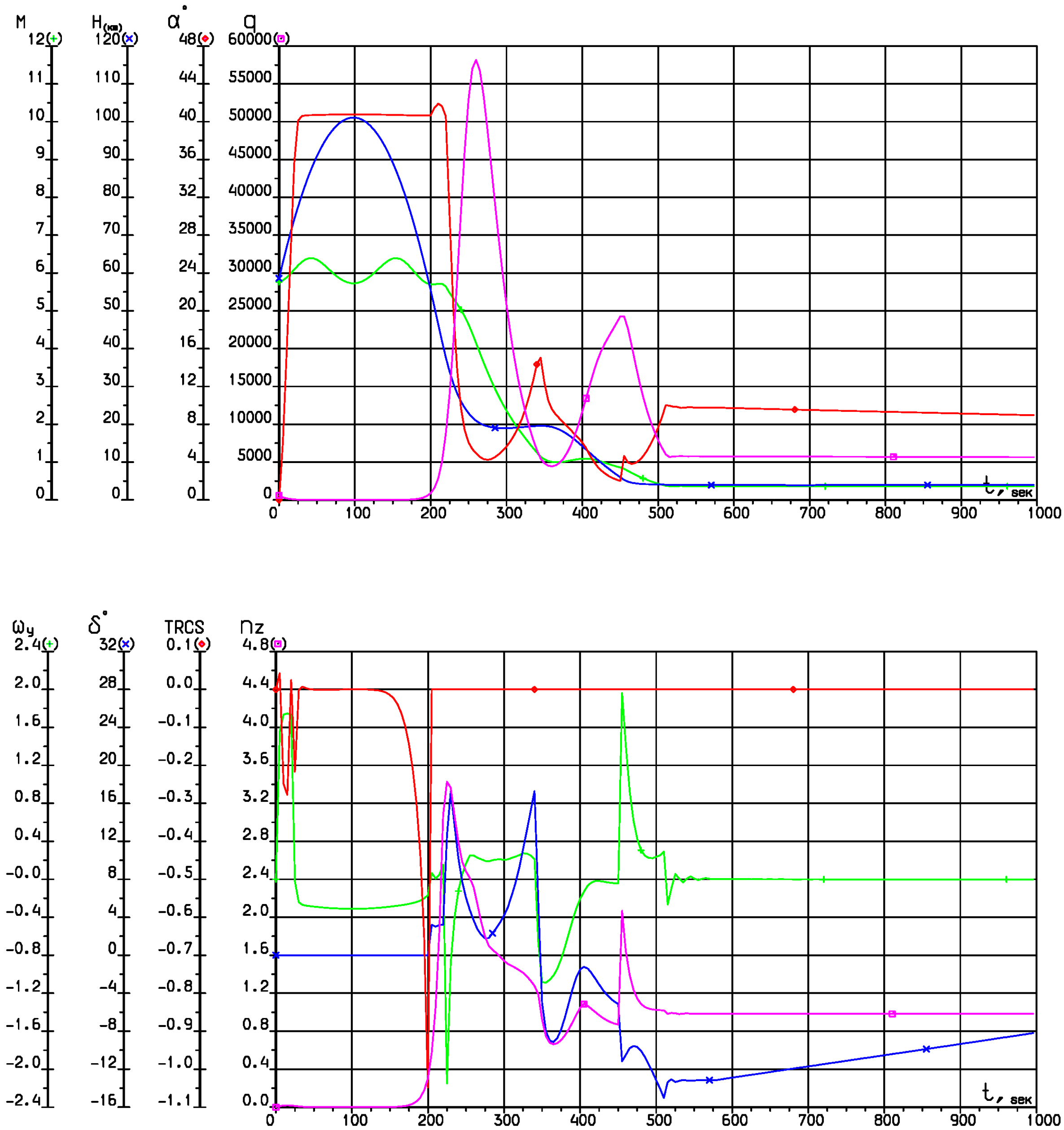

Figure 10 Trajectory Data for initial X11 return flight

An elaborate method is implemented to calculate the required fuel mass of the turbojets for the powered return flight to the launch site. The complete flight is controlled along an optimized flight profile. Aerodynamic data, vehicle mass, and engine performance (available thrust and sfc) are analyzed in such a way, to determine the stable cruise condition with the lowest possible fuel consumption per range (g/km). This is not a trivial task, since engine performance is dependent on altitude and Mach number, and the equivalence of drag-thrust respectively lift-weight is usually not exactly found at maximum L/D. The changing booster mass, due to fuel consumption, and a minimum necessary acceleration performance have also to be taken into account.

The powered return trajectory is automatically controlled to follow the optimum flight condition, always directly heading to the launch site. Fuel flow is integrated to get the exact consumption. In this study, 20% fly-back fuel reserves are included to take into account adverse conditions like head winds.

5.3 Performance Comparison

The principal characteristics of both launcher configurations are listed in Table 4. It is to note that the achieved payload mass is comparable for equal ascent propellant mass.

The trajectory data for a launch with the X14 booster is plotted in Figure 9, key parameters of the return flight of the X11 booster are plotted in Figure 10.

	X11	X14
Booster propellant	Kerosene	Methane
Length [m]	33.96	34.14
Fuselage Diameter [m]	3,80	3.95
Wing Span [m]	15.00	15.36
Booster mass empty* [kg]	31 914	35 209
Ascent Propellant [kg]	200 000	200 000
Residual Propellant [kg]	2 295	2 263
Reserve Propellant [kg]	1 800	1 800
Fly-Back Fuel [kg]	7 800	8 700
Booster Structural Index [-]	0.1506	0.1655
Booster GLOW [kg]	243 809	247 972
LV GLOW [kg]	740 754	749 068
Take-off T/W [N/kg]	12.06	12.01
GTO Payload [kg]	**11 667**	**11 656**
Separation Altitude [km]	58.5	60.2
Separation Mach number [-]	5.74	5.86

* incl. margins

Table 4 Principal Launch Vehicle Characteristics

As a baseline of the study, the length of the boosters had to be kept quasi constant. The methane booster has therefore an increased cross-section as compared to the kerosene booster. This is due to the fact that the high density of LOX in combination with the higher mixture ratio of LOX / methane cannot offset the effects of the low density of methane. The nose section has to be enlarged to fit to the augmented diameter in consequence, which leads to a slightly greater total length of the booster. The attachment points to the core stage are thereby unaffected. The methane booster wings are required to be slightly bigger as the kerosene booster wings to support the higher loads during the atmospheric reentry and the turn maneuver. This is due to the higher separation conditions and the increased booster empty mass. The difference of 3.3 tons for the booster empty mass is mainly due to increased fuselage mass (+ 330 kg), LOX tank mass (+ 285 kg), propellant tank mass (+ 450 kg), propellant supply (+ 200 kg), rocket motor (+ 1000 kg), cryo insulation (+ 350 kg) and in consequence also increased wing mass (+ 190 kg) and landing gear mass (+ 140 kg). The booster GLOW mass of the methane booster is penalized furthermore by 1100 kg of additional fly-back propellant.

5.4 Sensitivities Analysis

Both boosters were designed and evaluated using the same tools. Due to the fact that the principal lay-out of both stages is identical and gross sizes and masses are comparable, it can be expected that any uncertainties, which are inherent to preliminary design tools, will affect both vehicles in a similar manner.

There is however a greater uncertainty concerning the mass estimation for the 'SE-12' methane motor. It was therefore advisable to assess the impacts of booster GLOW variations for the X14 vehicle. The results are shown in Figure 11.

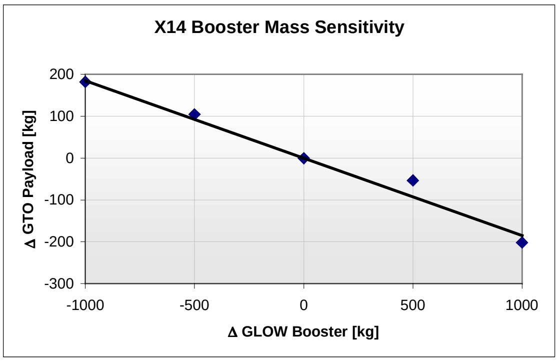

Figure 11 Variation of X14 Booster GLOW

The level of confidence for the calculated specific impulse for both motors is considerably higher. The impact of Isp variations has anyhow been assessed by a parametric study. The results are shown in Figure 12.

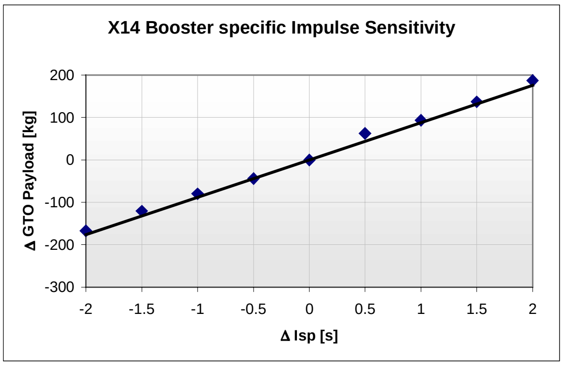

Figure 12 Variation of SE-12 specific Impulse

6 CONCLUSION

The aim of this study was to compare and assess the merits of methane and kerosene as propellants for a reusable booster stage. Some initial findings confirmed already frequently cited statements, e.g. that the specific impulse of a LOX / methane motor is about 10 s higher than for a LOX / kerosene engine with the same cycle. The comparison of the performance of both propellant combinations for a complete vehicle revealed however interesting new results. The study showed that the advantage of the higher energetic content of methane was counterbalanced by an increased motor mass and an increased booster size, hence higher aerodynamic drag and increased mass. The payload performances of the reusable kerosene and methane booster are therefore almost identical with some edge for kerosene. In view of the increased size and dry mass of a reusable methane booster stage, one can expect a cost disadvantage for CH_4 from a launch vehicle system level point of view.

However, if significantly lower operational and maintenance expenses of the reusable methane rocket engine are achievable, a methane powered fly-back booster could get competitive in comparison to a kerosene powered solution.

No studies have been made so far by DLR for other configurations (e.g. expendable liquid booster). We expect however a similar outcome for most cases.

Under these circumstances, it seems reasonable that more research effort is spent to investigate the thermo-chemical behavior of the two propellants. The various points mentioned in paragraph 2 and 3 (coking, sooting, cooling properties, availability, etc.) are by no means an exhaustive list of open questions. Considering that there is only limited practical experience and no design heritage of frequently flown rocket motors using methane or kerosene propellant in Europe, it is essential to study in more detail the numerous advantages and drawbacks of both propellants and their technical implications before a development decision is to be made. Germany investigates in cooperation with Russian companies combustion and thermodynamical properties of methane and kerosene within the ongoing TEHORA 2 program.

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