STUDIES ON THE CHARACTERISTICS OF DE-BOOST MOTORS FOR A SMALL RECOVERABLE ORBITAL PLATFORM

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Abstract

To induce the re-entry of a small recoverable orbital platform from a low earth orbit adjustments have to be made to propulsive and mass characteristics of the de-boost engine. According to necessary velocity decay, re-entry angle and trajectory, and vehicle mass, the de-boost engine should use liquid or solid propellant. While liquid propellant engines are more accurate, solid propellant engines are less expensive. The main parameters that may define the type and size of the de-boost engine are specific impulse, thrust level and burning time. Also uncertainties due to variation of the propulsive parameters should be considered due to recovery area limitations. The present article summarizes the results of a study to define a de-boost engine to be used for the retrieval of a small orbital platform. It describes and compares characteristics of liquid and solid propellant engines, and finally concludes with recommendations for the development of the engine.

Keywords: atmospheric re-entry, de-boost motor, orbital platform, liquid propellant, solid propellant

1. INTRODUCTION

The return of small orbital platforms from low altitude orbit back to the surface of the earth is one of the most complex problems in aerospace engineering The problem becomes more complex if the platform is to be recovered safely and later reused.

Errors in the operation of de-boost engines and disturbances due to atmospheric parameters are the major reasons for high landing point dispersions. Moreover the thrust level and the burning time of the de-boost engine highly influence the re-entry trajectory.

The requirement for a highly accurate de-boost engine leads to the choice of liquid propellant engines. If cost and development aspects are envisaged, solid propellant engines become more adequate.

A recoverable orbital platform, called for short SARA, as shown in figure 1, is under development at IAE-Instituto de Aeronáutica e Espaço (Moraes, 1998). It is a recoverable satellite for utilization as a platform for micro-gravity experiments. The satellite will carry a payload mass of 25 kg and is specified to have a launch mass of about 150 to 200 kg. Its orbit will be circular with an altitude of 300 km and –2 degrees of inclination. According to the life time of on board energy its stay in orbit should not exceed 10 days. After conclusion of the orbital experiments the return procedure will be started, providing firstly the right positioning of the satellite and the de-boost impulse. Following the spacecraft will re-entry into the atmosphere and finally it will land by means of a high performance parachute system (Koldaev & Moraes, 1997).

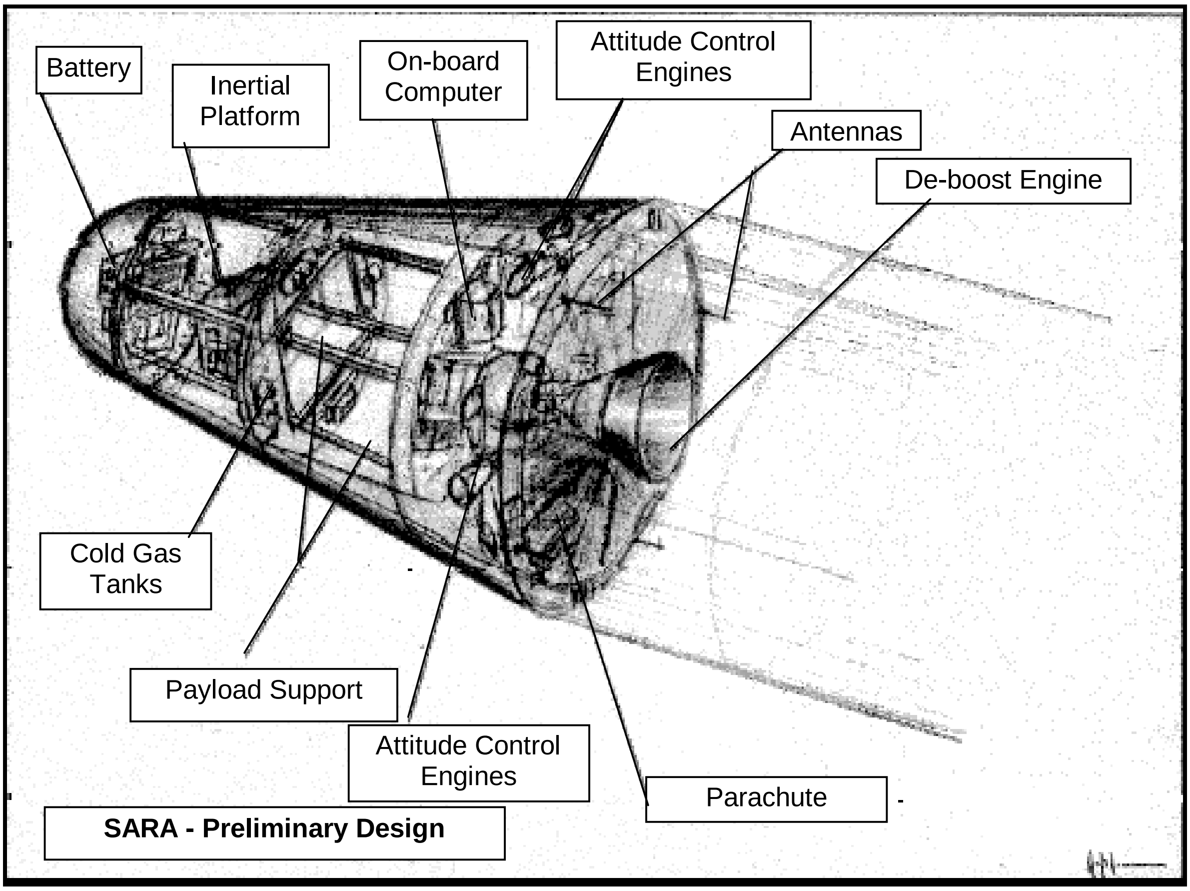

Figure 1. General View of SARA Vehicle

2. ASPECTS OF RE-ENTRY

To provide re-entry of the SARA spacecraft, it is necessary to apply a de-boost impulse which produces a velocity reduction from 235 to 250 m/s, in the direction opposite to orbital motion (Sikharulidze, 1998). The burning time of the de-boost engine should be between 50 to 200 s.

There are different reasons for errors during de-boost impulse action. As a result, the outer-atmospheric trajectory may differ from the nominal one (Sikharulidze, 1999). The most important aspects are:

- The re-entry point into the atmosphere may be displaced with respect to the nominal point. The landing point will have downrange and crossrange displacements accordingly;
- The flight time at the outer-atmospheric trajectory may differ from nominal value. So, the geocentric longitude of the re-entry point will be different from the expected;
- De-boost impulse errors may change re-entry angle. Variation of re-entry angle significantly influences descent trajectory into the atmosphere.

- De-boost impulse errors may change re-entry velocity also changing the descent trajectory into the atmosphere.

Some of the most significant disturbing factors are due to errors of de-boost impulse, as shown below:

- **time of execution**
  May appear as the result of incorrect determination of engine switch on and switch off times.
- **de-boost value**
  May appear as the result of incorrect determination of the required value. Another possibility is linked to the execution errors. The error of de-boost value arises if the control system has no integrator and parameters of engine (thrust, specific impulse) differ from nominal values. Also, errors occur when there is dispersion of the engine impulse in the process of switch off.
- **impulse orientation in the plane of motion**
  Arises due to improper determination of local vertical position (sensor error). For example, as the result of gyro drift in the orbit plane or error of infrared vertical positioning. Another reason is linked with the execution of attitude orientation before the de-boost maneuver.
- **side component of the de-boost impulse**
  Arises as the result of improper determination of orbit plane. For example, due to gyro drift outside of orbit plane.

Other disturbing factors are not directly related to the execution of the de-boost impulse or performance of the engine, but directly related to the engine technology, i.e., type of engine or type of propellant. These errors are the following:

- error due to determination of center of mass (c.m.) position,
- movement of c.m. after expenditure of propellants, gas, etc.

3. CONFIGURATION OF ENGINES

In this chapter possible configurations for the de-boost engine system will be presented and discussed. Conceptual and technological aspects, also results of a preliminary analysis concerning mass of the system, necessity of development tests, and complexity and number of parts will be discussed in detail.

In this work a preliminary pattern is considered that will have to evolve for the final pattern in the development of the propellant. In the present article the propellant will be considered as part of the Propulsive System for Atmospheric Re-entry Induction (SPIRA).

Three alternatives to the SPIRA development will be presented (Villas Bôas, 1999). The first two are based on liquid propellant technology, while the third is based in solid propellant technology.

3.1 Bi-propellant Liquid Rocket Engine

This alternative is composed of a liquid rocket engine system using the propellant pair unsymmetrical dimethylhydrazine (UDMH) and nitrogen tetroxide (NTO). The rocket engine chamber feeding is provided by means of an inert gas (nitrogen) pressurization sub-system. The pressure is applied to positive expulsion tanks containing an internal metallic diaphragm. Other main components are feeding valves, gas and liquid pyrotechnically operated separation valves, gas pressure regulator, piloting solenoid valves. These components are based on the components currently being developed for the roll control system (Yoshino, 1999) of the Brazilian launch vehicle VLS-1 (Boscov, Moraes et all., 1990) . The majority of the components will be the same as those used for VLS and only the development of new propellants tanks is needed, due to the reduced propellant consumption required for SPIRA. In this study a pressure chamber of 0.59 MPa and a mixture ratio (Oxidizer/Fuel) of 1.85, providing a specific impulse of 2471 m/s, are considered. The thrust reaches 390 N, and the propellant mass is 13.6 kg.

The pneumo-hydraulic scheme for this solution is presented in figure 2.

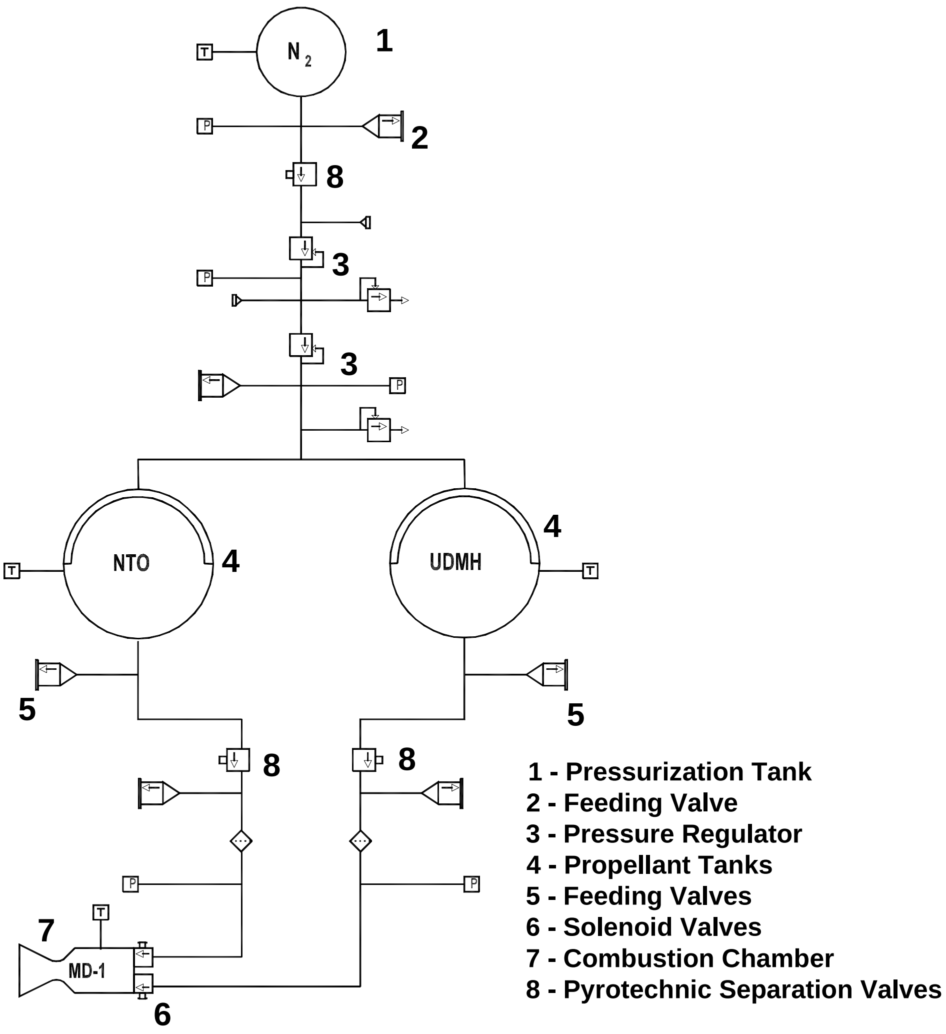

Figure 2. Bi-propellant Liquid Rocket Engine Scheme

3.2 Monopropellant Liquid Rocket Engine

This alternative is composed of a liquid rocket engine system using hydrazine as a mono-propellant. The rocket engine chamber feeding is provided by means of an inert gas (nitrogen) pressurization sub-system, in the same way as the former alternative. The pressure is applied to a positive expulsion tank containing an internal metallic diaphragm. Other main components are feeding valves, gas and liquid pyrotechnically operated separation valves, gas pressure regulator, piloting solenoid valve. A catalytic material produces ignition of the mono-propellant. For this concept a pressure chamber of about 6.9 MPa, 40% dissociation of hydrazine, providing a specific impulse of 2400 m/s (Sutton., 1986) are considered. The thrust reaches 750 N, and the propellant mass is 13.9 kg. The pneumo-hydraulic scheme for this solution is presented in figure 3.

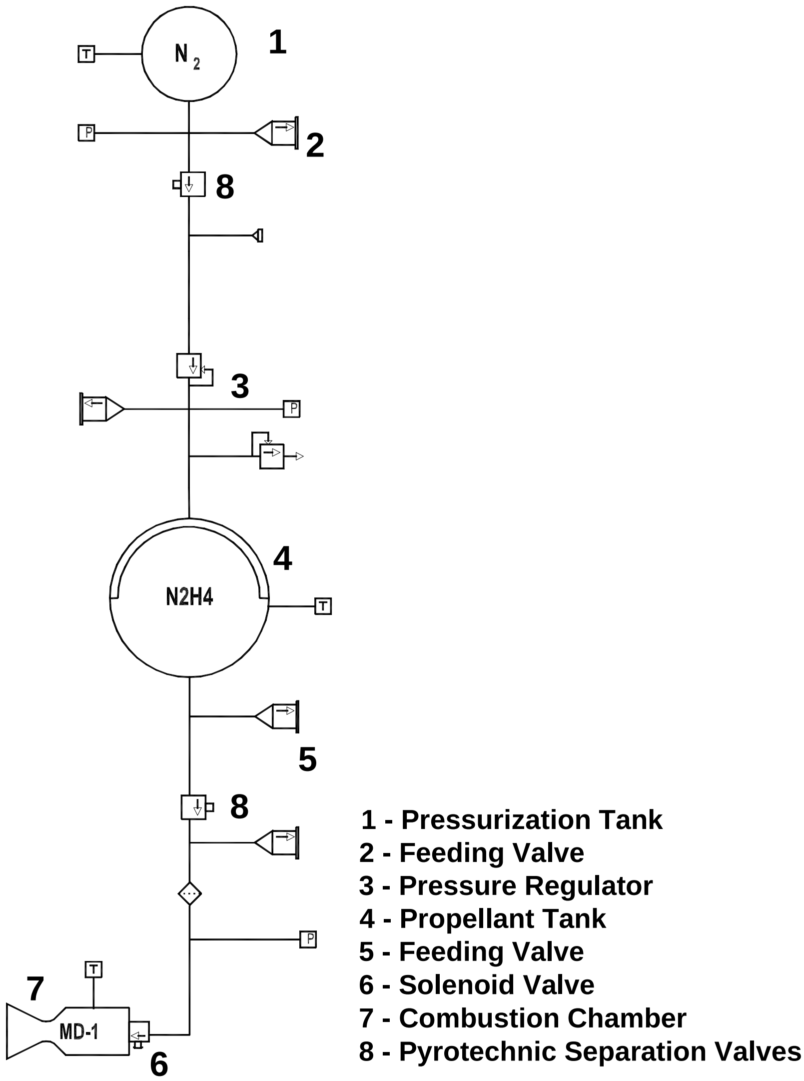

Figure 3. Monopropellant Liquid Rocket Engine Scheme

3.3 Solid Propellant Engine

This configuration is based on the technology developed at IAE for the Roll Control System (PCR/S-IV) of the sounding rocket Sonda-IV (Boscov, 1995). The engine makes use of solid propellant of the end burn grain type. The propellant grain proposed for the SPIRA will have variable burning area, with the final thrust being about 5 to 6 times lower than the initial thrust, as shown in figure 4. This will be provided to reduce the disturbing forces at the end of the burn.

Beyond the reduced thrust in the propellant end of burn, the use of a thrust cutoff system will be considered. This system is composed of a pyrotechnically operated device that ejects a drain plug and opens a hole in the motor envelope. The chamber pressure is instantaneously reduced and consequently also the thrust. This device could be operated only during Phase 2. The main characteristics of propellant are as follows:

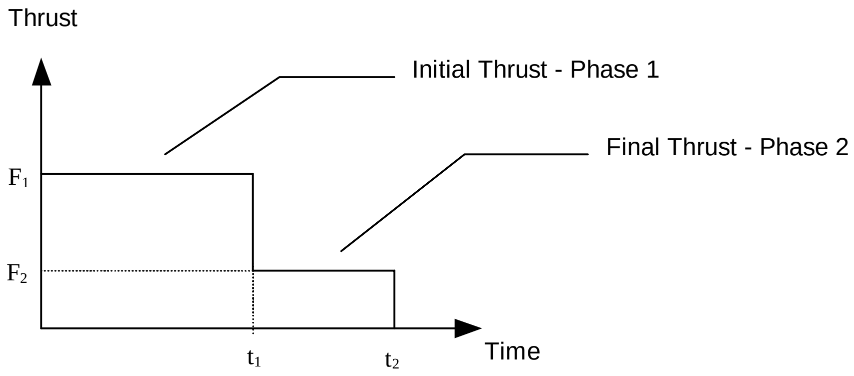

Figure 4. Thrust Curve of Solid Propellant Engine

Propellant Type: Hydroxilated Polybutadiene - HTPB
Propellant Mass: 14.9 kg
Specific Mass: 1670 kg / m3
Specific Impulse: 2250 m / s
Burn Speed: 4.10^{-3} m / s
Chamber Pressure: 3.5 MPa

The main parameters of figure 4 are as follows:

F_1: 600 N t_1: 52.7 s
F_2: 100 N t_2: 72.7 s

The propellant block dimensions are presented in figure 5 and a simplified drawing of the engine is presented in figure 6.

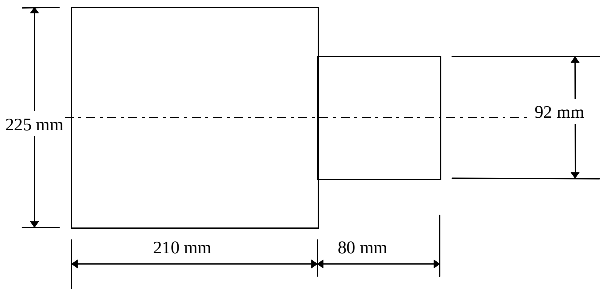

Figure 5. Solid Propellant Block Dimensions

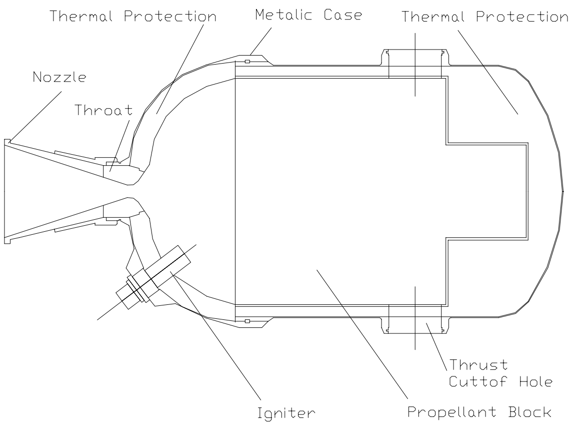

Figure 6. Simplified Drawing of Solid Propellant Engine

4. ANALYSIS

In this chapter the three solutions will be analyzed with respect to mass, development term and cost, production cost, operating precision, handling, safety and toxicity. The results are presented in the table 1.

Table 1. Comparative Analysis of Solutions.

Parameter	Liquid Bipropellant	Liquid Monopropellant	Solid Propellant
Mass	*Higher*: 47.3 kg	*Medium*: 40.3 kg	*Lower*: 35.1 kg
Development Term	*Higher*: high number of parts, more complex, development of new technologies.	*Higher*: high number of parts, more complex, development of new technologies.	*Lower*: low number of parts, less complexity, available technologies.
Development Cost	*Lower*: utilization of parts of SCR/VLS-1, most of the tests will be performed in the SCR/VLS-1	*Medium*: utilization of some parts of SCR/VLS-1, some tests will be performed in the SCR/VLS-1	*High*: new development, higher number of tests.
Production Cost	*Higher*: high number of parts	*Medium*: medium number of parts	*Lower*: low number of parts
Operating Precision	*Higher*	*Higher*	*Lower* (may be improved by use of a thrust cutoff system)
Handling, Safety and Toxicity	Required high level of care, careful operations, high toxic propellants, possibility of spills.	Required high level of care, careful operations, toxic propellant, possibility of spills.	Safe, non-toxic propellant, no possibility of spills.

5. CONCLUSIONS

A study has been made in order to evaluate the most adequate de-boost engine to induce the re-entry of a small recoverable orbital platform from a low Earth orbit. While liquid propellant engines are more accurate, solid propellant engines are less expensive.

The results have shown that concepts using liquid propellant engines have higher precision, but are heavier, the production costs are higher and the development terms are also longer. The technologies are newer, and are still under development at IAE. However, the development will be made indirectly, through the system currently being developed for the SCR/VLS-1. For the bi-propellant solution it will only be necessary to develop new tanks, with identical technology to the ones on the SCR/VLS-1. For the mono-propellant solution it will also be necessary to develop a new catalyst combustion chamber. Manufacturing costs of both liquid propellant solutions will be higher.

A concept using solid propellant apparently has lesser accuracy, which could be improved through the use of the thrust cutoff system, and rigorous quality control during the manufacture of the propellant block. Through the manufacture of twin test motors, it is possible to foresee the characteristics of the flight engine. Manufacturing cost is lower, and the technologies are known, and already used successfully in other previous designs at IAE.

Consideration of the use of the alternative concept with solid propellant is recommended, which has the features of less mass, minor cost, and due to advantages in handling and operation. More detailed studies of the required accuracy should be carried through before a final decision is taken.

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