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SSTO Reusable Launchers: a Critical Comparison of Propulsion Concepts

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Abstract
Single stage reusable launchers could be the best alternative to disposable rockets, but they need a proper propulsion system which can guarantee several flights in its lifespan. An overview of different architectures is proposed, including a critical review, which will select RBCC and SABRE as best candidates. Numerical models are developed for the chosen systems and a rocket-powered baseline to compare their performance for a LEO SSTO flight. Results demonstrate the advantages in terms of mass savings for both RBCC and SABRE, highlighting the properties that allow to select SABRE as the most attractive SSTO engine.

1. Introduction

Space industry has recently experienced an overwhelming growth, pushed both by public and private companies with their necessity of putting satellites in orbit for research and commercial purposes. The increasing demand of space vehicles cannot be further satisfied by expendable rocket launchers because of their unsustainable economical and environmental impact. Recent projects from private companies brought to the development of reusable parts of launchers, mainly lower stages, with complex and heavy fly-back solutions, and still the upper stages are disposable. Furthermore, in order to significantly reduce the economic impact of recovery and refurbishment of lower stages, they have to be employed for several launches, a situation that must be carefully evaluated not to reduce safety. Some projects managed to realise reusable orbiters capable of gliding back and land on the ground, which unfortunately still rely on expendable rocket launchers for their ascent. A valid alternative is needed: the most attractive solution is to develop a fully reusable Single-Stage-To-Orbit vehicle. A purely rocket powered SSTO would require very high propellant mass fractions. It is believed that the best strategy would be to employ propulsion systems capable of increasing the net specific impulse during the trans-atmospheric ascent: air-breathing propulsion technologies can be employed, exploiting atmospheric oxygen for the combustion reactions, reducing the amount of stored oxidizer needed. The main challenge is to develop a propulsive architecture suitable for velocities that range from null up to high Mach numbers in atmosphere, and capable of providing thrust in vacuum conditions for orbit insertion. Aim of this paper is to present different propulsion systems suitable for a Single Stage vehicle for LEO missions, select the architectures that seem to better satisfy all the requirements and carry out a preliminary performance analysis to identify the most promising technology.

2. SSTO Launchers - A Comparison of Different Architectures

Different propulsion strategies have been proposed through years to provide an efficient solution to all issues concerning orbital insertion adopting a SSTO strategy [1].

2.1 Thermochemical Rockets

Advantages of these architectures are the high thrust-to-weight ratio, the ability of working in vacuum conditions and the high experience gathered in several decades of employment for space missions. Solid Rocket Motors are widely used for sounding rockets, military missiles and boosters. They provide very high thrust and structural simplicity, but restarts are not available, thrust profile is established during the design and combustion products of conventional compositions are armful to the environment. Specific impulses reached are low, approximately between 300s and 350s.

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Hybrid Rocket Engines' main characteristic is the different state in which fuel and oxidizer are stored. They provide a good level of specific impulse, between SRMs and LREs, and high flexibility in operation. Nevertheless, low regression rates limit the thrust produced by these systems. Liquid Rocket Engines provide high specific impulses, allow for thrust control and restart during the mission, and some propellant mixtures are environment respectful, but they usually rely on turbomachinery, a negative contribution in terms of structural complexity and weight. The main drawback of rocket systems is the necessity to store both fuel and oxidizer needed for the mission.

2.2 Ramjets and Scramjets

Ramjet engines exploit ram compression to slow down the incoming airflow and increase its static pressure: incoming supersonic airflow is compressed and decelerated to subsonic conditions, usually through a series of oblique shockwaves generated inside the inlet. Air is then mixed with fuel and burnt in the combustion chamber. High temperature products are eventually expanded into a nozzle. Conventional hydrogen fuelled ramjets reach up to Mach 5 or 6, higher flight Mach numbers require different propulsion systems. Scramjets allow to accelerate a vehicle up to higher velocities. These devices have similar configurations with respect to ramjets: they consist of ducts without any moving part embedded. The main difference is that in scramjets the airflow never reaches subsonic conditions: incoming air is decelerated and compressed through a series of oblique shockwaves, but its velocity still remains supersonic. Scramjets theoretically allow to provide thrust in hypersonic flight regimes, ideally from Mach 5 and even above Mach 10. In this sense, a dual mode ramjet-scramjet engine is able to cover a wide range of flight Mach numbers, but it would be able to provide thrust only if the vehicle is already moving, giving reasonable performances above flight Mach numbers between 2 and 3. Different solutions must be employed to accelerate the vehicle from steady conditions on ground and in vacuum for the in-space phase of the ascent.

2.3 Liquid Air Cycles

Liquid Air Cycle Engines (LACE) combine air-breathing solutions with conventional rockets. The fuel employed in this engine is liquid hydrogen, burnt inside a combustion chamber with oxidizer taken from atmosphere. The working process requires to pump the liquid fuel into a precooler and a condenser, exchanging heat with the incoming air flow. This procedure allows to liquefy air, which is lately pumped inside the combustion chamber and burnt with hydrogen coming from the cooling jacket, like in conventional LREs. Combustion takes place at high pressure, and the gaseous products are then expanded into a supersonic nozzle. The basic LACE engine ideally works with an equivalence ratio of 7 to 8, resulting in a specific impulse around 1000 s. The main mechanism might be improved in different manners, for example by employing a small turbine to expand the hydrogen and further reduce its temperature, like in cryo-jets, providing a high theoretical specific impulse of 3000-4000s. On the other hand, a better air cooling ability collides with the intensification of risk of solidification of moisture inside the heat exchanger. Ice accumulation reduces heat transfer capability and eventually clogs the airflow path. A more complex variant of the LACE, the Air Collection Enrichment (ACE) system, incorporates a liquid oxygen separator after the liquefier, which allows to simultaneously feed the rocket combustion chamber and fill the oxidizer tanks during the air-breathing ascent. Both conventional LACE and ACE technologies are not ideal for a SSTO launcher because of their high fuel consumption: basic LACE at sea level conditions require fuel air ratios even 8 times greater than the stoichiometric one to efficiently cool down the air. Furthermore, their technological complexity does not compensate the small gain in terms of specific impulse with respect to conventional thermochemical rockets.

2.4 Rocket Based Combined Cycles

Rocket-Based Combined Cycles (RBCC) are composed of different subsystems, particularly a dual mode ramjet-scramjet and a rocket, that work collaboratively in different combination in response to changing flight environments. A RBCC engine consists of an inlet, an isolator, an ejector primary rocket, a combustion chamber and a nozzle. In the first phase of the mission, the rocket is activated to generate thrust in static condition, working from null velocities up to supersonic flight as an ejector. This allows to bring the vehicle in the correct configuration for activating the ramjet mode, which is expected to take place at a flight Mach number between 2 and 3. Rocket is shut down, and the ramjet engine accelerates the vehicle in an efficient way up to Mach 5 or 6, where the transition to scramjet mode takes place. Inlet and isolator slow down and compress the airflow through a series of oblique shocks. The flow enters the combustion chamber in supersonic conditions, where it is mixed with fuel and burnt. Combustion products are finally expanded in the supersonic nozzle, providing thrust. The scramjet accelerates the vehicle up to about Mach 10, when it reaches an altitude where air is too thin to allow an air-breathing device to work properly. The final working mode transition of the RBCC is performed: the inlet is closed to avoid air entering the engine, while the primary rocket is again operating, providing all the thrust necessary to reach orbit. RBCC engines may be divided into two main families according to their conceptual layout. The first family is represented by axisymmetric configuration engines, in which

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the rocket is installed in the central axis as a conical rocket. The second layout is more similar to typical scramjets, as it is based on a rectangular section design. This configuration allows to integrate the inlet with the vehicle body, as well as the nozzle, obtaining a beneficial effect form the forebody compression and afterbody expansion of the flow entering and exiting the engine, respectively, like in strut-jets. RBCC engines, particularly strut-jets, are suitable for both horizontal and vertical take off, thanks to the high thrust provided by the rocket in the first acceleration, and constitute perfect candidates for SSTO LEO missions or TSTO launches.

2.5 Turbine Based Combined Cycles

The idea behind these devices is to embed a turbojet engine along with a dual mode ramjet-scramjet system. The turbo-compressor allows to feed the combustion chamber with high pressure air at subsonic and low supersonic velocities. At higher Mach numbers, the incoming airflow is deviated in order to bypass the turbojet section, exploiting the compression effect of the geometry of the duct like a conventional ramjet system. Integration of both systems is fundamental to reach high flight velocities. Different architectures have been proposed for TBCC engines. The simplest one considers two different flow paths for the different working modes, significantly contributing to the increase of the total weight of the engine. Other configurations propose to integrate the turbojet and the dual mode ramjet-scramjet subsystem in the same air flow path, accounting for specific devices capable of deviating the flow towards the correct subsystem according to flight conditions.

2.6 Precooled Hybrid Air-Breathing-Rocket Engines

These kind of engines are still matter of research. They come from the basic idea of LACE technology, adding some improvements to grant a high specific impulse, and combining air-breathing and rocket propulsion in two different phases of the missions employing the same combustion chamber and nozzle. The first attempt to realize such system was Rolls Royce air augmented rocket engine RB545, developed as propulsion system of the abandoned HOTOL space launcher. The engine employed high pressure hydrogen to cool down the incoming airflow directly. The fuel flow was then split in two different streams: 2/3 of the available flow were forced to expand in a turbine to provide power to the compressor. The remainder hydrogen was injected in the combustion chamber.

Synergetic Air-Breathing Rocket Engine, also known as SABRE, is a hybrid air-breathing rocket engine developed by Reaction Engines Ltd., originally designed as propulsion unit of the concept SSTO vehicle SKYLON. It works in both air-breathing and rocket mode employing the same combustion chamber, associated pumps, pre-burner and nozzle. After being slowed down and compressed by an axisymmetric shock cone inlet, incoming airflow is split into two different streams. The first flow enters the central path, passing through a precooler and a heat exchanger to reduce its temperature to the vapour contour, without liquefying and avoiding clogging issues. The flow encounters a turbo-compressor before entering the combustion chamber. Both the heat exchanger and the turbo-compressor are powered by a separate helium cycle, kept at low temperature employing LH2 fuel. A pre-combustor is located immediately after the compressor, where a small amount of air is burnt in excess of hydrogen and then forced into a heat exchanger to further heat up the helium before its turbine expansion. The fuel rich combustion products are then injected into the main combustion chamber to complete their reaction. After combustion is completed, hot high-pressure gases are finally expanded in the E-D nozzle, in which a central spike pushes the flow against the walls of the bell creating a separate flux region inside the divergent part of the nozzle. The secondary airflow enters a bypass ring region that feeds ramjet burners located all around the main nozzle exit. This process allows to consume the excess hydrogen used in the heat exchanger to provide a sufficient cold helium mass flow rate to reduce air temperature in the precooler. The complex air-breathing phase of the SABRE is able to provide thrust from steady conditions on ground up to a flight altitude of 28.5 km and a Mach number around 5.5. At this point the engine switches into pure rocked mode, feeding the combustion chamber with stored LOX.

3. Advantages and Performance Analysis

3.1 Selection of best Candidates

Among all candidates presented, only a reduced number of technologies can be considered suitable for a SSTO mission. Criteria of selection are based on the necessity to integrate in a unique propulsion unit all the technologies required to bring the launcher from steady conditions on ground up to the target orbit. Ramjets, scramjets and dual mode engines are not an appropriate choice to carry out the entire mission, since they would require two different subsystems to provide thrust during the initial and final phases of the ascent. Liquid Air Cycle Engines could work even in vacuum conditions if a secondary plant is included to feed the combustion chamber with stored oxidizer, but the necessity to bring air in liquid state requires extremely low temperatures, and the amount of fuel needed for the liquefying process

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is very high: fuel/air mixture of 0.2 are required. Turbine-Based Combined Cycles allow to work from steady conditions up to supersonic Mach numbers. On the other hand, they are not able to provide thrust in space applications, hence they would require a second propulsion unit for acceleration in vacuum conditions.
The unique propulsion architectures capable of generating thrust efficiently both in trans-atmospheric and in-space phases of the launch are Rocket-Based Combined-Cycle engines and the innovative Reaction Engines Limited SABRE. These technologies make use of hydrogen as fuel, both in air-breathing and rocket working modes: this fact represents an advantage for future, sustainable flights, since combustion products are not toxic. To make a significant analysis of the advantages brought by these technologies, a preliminary mission design will be carried out in order to compute the amount of propellant and structures needed to complete a Low Earth Orbit flight. A rocked-propelled, single stage baseline will participate to the comparison, providing a reference point to quantify the differences between the currently employed technologies and the different innovative solutions proposed.

3.2 Mission Definition

A target orbit of 300 km is selected, corresponding approximately to the lower limit of exploitable Low Earth Orbits interval. A hypothetical payload of 5000 kg is considered for the analysis. The ΔV has been computed through the following formula

$$\Delta \mathrm{V} = \sqrt{\frac{\mu}{R+h}} \tag{1}$$

where μ is the standard gravitational parameter of Earth, R is Earth radius and h is the distance of the selected orbit from the surface of Earth, both in meters. For the preliminary design of the mission, average contributions to the velocity variation are considered.

4. Models Development

4.1 Atmosphere Model

The model is developed basing on data from the 1976 U.S. Standard Atmosphere Model, and it provides temperature, pressure, density, molar mass and sound velocity in International System units according to the altitude considered. The target orbit selected for the comparison is 300 km high, atmospheric data range from ground level up to the altitude of the orbit selected, including the troposphere (0 – 14.5km), the stratosphere (from tropopause up to 50km), the mesosphere (from stratopause up to 85 km), part of the thermosphere (above 110 km) and the relative pause layers. The model is built to solve for air properties at such high altitudes only for completeness: as it will be clear in following sections, air-breathing working modes of the selected engines shut down at lower altitudes in atmosphere.
Atmospheric properties allow for calculation of several effects, including dynamic pressure, produced by the resistance imparted from the air to the vehicle flying across the atmosphere. It is the main driving parameter of aerodynamic forces once the geometry and flight configuration of the vehicle are defined. To avoid excessive loads on the structure of the spacecraft, dynamic pressure corridors have been defined, dictating the flight Mach number not to be exceeded in order to grant a suitable value of dynamic pressure according to operating altitude. Typical limit values go from 50 kPa for rockets and manned vehicles up to 95 kPa for military missiles. Low dynamic pressure trajectories allow to reach high velocities at elevated altitudes, keeping considerably small loads on the structures. Nevertheless, flying at high altitudes means crossing reduced air density regions, which results into insufficient mass flow rate for the engine in air-breathing mode. For this reason, a trade-off has to be accomplished in order to keep reasonable values of aerodynamic drag and the proper mass flow rate necessary for the correct operation of the engine.

4.2 Rocket-Based Combined Cycle Numerical Model

Several codes have been developed to compute performance parameters for Rocket Based Combined Cycle engines: the first numerical tool, SCRREAM, was developed and presented by J. Olds and J. Bradford [2]. Hybrid Propulsion Optimizer (HyPro) developed by Mogavero et al. include the ability to analyse the full working process of a RBCC. In 2010, Williams developed ERIDANUS [3], a MATLAB code written to predict performance parameters for all the working modes of a RBCC engine. Eventually, in 2020 Zhang et al. developed the RBCC analysis tool Skye [4], to carry out the preliminary design of a concept TSTO launch system and its optimization.
None of this computational tools has been employed in this work: an ad-hoc code is written to compute performance parameters for the four different working modes of the considered engine. The RBCC is modelled on the basis of the works of T. Zhang et al., F.S. Billig and J. R. Olds. The working cycle of the engine is split into four different modes,

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basing on the flight regime. Between Mach 0 and 2, the engine works in ejector mode; between Mach 2 and 5, the engine is functioning in ramjet mode; from Mach 5 up to Mach 10, the RBCC works in scramjet configuration; from Mach 10 up to orbital velocities, the pure rocket working mode is activated.

Key Assumptions. In order to develop a preliminary and simplified analysis tool, some assumptions have to be made: the code computes flow parameters solving one-dimensional compressible equations for conservation of mass, momentum and energy for a hydrogen fuelled RBCC, considering the embedded primary rocket fed by cryogenic LOX/LH2 propellant mixture. Each component of the engine is treated as a control volume, in which air, combustion products and their mixture behave as calorically and thermally perfect gases, neglecting friction and boundary layer effects. No transients are considered, transitions from a working mode to another take place instantly. In ejector mode, flow mixing is considered ideally completed. In air-breathing phases, combustion is modelled as heat addition, providing an efficiency of 90% to account for performance losses. In subsonic combustion working modes, namely ejector and ramjet, the flow is thermally choked. In scramjet mode, the compression process inside the inlet is characterized by complex oblique shockwaves trains, which are modelled considering a unique strong oblique shockwave, relying on an empirical relation to compute total pressure loss. The expansion process is modelled accounting for a variable geometry ramp nozzle. These assumptions, which are mainly the same adopted by Zhang, Billig and Williams, make the solution of the simulations different from real behaviour of the gas flows inside the engine. Nevertheless, they allow to build a simplified yet enough accurate model suitable for a preliminary study.

Stations Identification. Figure 1 shows all the stations identified in the engine flow path, corresponding to critical positions where flow properties are calculated.

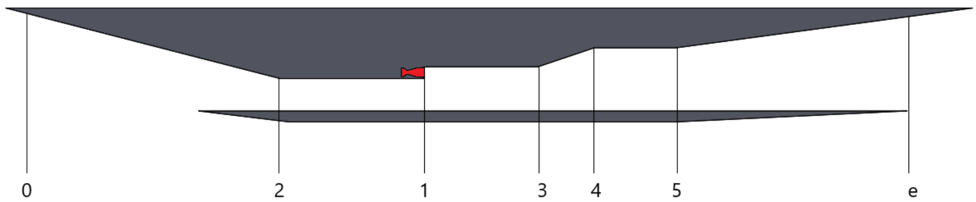

Figure 1: RBCC stations identification

Station 1 denotes the exit area of the primary rocket nozzle and the beginning of the mixing region; station 2 corresponds to the exit section of the inlet; station 3 marks the end of the mixer; station 4 identifies the entry section of the combustion chamber; station 5 denotes the exit of the combustion chamber and the beginning of the nozzle: it marks the choking position of the gas flow for ejector and ramjet modes; station *e* corresponds to the exit section of the main nozzle. Inlet entry area and main nozzle exit area are considered variable: these devices are modelled as ramps capable of changing their position in order to maintain the correct velocity for air at the entry of the mixer and provide optimal expansion according to atmospheric pressure, respectively. Areas of section 2 and 3, maximum areas of inlet and nozzle and primary mass flow rate are assumed to be the same as the ones of Hyperion propulsion systems described by J. Olds. Station 4 has a different area with respect to the one proposed by Olds because of a different assumption: the engine designed for Hyperion employed a physical throat for the expansion process in the main nozzle for subsonic combustion, while, on the other hand, the RBCC system considered in this work employs a thermal throat. Because of the adoption of a thermal chocking, the section of the combustion chamber had to be reduced.

Table 1: Engine dimensions

Maximum Inlet	A_2	A_3	A_4	**Maximum** A_e
2.51 m^2	0.77 m^2	1.05 m^2	1.5 m^2	8.83 m^2

Ejector Mode. The ejector mode is modelled starting from the LOX/LH2 liquid primary rocket of Hyperion spaceplane, which provides a total mass flow rate of 97.975 kg/s. The combustion chamber pressure is set to 137 bar, with a mixture ratio of 8, corresponding to the stoichiometric value to avoid spontaneous combustion between air and unburnt hydrogen in the plume of the rocket during mixing process. Primary rocket flow properties are computed with the NASA software Chemical Equilibrium and Applications, solving for combustion temperature, density of gases in combustion chamber and molar mass of combustion products, whereas specific heats and their ratio are computed according to the perfect gas theory. Flow composition is assumed to be frozen in the primary nozzle. Knowing mass flow rate and combustion chamber mixture properties, primary throat area can be computed. Secondary flow properties are computed setting a Mach number of 0.8 at the exit of the inlet, consistent with Billig's results [5]. In order to solve

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for other flow properties at entry station of the mixer, total pressure recovery of the air intake is modelled applying the empirical relation (2), in which P_{t2} and P_{t0} represent the total pressures in station 2 and 0, and M_0 is the Mach number of incoming air in front of the intake.

$$\frac{P_{t2}}{P_{t0}} = 1 - 0.0291\,M_0 - 0.0206\,M_0 \tag{2}$$

Mixing process computation is carried out with the hypothesis of constant pressure: the rocket plume enters the mixer and continues its expansion process, while the secondary air-flow is accelerated and expanded by the primary flow, until the two flows match the same pressure value, computed depending on the conditions of the two flows. Define P_2^* and P_1^* the pressure reached respectively by the secondary and primary flows when one of them completely occupies the whole mixing region. If $P_1^* < P_2^* < P_1$, the matching pressure of the mixed flow is P_2^*, otherwise if $P_1 < P_2^* < P_{t1}$, the mixed flow settles at pressure value P_1 [4]. If the flow velocity after the mixing process is supersonic, a normal shockwave reduces its speed to subsonic values before entering the expansion section upstream of the combustor [9]. Properties of the flow are computed according to the shockwave relations. The following compression process that the flow experiences is assumed as isentropic: isentropic relations are used to compute properties of the flow entering the combustion chamber given the area ratio between station 3 and station 4.
Combustion process in section 5 is treated as heat addition, and the flow is brought to choking conditions. Properties of the gases at the exit of the combustion chamber are computed according to Rayleigh choked flow equations: total enthalpy balance allows to compute the mass flow rate of the fuel necessary for the combustion reactions, providing an efficiency of 90% to account for losses that might affect the process. The final expansion phase is assumed isentropic, and properties at the exit of the nozzle are computed solving isentropic one-dimensional compressible relations. The expansion is assumed optimal at each altitude, accounting for a variable, ramp-like nozzle. If this process results in a exit area larger than the maximum allowable one, the code fixes A_e at its maximum value, and computes flow conditions with the new area ratio. Eventually, once the flow properties at the exit of the nozzle are known, thrust, specific impulse and specific fuel consumption can be computed.

Ramjet Mode. Ramjet mode is modelled similarly to the ejector part, but, in this case, the primary rocket is not working. Incoming air is forced through a normal shockwave on the cowl lip of the inlet, that brings its velocity in subsonic conditions. Airflow Mach number is set to 0.8 at the exit of the inlet, consistent with the previous working mode. The flow experiences a series of isentropic compression and expansion processes passing through the isolator and reaching the combustion chamber. Here, the flow is thermally choked following the same heat addition modelled in the ejector section. Eventually, isentropic expansion process is simulated with the same procedure applied in the first propulsion mode, computing molar mass of combustion products exploiting the "combustion" problem set up provided by CEA code. Thrust specific fuel consumption and Is are obtained similarly to the previous case, but the primary mass flow rate is null since the rocket is shut down.

Scramjet Mode. The scramjet mode is more complex to be efficiently modelled: assumptions and simplifications allow to provide an approximated yet valid model. Total pressure recovery of air in the inlet is computed according to the empirical formula (3), in which the properties considered are the same of (2):

$$\frac{P_{t2}}{P_{t0}} = \frac{800}{M_0^4 + 935} \tag{3}$$

Given this relation to compute total pressure in station 2, the inlet compression effect is simulated as a unique strong oblique shockwave: starting from the total pressure recovery relation, the equivalent angle of deflection of the flow has been computed through oblique shockwave relations. This procedure allows to obtain an equivalent normal component of the Mach number of the free stream, model the oblique wave and compute flow properties across the shock. After the compression process is carried out by the inlet, the flow is assumed to cross in an isentropic way the isolator. Combustion is assumed to take place in a stoichiometric mixture composition. There is a limit to the heat that can be added in the scramjet combustion process: according to Rayleigh equations, heat addition in a supersonic flow slows it down, eventually making it reach choking conditions. The code developed for the scramjet part automatically reduces the equivalence ratio if heat addition reaches unacceptable results of subsonic flow after combustion, allowing to respect the physical limit stated by the choking condition. Nozzle expansion is eventually modelled solving similar equations with respect to the ones employed in the previous working modes, accounting for total pressure losses connected with high velocities by introducing an expansion efficiency set to 90%.

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Rocket Mode. For the final propulsive phase, the air inlet is closed and the primary rocket embedded in the engine is re-activated. Rocket mode is modelled according to conventional rocket theory. Throat dimensions, propellant mass flow rate and combustion chamber pressure are the same as the ejector primary working mode. CEA software allows to compute combustion temperature, density of reactants and molar mass of combustion products, considering stoichiometric mixture ratio. Shifting equilibrium is chosen between the combustion chamber and the throat; in the divergent part of the nozzle, frozen composition assumption is adopted. Also in this case, the nozzle can adapt its shape in order to optimize the expansion process according to changing working environment, up to the altitude at which limit exit area allowed by engine geometry is reached. Above this altitude, the nozzle efflux area is fixed, and the rocket works in under-expanded conditions.

4.3 SABRE Model

There is scarce practical knowledge about the engine. The simplified model of the engine is constructed starting from J. Zhang et al. work [6]: a detailed study of each component of SABRE (Synergetic Air-Breathing Rocket Engine) would require in-depth analyses that fall outside of the purposes of this work. Zhang at al. modelled the air intake by determining a total pressure recovery coefficient for the adiabatic compression process. In this way, temperature and pressure values at the exit of the inlet are computed according to flight altitude and Mach number. Exiting from the inlet, air is forced inside the shell of the heat exchanger, where it is cooled down by the helium passing through the tube bundle. Air temperature at exit section of the precooler is 133 K. After the cooling process, air enters in the compressor. There are no available parameters to determine the performance of this device, hence Zhang et al. modelled it according to reference turbomachinery, modified with defined coupling factors. The result in design point showed that air exiting from the compressor has a pressure of 145 bar and a temperature around 650 K, corresponding to a compression ratio of 111. Eventually, part of air is injected in a secondary combustion chamber, where it is burnt in excess of fuel to heat up the helium coming from the precooler, before its expansion in the turbine that drives the compressor. Fuel rich combustion products are then injected into the main combustion chamber, along with the remaining air, which experiences a 40% pressure drop during the injection process. Solution of the model at different altitudes allowed the authors to compute air mass flow rates in the main and secondary combustion chamber and their pressure and temperatures. Fuel mass flow rate is kept constant at 34.6 kg/s. With all the properties resulting from the model presented by Zhang et al., CEA software is employed to compute density of reactants in the combustion chamber and combustion products properties, in order to model the expansion process for each point. A linear interpolation of the discrete results allows to solve for flow properties at the exit section of the nozzle as function of altitude. An expansion-deflection nozzle is under consideration for SABRE, which makes use of a centre body in the throat, that changes its position according to the flight altitude. In this work, a conventional nozzle with an expansion area ratio of 46 is assumed, providing an over-expanded thrust up to 15 km, where the exit pressure matches the atmospheric one. Above this altitude, the engine works in under-expanded condition. Given properties of the mixture in the combustion chamber, expansion process is modelled using isentropic relations for velocity, temperature and pressure of exhaust gases given the expansion ratio of the nozzle.
For the rocket working mode, combustion chamber pressure is fixed to the value it reaches at the end of the air-breathing phase: 104 bar. The engine works with cryogenic mixture LOX/LH2, under the assumption of stoichiometric composition. As previous cases, CEA software allows to compute combustion chamber temperature, density and all properties of gases in expansion process, considering mixture chemistry as frozen. Throat area is fixed by chosen nozzle design, while exit area is changed to provide better performance in vacuum operations. This hypothesis can be translated into a conventional double-geometry nozzle, where a secondary section is dropped to enlarge the divergent part.

4.4 Thermochemical Rocket Baseline

For the pure-rocket SSTO preliminary design, an existing rocket engine is selected to start the computation process. Particularly, Aerojet Rocketdyne AR-25, also known as Space Shuttle Main Engine SSME, is chosen. Starting from data reported from NASA, CEA software is employed to compute other parameters concerning combustion process and gaseous products. Computation is carried out with the hypothesis of shifting equilibrium between combustion chamber and throat, while frozen equilibrium assumption is adopted for the remaining divergent expansion section. With the computed properties, a MATLAB code solves equations for exhaust gas pressure, density, temperature and velocity, thrust and specific impulse under the assumption of isentropic expansion process.

4.5 Performance Indicators

Air-Breathing Trajectories. To provide a comparison of the air-breathing phases of the ascent for the combined cycle engines, a preliminary, simplified trajectory design is carried out. This simulation allows to calculate the range covered

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by the spaceplane before the rocket working mode is switched on. For air-breathing engines, trajectory design and optimization is a complex problem: simplifications are needed in order to easily come to a solution. A constant dynamic pressure trajectory is selected both for RBCC engine and SABRE, in order to maintain a stable force environment during trans-atmospheric ascent. The selected value for dynamic pressure is 43000 Pa, which allows to reach Mach 5 at 25 km of altitude and to keep contained the aerodynamic loads on the structure. Once the constant dynamic pressure trajectory is fixed, Mach number and velocity at each altitude can be obtained. Knowing performance parameters of the engines and velocities computed according to the modelled trajectory, the Breguet formula can be solved for the range. A vector of different altitudes is defined to compute velocities in different points of the trajectory according to altitude and Mach number fixed by dynamic pressure. Altitude and velocity are given as input to the engines models outlined above to solve for their performance parameters in the defined locations of the trajectory. Breguet formula takes into account also aerodynamic performance of the spaceplane, which contributes to the range calculation. To provide realistic values of lift and drag coefficients, data relative to SKYLON reported by Mehta et al. are employed [7]. The sum of all the range steps returns the overall air-breathing trajectory of the engines under examination.

Preliminary Masses Sizing. The conclusive part of the work focuses on the preliminary sizing of the masses of propellant and structures of launchers. Two different configurations are adopted for combined-cycle and pure-rocket engines. The first configuration is the one considered for combined engines RBCC and SABRE: a spaceplane-like architecture is proposed. The model is based on the configuration outlined by Reaction Engines Ltd. for the SKYLON spaceplane, considered also for aerodynamic characteristics in range computation. The obtained structural index is employed to compute the total mass of the launcher at take off. The amount of propellant consumed during trans-atmospheric ascent is calculated applying Tsiolkovsky equation in discrete positions: trajectory is divided into several steps, each denoted by an altitude increment of 500 m. At each step, dynamic pressure allows to compute flight velocity and Mach number. With known flight conditions, engines models return the particular specific impulse for the portion examined. The ΔV to be considered is given by the difference between flight velocities at the end and at the beginning of the trajectory segment. The propellant mass needed for the last part of the ascent is computed taking into account a velocity variation provided by the difference between ideal target velocity and the velocity of the spaceplane at air-breathing mode cut off, with a 10% increment to take into account drag and gravity losses. For the rocket-propelled launcher, masses computation is carried out considering a conventional vertical take-off configuration. To account for gravity losses and atmospheric drag contribution, the ideal velocity variation needed for orbit insertion of 7.73 km/s is incremented to 9.38 km/s. Once the performance parameters of the engine are known, values for masses of propellant and structures are computed. A structural index of 0.1 has been selected starting from typical values employed in literature [8]. Once the overall masses are known, the conclusive calculation refers to the evaluation of the number of engines needed. The computation is carried out considering the requirement stated on the Thrust-to-Weight ratio, which must be higher than one for vertical climbing in rocket mode. For the pure-rocket launcher, the requirement is imposed as a Thrust-to-Weight ratio of 1.4 at lift off. With gross take off weight and thrust of the engine at sea level, the number of engine is computed as the closest integer to the ratio. For RBCC engine and SABRE, the same procedure is applied, but considering the configuration of the spaceplane at air-breathing mode cut off. Masses and thrusts of the engines at ignition of the pure rocket propulsive phase are known from the solution of the air-breathing trajectory simulation and from the models previously outlined.

5. Simulation Results

Engines performance is measured through specific impulse, thrust and thrust specific fuel consumption. These parameters supply a general overview of a propulsion system main qualities. Once the vector containing the altitude range considered and the dynamic pressure value for the trajectory is set, the codes of RBCC end SABRE models take these parameters as input and compute all performance indicators. Results are shown in forms of vectors and can be plotted to examine their behaviour with respect to flight altitude. The air-breathing trajectory modelled with Breguet equation is included. In this sense, an estimated ground range can be computed in order to provide a preliminary value of the duration of the air-breathing propulsive phase.

5.1 Rocket Based Combined Cycle

Specific Impulse. Specific impulse starts off with rocket-like values, below 500s. As the Mach number increases, also Is starts growing. This behaviour is the result of the initial ejector working: as the velocity of the vehicle increases, the air mass flow rate inside the engine increases, making the ramjet-like working contribution more and more significant, up to a specific impulse value of 603.7 s when the vehicle reaches Mach 2 and an altitude of 13.5 km (Figure 2). As the first transition takes place, the Is experiences a drastic growth. Ramjet mode shows the highest values of specific impulse, ranging from a maximum of 4233.9 s at starting and a minimum of 3048.2 s. Final air-

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breathing working mode powers the engine from Mach 5 at 25.5 km of altitude up to Mach 10 and 35 km. Scramjet specific impulse spans from 2983.5 s down to 1206.5 s when it is shut down to let the pure rocket working mode begin. Scramjet Is shows a steep trend and reaches quite low values for an air-breathing device. This behaviour may be explained by the low level of thrust that this propulsion architecture provides, caused by the reduced difference between the flight velocity and the exhaust gases speed. Final propulsive phase is carried out by a pure-rocket working mode, and specific impulse experiences a drastic reduction settling down to a value of 438.03 s, consistent with typical liquid rocket behaviours.

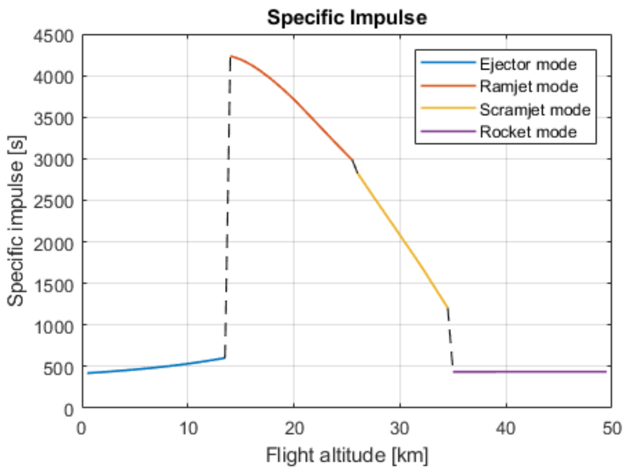

Figure 2: RBCC Specific Impulse

Thrust. In the ejector mode, thrust reaches its peak: starting from an initial value of 465.73 kN, it grows significantly reaching its maximum of 706.86 kN at primary rocket shut down (Figure 3). This trend efficaciously demonstrates the benefits of air augmented rocket propulsion: as the pressurized air mass flow rate inside the engine increases, the ramjet-like behaviour of the engine sums with the pure rocket contribution coming from the primary. After the transition into ramjet mode, the thrust experiences a drastic reduction, as the primary rocket is shut down. Ramjet thrust spans from 64.43 kN up to 85.08 kN at Mach 5, where the second transition takes place. In scramjet mode, thrust reaches its lowest values, ranging from 39.76 kN down to 9.23 kN. These extremely low values are connected to the above mentioned reduced difference between flight speed and exhaust gas velocity. Furthermore, assumptions made on the inlet lead to extremely high stagnation pressure loss, which reduces the overall effectiveness of the engine. In pure rocket mode, thrust reaches again high values, as it settles around 468.64 kN.

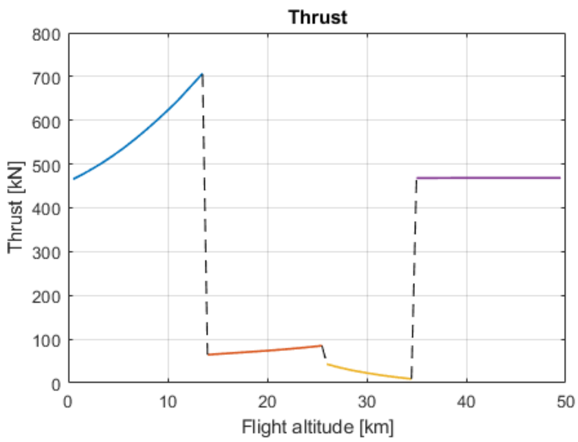

Figure 3: RBCC Thrust Profile

Thrust Specific Fuel Consumption. In ejector mode, TSFC starts with rocket-like value of $2.419 \cdot 10^{-4}$ kg/(Ns), and progressively reduces due to the benefit of the secondary airflow (Figure 4). The reduction of specific fuel consumption is connected to high thrust levels reached in this phase, but TSFC still remains high because of the presence of the primary rocket mass flow rate. Lowest values of consumption are reached in ramjet mode, in which an increase between $2.408 \cdot 10^{-5}$ kg/(Ns) and $3.417 \cdot 10^{-5}$ kg/(Ns) is noticeable. In scramjet mode, specific fuel consumption increases in connection with the above mentioned poor thrust values: it starts from $3.607 \cdot 10^{-5}$ kg/(Ns) at Mach 5, and reaches a

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value of $8.449 \cdot 10^{-5}$ kg/(Ns) before shutting down. Eventually, during pure-rocket ascent, TSFC settles again on an almost constant value of $2.327 \cdot 10^{-4}$ kg/(Ns).

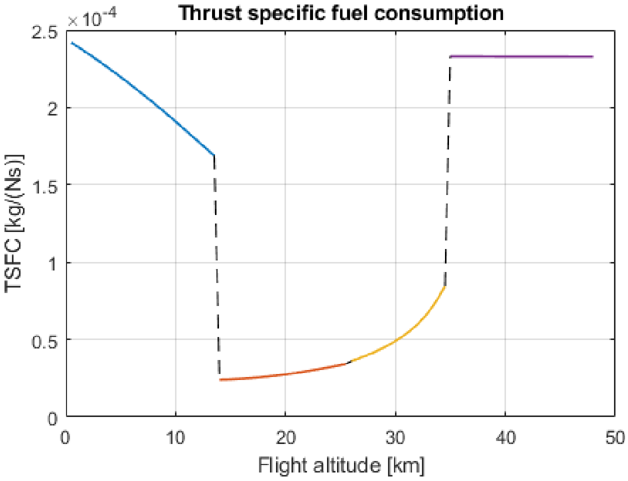

Figure 4: RBCC Thrust Specific Fuel Consumption

Air Breathing Trajectory. In ejector mode, climbing is steep: 13.5 km altitude and flight Mach number 2 are reached when the spaceplane is 117.37 km far from its take-off position. Ramjet mode covers a range of 524.47 km, bringing the vehicle to a location 641.85 km away from the runway position. Scramjet mode is the one that covers the widest range, due to the fact that climbing is much slower with respect to previous phases, in order to keep the correct constant value of dynamic pressure. In this phase of the ascent, the vehicle covers 1938.99 km, which results in a total distance of 2580.83 km covered in air-breathing engine working modes.

5.2 SABRE Engine

Specific Impulse. Air-breathing specific impulse starts from 2596.0 s and grows up to reach its maximum of 3550.5 s at 17 km altitude and a flight Mach number of 2.65 (Figure 5). Above this altitude, Is starts decreasing, reaching the value of 3493.4 s at air-breathing mode shut down. In its first propulsive mode, SABRE maintains a high specific impulse for all the duration of the phase. The pure-rocket working mode starts at an altitude of 25 km, where the flight Mach number reaches the value of 5. At transition, the engine presents a specific impulse of 423.3 s, that keeps increasing until it settles at a value of 431.0 s. Rocket-mode maximum specific impulse values are different with respect to the ones declared by Reaction Engines Ltd in SKYLON users' manual: REL reports maximum values of 3567.8 s for the first propulsive mode and 458.72 s for the second one. This mismatch may be explained considering the assumptions made during the gas expansion modelling: a conventional nozzle with fixed expansion ratios for the two phases of the ascent is adopted. Particularly, to keep nozzle exit radius into reasonable measures, the expansion ratio is fixed to 150, which limits the performance achievable by the engine.

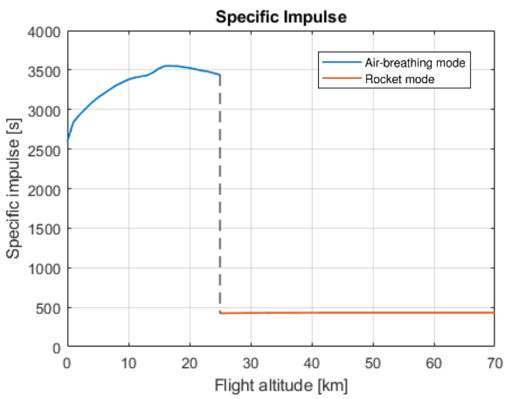

Figure 5: SABRE Specific Impulse

Thrust. In air-breathing propulsive phase, thrust starts with a value of 881.14 kN, then increases to reach its maximum of 1205.14 kN at Mach 2.65, corresponding to the maximum value of specific impulse. Thrust then reduces down to 1167.40 kN at Mach 5, corresponding to the air-breathing mode cut-off. Rocket-mode shows a slight increase in thrust, starting from 1346.52 kN and rapidly settling at 1372.28 kN (Figure 6). The main advantage of SABRE is connected

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to the very high thrust provided in air-breathing mode: in this phase, overall thrust reaches values that approach the ones that characterize the pure-rocket working mode. This advantage allows to rapidly climb in atmosphere for the first phase of the ascent.

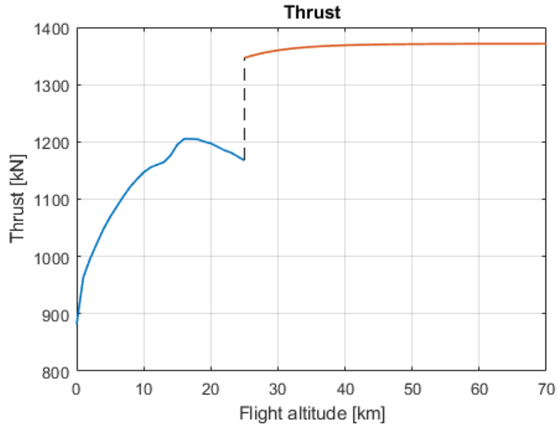

Figure 6: SABRE Thrust Profile

Thrust Specific Fuel Consumption. Thrust specific fuel consumption starts from typical values of air-breathing devices, ranging from 3.927 ·10−5kg/Ns to 2.964 $\cdot 10^{-5}$ kg/(Ns), with a minimum of 2.871 $\cdot 10^{-5}$ kg/(Ns) corresponding to maximum values of thrust and specific impulse. In pure-rocket working mode, TSFC initiates from 2.408 $\cdot 10^{-4}$ kg/(Ns) and gradually settles around 2.365 $\cdot 10^{-4}$ kg/(Ns), consistent with conventional values of liquid rocket engines. The trend is shown in Figure 7.

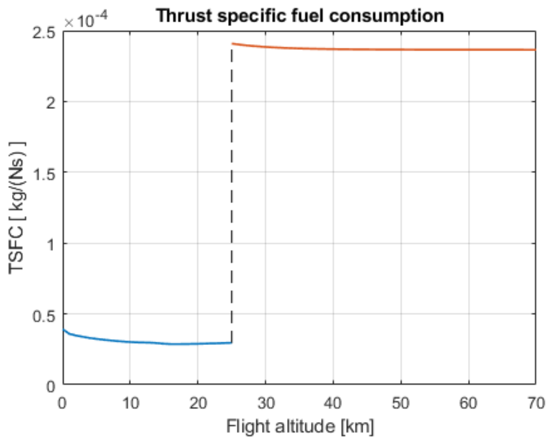

Figure 7: SABRE Thrust Specific Fuel Consumption

Air-Breathing Trajectory. For SABRE engine, air-breathing trajectory is fully covered by the only airbreathing working mode of the engine. This propulsive phase range is reduced with respect to the air-breathing ascent of the Rocket-Based Combined Cycle engine case, due to the fact that the transition to pure-rocket working mode happens at Mach 5, corresponding to an altitude of 25 km. Total range covered by SABRE airbreathing phase of the ascent is 541.61 km.

5.3 Rocket Baseline

Specific Impulse. According to typical rocket-like values, it starts from a value of 343.9 s at sea level, increase rapidly within the initial 30 km of the climb and settles on a value of 421.6 s (Figure 8). The results obtained with this model are quite below the performance parameters reported for Shuttle main engine. The discrepancy is connected to the computation based on the results provided by CEA: values of specific heats ratio are obtained assuming isentropic expansion and starting from the CP reported by the software, which assumes a frozen composition.

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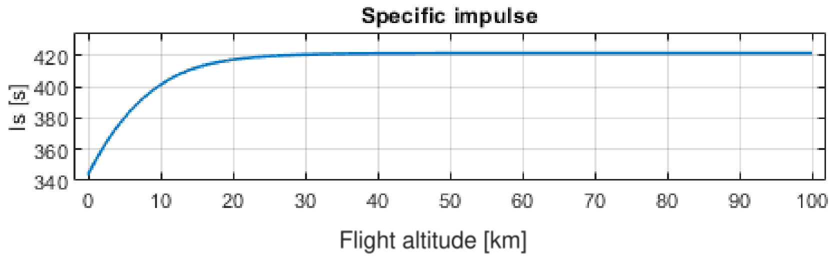

Figure 8: Rocket Baseline Specific Impulse

Thrust. Thrust starts from a value of 1866.81 kN at sea level, and increases towards its maximum value of 2288.73 kN, which is kept almost constant above 40 km of altitude (Figure 9). Such high values underline the advantage provided by rocket engines, as they are able to generate very high thrusts with respect to conventional air-breathing devices.

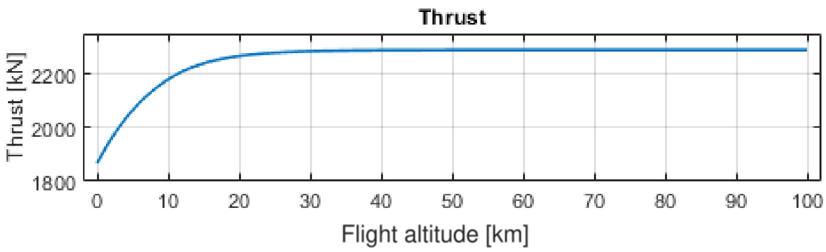

Figure 9: Rocket Baseline Thrust Profile

Thrust Specific Fuel Consumption. Maximum propellant consumption takes place in the very first phase of the ascent, as it starts from a value of 2.965 $\cdot 10^{-4}$ kg/Ns. TSFC gradually decreases during the climb, and it settles to the constant value of 2.418 $\cdot 10^{-4}$ kg/Ns (Figure 10). Results obtained for TSFC show the high expenses in terms of propellant consumption connected to rocket propulsion.

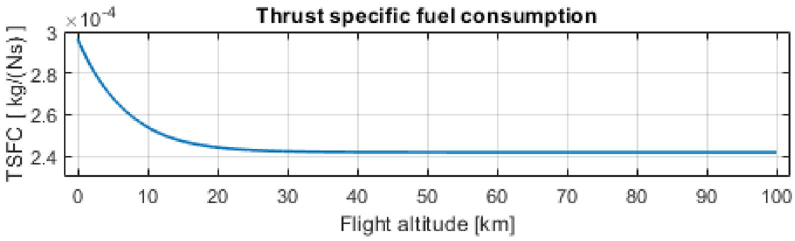

Figure 10: Rocket Baseline Thrust Specific Fuel Consumption

6. Performance Comparison

For what concerns specific impulses, Rocket-Based Combined Cycle results clearly show the advantages introduced by the air-breathing working modes. The ramjet provides an extremely favourable increase of the Is that translates into high propellant savings. High values are maintained even by the scramjet working mode, even though the performance experiences a serious reduction as the engine climbs and accelerates across the atmosphere. Rocket mode and ejector mode present low values of Is, connected to the fact that for both cases the primary rocket is active. Concerning the pure-rocket propelled launcher, specific impulse follows trends typical of conventional rockets. Results are consistent with the main drawback of rocket propulsion: such low values of specific impulse are connected with the fact that all the propellant employed for thrust generation comes from tanks mounted on board, which means that all the mass expelled by the engine comes from stored fuel and oxidizer. This is the main limitation of rocket engines, which establishes the necessity of massive systems to carry all the propellant needed for the mission. Eventually, SABRE shows expected conventional rocket-mode values of specific impulse in its second propulsion mode, whereas air-breathing phase shows very high Is, similar to those reached by the RBCC engine in ramjet working mode. Again, this fact underlines the advantages of combined air-breathing and rocket propulsion modes to improve the performance of spaceplanes in a perspective of Single-Stage-To-Orbit launches. Trends highlighted by the specific impulse analysis are confirmed by the computation of thrust specific fuel consumption: for what concerns rocket modes of all engines,

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the mass of propellant per unit thrust are one order of magnitude higher with respect to the values obtained for air-breathing modes. In air-breathing working modes of the two combined cycle engines, SABRE and RBCC ramjet mode show comparable performance, while RBCC scramjet propulsive phase demonstrates a significant increase of fuel consumption. As previously mentioned, this is connected to the low thrust provided by the engine. Analysis of thrust of RBCC engine highlights the strong differences between purely air-breathing propulsion and rocket-based propulsion. In ejector and rocket modes, thrust assumes values that are one order of magnitude higher with respect to ramjet mode, and even two orders of magnitude above values reached in scramjet phase. Again, this confirms the poor thrust capability of the scramjet working mode, that leads to slow accelerations during climbing. The pure-rocket engine based on SSME shows very high thrust values, consistent with the characteristics of rocket propulsion. SABRE thrust demonstrates the great advantage of this propulsive technology. Differently from RBCC engine, SABRE thrust in air-breathing working mode approaches the high values reached by the rocket phase. This benefit comes from the high pressure ratios reached by the compressor and the architecture of the combustion chamber, which is the same for the two working processes, the only difference being the oxidizer source. Air-breathing performance of SABRE clearly overcomes ramjet and scramjet modes of RBCC engine, representing a good candidate to rapidly cross atmosphere exploiting the advantages in terms of propellant savings connected with air-fed propulsion technologies.

Results on Masses Computation. Results concerning Rocket-Based Combined Cycle engine are reported in Table 2.

Table 2: RBCC Masses Computation Results

Parameter	Value
Velocity covered in air-breathing phase	3076.2 m/s
Velocity covered in rocket mode	5119.2 m/s
Total propellant mass	43.185 10^3 kg
Structural mass	8.262 10^3 kg
Total take-off mass	56.446 10^3 kg

With the computed results, the total mass at air-breathing modes shut down is 43.652 ·103 kg. With this value and a Thrust-to-Weight ratio of 1.4 imposed, the total required thrust at the beginning of the rocket phase of the ascent would be 656.10 kN. Since one engine provides 468.64 kN of thrust, to meet the requirement the spaceplane would need two RBCC engines, in order to obtain a thrust of 937.28 kN in pure-rocket mode, and a real Thrust-to-Weight ratio of 2.19. Table 3 shows the results of masses computation concerning SABRE.

Table 3: SABRE Masses Computation Results

Parameter	Value
Velocity covered in air-breathing phase	1495.0 m/s
Velocity covered in rocket mode	6858.5 m/s
Total propellant mass	121.660 10^3 kg
Structural mass	23.278 10^3 kg
Total take-off mass	149.940 10^3 kg

The same procedure applied for RBCC engine was adopted to compute the number of engines necessary to climb towards orbit in pure-rocket mode. The Thrust-to-Weight requirement imposed is the same mentioned in the previous subsection. The results revealed a minimum required thrust of 1919.82 kN, a condition that is satisfied by adopting two engines, with a resulting total thrust of 2742.6 kN and a Thrust-to-Weight ratio of 1.95.
Values computed for the pure-rocket SSTO launcher are reported in Table 4.

Table 4: Rocket Baseline Masses Computation Results

Parameter	Value
Velocity variation to be covered	9380.0 m/s
Total propellant mass	322.670 10^3 kg
Structural mass	32.271 10^3 kg
Total take-off mass	359.940 10^3 kg

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With the requirement on the Thrust-to-Weight imposed at lift off and 2289.2 kN of thrust provided by a single engine, the number of LREs needed for take off is three, reaching a total thrust of 5600.3 kN and a TtW ratio of 1.59.
As expected, the conventional rocket launcher presents the highest demand in terms of propellant mass. This is the result of rockets drawback above mentioned: the low specific impulse maintained for the totality of the mission comes from the necessity of draw both fuel and oxidizer from tanks. In this sense, all the propellant carried on board represents a source of mass to be accelerated during the ascent, which translates into humongous systems to climb towards orbit in a pure-rocket single stage mission. SABRE is the second most demanding engine, but the difference with the rocket launcher is already clear: the air-breathing working mode of the propulsion system allows to save a huge amount of oxidizer during the first phase of the ascent. Hence, the spaceplane has to carry a significantly reduced amount of propellant with respect to the previous case. Eventually, RBCC-propelled launcher shows the best performance among the architectures under consideration. The take off mass is almost one seventh of the one computed for the rocket launcher, and one third of the total mass resulting from computations for SABRE. The advantage of this propulsion system is the high velocity it can reach in air-breathing working phase: the transition into pure-rocket happens at Mach 10, drastically reducing the total ΔV to be covered by the final propulsive mode.
The comparison of the results concerning masses computation successfully demonstrated the advantages connected with combined cycle technologies: SABRE-propelled spaceplane has a total mass 58.34% lower than the pure-rocket one; for RBCC engine, this reduction reaches 84.31%. Such systems reduce drastically the propellant demand of launch vehicles, leading to more performing launchers in terms of consumption and dimensions. Furthermore, RBCC engine demonstrated to be the best choice between the two air-breathing devices, thanks to the ability to reach very high velocities and altitudes in scramjet mode. This advantage would lead to crown the Rocket-Based-Combined-Cycle as the most suitable propulsion systems for a SSTO launcher in a spaceplane configuration. Nevertheless, a final consideration concerning air-breathing trajectories will be carried out in order to highlight the advantages and drawbacks of both combined cycles.

Air-Breathing Trajectories. Both cases show a similar result until 25 km of altitude, where SABRE engine experiences its transition into pure-rocket working mode. RBCC engine continues its air-breathing ascent up to approximately 35 km of altitude. As previously explained, RBCC architecture appear to be the most advantageous because of its extremely low propellant demand. Nevertheless, to reach 35 km altitude and flight Mach number 10, this device covers a range above 2500 km. Such an extended range and the very low thrust provided by ramjet and scramjet working modes of the engine translate into the necessity of long residence time in atmosphere at very high flight velocities. In this sense, the spaceplane structures could experience intense stresses connected to heating of the surfaces, and the overall time-to-orbit of the mission would be significantly higher with respect to conventional launchers and SABRE-propelled vehicles. In conclusion, both SABRE and RBCC engine demonstrated their advantages for a single-stage LEO mission. The best solution between this two candidates has to be selected according to requirements connected to surface heating and time constraints to reach orbit, for which SABRE demonstrates a better performance.

7. Conclusions

First aim of this analysis was to provide an overview of the attempts made to develop SSTO vehicles. Particular attention was given to different propulsion architectures proposed through years; a preliminary qualitative comparison allowed to choose Rocket-Based Combined Cycles, Synergetic Air-Breathing Rocket Engine as best candidates among all technologies in perspective of a LEO flight, along with conventional LREs. Numerical models were developed to compute specific impulses, thrust profiles and propellant consumption of the different working modes of each propulsion system during its ascent. To account for varying conditions of the working environment, constant dynamic pressure trajectories were modelled for air-breathing configurations. Simulations results managed to demonstrate the outstanding advantage coming from combined cycles employment: specific impulses of air-breathing working modes are one order of magnitude higher with respect to conventional rockets' ones. This characteristic translates into a great reduction of fuel consumption. These engines put together the best qualities of rockets and air-breathing devices: propulsive phases that exploit air for combustion do not require stored oxidizer, resulting in a drastic reduction of stored propellant. Furthermore, rocket working modes provide thrust in vacuum conditions, necessary to fulfil in-space operations. Performance results coming from simulations were employed to carry out a preliminary computation of total take off masses and propellant fractions. A spaceplane configuration has been employed for combined cycle technologies, whereas a conventional vertical take off slender body has been considered for the all-rocket powered vehicle. Results demonstrated that the most demanding propulsive solution is the liquid rocket engine. Both SABRE and RBCC show a drastic reduction of gross lift off masses and propellant loads. The total take of mass of SABRE is 58.34% lower than the one obtained for the rocket baseline. RBCC engine gave even better results: the total take off mass of the spaceplane propelled by the rocket-based combined cycle is 84.31% smaller with respect to the rocket powered launcher. Furthermore, propellant mass fraction at take off is significantly reduced by air-breathing

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technologies: for the rocket launcher, 89.65% of lift off weight consists of propellant, while for SABRE and RBCC the propellant weight fraction amounts to 81.14% and 76.40% respectively. Results clearly show that the only possible solution to reduce launch costs by employing a reusable SSTO is to select one of the combined cycle engines. RBCC seems to be the best candidate, since the high velocity reached in scramjet mode lowers the remaining ΔV covered by the rocket working mode. Eventually, a comparison between the air-breathing trajectories of the combined cycle engines was carried out. Results show that SABRE covers a range of 541.61 km before switching to the final phase of the ascent, whereas RBCC engine needs 2580.83 km to reach the flight conditions at which the rocket mode can be activated. In this sense, the Rocket-Based Combined Cycle presents a drawback: such long distances covered in air-breathing mode translate into long residence time in the atmosphere. Furthermore, the low values of thrust provided by ramjet and scramjet working modes contribute to the increase of the overall time-to-orbit. In this sense, accounting for propellants saving, high thrust and low time-to-orbit, SABRE engine demonstrated to be the ideal solution for a SSTO LEO flight.

7.1 Future Developments

This work provided a preliminary analysis and comparison of different architectures for a SSTO vehicle. Strong assumptions were adopted in order to develop simplified analysis tools to be employed for the simulations. Future works concerning RBCC engines should focus on improving this model by accounting for real gas behaviours, friction losses and transients of transitions from one working mode to another. Furthermore, the scramjet working mode could be improved by developing a suitable inlet that allow to reduce the performance loss by increasing the total pressure recovery. Once the structural configuration is fixed, the correct drag coefficient and reference surfaces can be computed: in this sense, the force balance in longitudinal direction could establish a new requirement on the number of engines necessary to successfully carry out the conclusive phase of the air-breathing ascent. Regarding SABRE, future developments might be carried out once details on components are available. The scarce knowledge and data availability of this technology allowed to realize a model which strongly simplifies the actual working mode of the engine, resulting in a approximated analysis. Nevertheless, the simulations managed to demonstrate the advantages coming from this technology, providing results that match well with performance parameters found in literature. Furthermore, a more detailed study could be carried out including the modelling of the contribution coming from bypass ramjet burners and the advantage established by the E-D nozzle. Eventually, trajectories optimization, structural and aerodynamic design of spaceplanes might result in a successful conceptual design of a revolutionary SSTO vehicle. A study on the development and operative costs could conclude the overview on spaceplanes technologies, hopefully establishing a turning point towards a easier and more accessible way to reach space.

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