



ASME Accepted Manuscript Repository

Institutional Repository Cover Sheet

Cranfield Collection of E-Research - CERES

ASME Paper

Title: Synergetic and performance characteristics of a high-speed pre-cooled propulsion concept

Authors: Athanasios Chatzistefanou, Spyros Tsentis, Anestis Kalfas

ASME Journal

Title: Journal of Engineering for Gas Turbines and Power

Volume/Issue: Volume 147, Issue 2

Date of Publication (VOR* Online): 27 September 2024_

ASME Digital Collection <https://asmedigitalcollection.asme.org/gasturbinespower/article/147/2/021016/1203566/Synergetic-and-Performance-Characteristics-of-a>

DOI: <https://doi.org/10.1115/1.4066352>

*VOR (version of record)

Synergetic and performance characteristics of a high-speed pre-cooled propulsion concept

Athanasios Chatzistefanou*

Laboratory of Fluid Mechanics and
Turbomachinery,
Aristotle University of Thessaloniki,
Thessaloniki GR-54124, Greece
Email: chatzistef@meng.auth.gr

Spyros Tsentis

Centre for Propulsion and
Thermal Power Engineering,
School of Aerospace Transport
and Manufacturing,
Cranfield University,
Bedfordshire MK43 0AL, UK,
Email: spyros.tsentis@cranfield.ac.uk

Anestis Kalfas

Laboratory of Fluid Mechanics and
Turbomachinery,
Aristotle University of Thessaloniki,
Thessaloniki GR-54124, Greece
Email: akalfas@auth.gr

ABSTRACT

Pre-cooled air-breathing cycles are promising candidates to power future high-speed flight as well as Single-Stage-To-Orbit vehicles, due to their increased efficiency over contemporary propulsion systems and launch vehicles. These concepts usually feature complex interactions in the synergy of their thermodynamic cycles, which are not yet well understood, especially at off-design conditions. In this study, a performance model for a pre-cooled, hybrid, air-breathing, rocket-cycle engine is developed for its air-breathing mode of operation. One-dimensional thermodynamic modeling is employed within a component-level approach, to evaluate the performance and operation of the cycle under investigation in the range of $1.35 \leq M_\infty \leq 5$ and conditions of up to 26 kilometers altitude. The model

*Corresponding author

is validated quantitatively and qualitatively for both design and off-design conditions. The specific impulse I_{sp} and specific thrust F_{spec} , as predicted by the model, agree within less than 5% for the design-point conditions at $M_\infty=5$. At off-design conditions, the model captures the trend of I_{sp} and agrees within less than 1% with respect to the data for the maximum value of I_{sp} . The maximum gross thrust F_{gross} point is predicted correctly at $M_\infty = 4$. The fundamental operating principles and synergetic characteristics of the engine at design and off-design conditions are investigated and reported. Parametric analyses quantify the influence of the engine's parameters on the leading performance metrics. A model which does not feature a bypass duct is created and compared for the same inflow conditions and mission profile. It is found that the engine without the bypass duct exhibits reduced specific impulse which can be up to 32% lower at off-design conditions. In addition, the corresponding fuel mass flow rate to achieve the same mission is increased by a factor of 1.5. It is demonstrated that the overall trend of engine efficiency cannot be properly captured without modeling of the bypass duct, especially at the region of $M_\infty < 3.5$, where the ramjet-like operation is critical. This highlights the importance of the bypass, which is typically neglected in the modeling of such high-speed, combined-cycle systems.

NOMENCLATURE

Roman letters

c_p Specific heat capacity at constant pressure [$JKg^{-1}K^{-1}$]

F Thrust [kN]

I_{sp} Specific impulse[s]

R Gas constant [$Jkg^{-1}K^{-1}$]

M Mach

TPR Total Pressure Recovery

O/F Oxygen to Fuel ratio

Q Heat transfer [kW]

W Work [kW]

Greek letters

γ Heat capacity ratio

η Efficiency

ϕ Equivalence ratio

Subscripts

comp Compressor

e Exit

mech Mechanical

prod Products

poly Polytropic

prop Propulsive

react Reactants

s Stoichiometric

spec Specific

turb Turbine

therm Thermal

∞ Free stream

0 Total

Superscripts

0 Reference conditions

1 INTRODUCTION

1.1 Background

The development of a Single-Stage-To-Orbit (SSTO) concept or a vehicle that can travel at hypersonic speeds, has been envisioned from the early days of space exploration and high-speed propulsion respectively. The successful development of this type of vehicle could revolutionize the aerospace industry. These vehicles share the need to operate from static conditions at sea level up to hypersonic speeds near vacuum.

The propulsion system of such a vehicle utilizes atmospheric air for part of the mission, or else it must carry onboard oxidizer. The latter solution, typically used in rocket propulsion, is significantly less efficient and heavier than air-breathing propulsion, making it hard to be economically viable. Typical concepts of high-speed air-breathing propulsion, namely ramjet [1] and scramjet [2] engines, showcase improved efficiency over rocket propulsion while employing simple cycles. The major drawback of these cycles is that they cannot produce thrust at static conditions and another system is required to propel them to their starting speed. A turbine-based combined cycle (TBCC) incorporates turbomachinery to solve this problem [3]. These cycles typically combine a turbojet engine with a high-speed system, such as a ramjet or a scramjet. The turbojet engine propels the vehicle until the necessary operating conditions of the high-speed engine are reached. However, the turbojet engine can be used only for a narrow Mach range, because the air temperature downstream of the intake rises rapidly with the Mach number, reaching the material limits of turbomachinery.

A technology that enables turbomachinery components to be used in a wider range, further exploiting their improved performance, is to cool the air, downstream of the intake. The first cycle that was designed based on that concept was the Liquid Air Cycle Engine (LACE) [4]. LACE was designed to cool the air until liquefaction and then to be fed to a conventional rocket combustor. This engine carried onboard liquid oxygen and could operate outside the atmosphere. An advancement on the LACE concept was the Air Turbo Ramjet Expander cycle (ATREX) [5]. ATREX was designed as the propulsion system of a hypersonic aircraft, or a first stage of a Two Stage To Orbit vehicle. It cools the airstream to a higher temperature and avoids liquification. The air is compressed by turbomachinery until ram air compression is sufficient for combustion. Hydrogen is then injected into the combustion chamber, after cooling the airstream. The Synergetic Air Breathing Engine (SABRE) [6] introduced an intermediate helium cycle to improve safety and overcome material limitations. SABRE utilizes cryogenic hydrogen for cooling and as fuel, but the pre-cooling process is established through the closed helium cycle. This design requires more heat exchangers and turbomachinery but promises improved efficiency and significantly improved thrust to weight ratio [7]. A derivative of this engine, namely the SCIMITAR concept, has been

proposed, which is able to cruise in hypersonic conditions [8]. The aforementioned concepts, rely on the cryogenic properties of the fuels but there are new concepts emerging that employ the pyrolysis process of hydrocarbon fuel, such as kerosene, to cool the air [9, 10].

1.2 Air-breathing Pre-cooled Cycles

Air-breathing, pre-cooled concepts employing a closed helium cycle have received increased interest due to their improved performance characteristics. Key aspect of their design is the inclusion of a bypass duct that operates similarly to a ramjet engine. The bypass duct increases the thermodynamic efficiency of the system by burning excess fuel while it reduces aerodynamic losses. Several aspects of the design have been investigated using various degrees of fidelity on a component level approach. The performance of the cycle and the importance of certain parameters during the ascent have been studied in [11, 12, 13, 14, 15, 16, 17, 18, 19]. It was found that the pre-cooling temperature of the air and the ratio of the flow rate between hydrogen and helium have significant effect on the performance. Traditional cycle analysis methods, such as exergy analysis, have been applied to highlight the source of inefficiencies of the engine [12, 13]. This showed that 75% - 84% of entropy generation is contributed by the pre-cooler and regenerator. Increasing the pre-cooling temperature of the air and replacing the single pre-cooler branch with a high and low temperature branch can significantly decrease entropy generation and fuel consumption. Based on these observations, the use of multiple regenerator and/or pre-cooler branches has been proposed, achieving higher specific impulse in the range of 70%-100% [17]. The effect of the number of segments of the pre-cooler and regenerator have also been investigated in [14, 15]. These studies revealed that an optimal configuration and flow control to reduce compression work exists, and when applied it increases exhaust speed by 7.7%. However, while the use of multiple branches improves efficiency, it narrows the operating range of the engine. The conditions of the air entering the pre-cooler change constantly throughout the ascent and are greatly affected by the intake. This interaction has been studied separately via numerical simulation and a major non-uniformity of temperature and pressure distribution was found [20]. During the transition from Mach 5 to Mach 2 the percentage of the total pressure loss occurring in the pre-cooler compared to the overall system, increases from 3% to 48.4% [21]. Additional numerical analysis revealed

that reducing the cone angle and increasing the internal contraction ratio can reduce the total pressure loss [22]. Indicatively, the total pressure recovery increased by 21% at Mach 3, although it decreased by 3% at Mach 5.

The control methods of the engine have been investigated in the literature [16, 17, 18]. It was shown that the performance of the engine can be controlled via the pressure ratio across the hydrogen turbines and the pre-burner combustion temperature [16]. The regulation of thrust through the bypass throat area and through the amount of helium circulated were also examined [17]. The latter method showcased a wider range of control, approximately $\pm 28\%$ compared to $\pm 11\%$ of the reference thrust, while the bypass method would be easier to implement. Using as control variables of the engine the fuel flow through the pre-burner and core combustor, the total fuel flow of the engine and the throat area of the core nozzle has also been explored [18]. These parameters also proved sufficient to control the engine. Finally, the effect of the bypass duct on the performance of the cycle, has been studied at the engine's design point and it was determined that it improves overall efficiency by 7% [19].

1.3 Scope Of Work

This paper presents an investigation on the performance characteristics of an air-breathing pre-cooled propulsion concept, which employs a closed helium cycle. A component level approach is implemented, utilizing one-dimensional thermodynamic relations. The modeling approach aims to capture the operating principles and synergies emerging from the fundamental physical processes and design-driven parameters of the system. An emphasis is given on the parameters that affect the specific impulse, thrust and the thermal loads of the heat exchangers of the engine. Finally, a model without the bypass duct is created and compared with the engine under different scenarios. The effect of the bypass duct has been previously investigated only at the design point [19]. To the authors's best knowledge the effect of the bypass duct for both design and off-design conditions has not been investigated in the open literature.

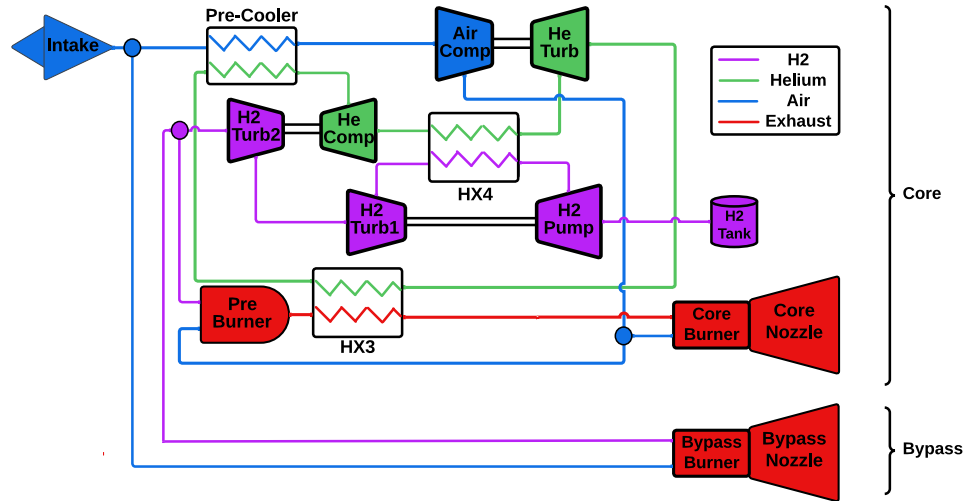


Fig. 1: Air-breathing Pre-cooled Cycle

2 METHODOLOGY

2.1 Cycle Operating Principle

The engine cycle that is investigated in this study is depicted in Fig. 1. It is inspired by the concept of the Synergetic Air-Breathing Rocket Engine (SABRE) which is currently under development [23, 24, 7, 25, 26, 6, 27]. The performance modeling is carried out based on its mission profile, adopted from [23] and shown in Fig. 2. The engine operation comprises two modes, namely an air-breathing mode and a rocket mode. In the air-breathing mode, the engine must operate from static conditions up to the design point of Mach 5 at an altitude of 26 km [26]. To operate in this range three working mediums are utilized, namely atmospheric air as oxidizer, liquid hydrogen as fuel and helium. The latter is employed in an intermediate cycle, cooling the air and driving part of the engine's turbomachinery. The system incorporates a bypass duct in order to reduce parasitic drag losses and to increase thermodynamic efficiency by burning excess fuel. The bypass duct operates similarly to a ramjet engine [1]. After air-breathing operation and speeds above Mach 5, the engine transitions to the rocket mode, in which liquid oxygen is utilized as an oxidizer. Only the air-breathing mode of the engine is investigated herein, which constitutes the innovative and challenging design aspect.

The components incorporated in the cycle are illustrated in Fig. 1. A variable-geometry, conical

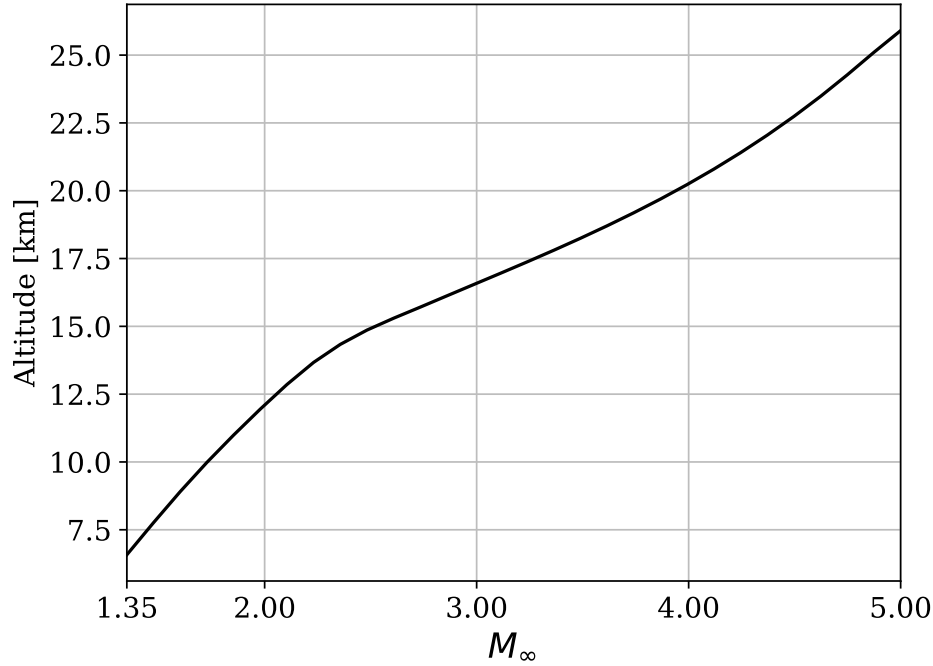


Fig. 2: Air-breathing mode trajectory [23]

and axisymmetric intake is employed. The cone translates along the symmetry axis to maintain the shocks attached to the cowl lip. The air that goes through the intake is split, and excess air is diverted to the bypass duct. The air that enters the core is cooled to temperatures of 140 K in the Pre-Cooler (PC) through the cold helium stream [24, 6]. The closed helium cycle includes all the heat exchangers of the engine, and has a major impact on the engine's performance. Helium is cooled approximately at 65 K, by the cryogenic hydrogen at heat exchanger 4 (HX4). Afterwards, helium is circulated by the Helium Compressor (HeC), which increases its pressure in the range of 190 bar at the inlet of the PC [12]. At the design point helium exits the PC at 950 K, which is the maximum allowable temperature due to material limitations [6]. The Helium Turbine (HeT) is designed to operate with constant inlet temperature of 1080 K [27]. To achieve this, helium is heated in the heat exchanger 3 (HX3). After expansion in HeT the helium re-enters HX4, closing the cycle.

Liquid hydrogen is stored at cryogenic conditions, specifically at 20 K and 3 bar [12]. It is circulated by the Hydrogen Pump (H2Pump), which increases the pressure to a constant value

of 260 bar throughout the air-breathing operation [12]. Following the heat exchange process at HX4, it expands at the Hydrogen Turbines 1 and 2 (HT1 and HT2) which drive the H2Pump and HeC, respectively. During the air-breathing operation, the amount of hydrogen required for heat load dissipation is higher than the quantity needed for stoichiometric combustion in the core [26]. Therefore some amount is diverted from the core to the Bypass Burner (BB). The amount that is not diverted to the BB, is injected in the Pre-Burner (PB), where fuel-rich combustion takes place.

Downstream of the PC, air is compressed by the AirCompressor (AC) at a constant pressure of approximately 150 bar throughout the air-breathing ascent [24, 6]. Exiting the AC, air is again split into two streams. One stream is directed to the Core-Combustor (CC) and the rest to the PB. The fuel-rich mixture burns in the PB, and the exhaust gases heat the helium at HX3. Following, they are directed to the CC where they mix with the primary air-stream and combust in fuel-rich conditions. The combustion occurs at a constant temperature of 2200 K, which is a typical temperature found in rocket engines [28]. Finally, the exhaust gases expand at the downstream CoreNozzle (CN). The bypass duct operates like a ramjet engine, where the diverted air stream from the intake burns with the excess hydrogen in the BB, and expands through the BypassNozzle (BN).

The model developed in this work, is used to investigate the cycle in the range of $1.35 \leq M_\infty \leq 5$. At $M_\infty = 1.35$ it is certain that conical shockwaves are formed in the intake cone according to compressible conical flow theory [29]. Operation from take-off conditions (Mach 0.5 [23]) up to $M_\infty = 1.35$ has a short duration and highly unstable phenomena dominate the operation of the intake. It is considered out of scope to investigate the operation in this time-limited range, as it has a minor effect on the cycle's overall performance. Additionally the operation during these conditions is strongly connected to the geometrical aspects of the intake which are unknown.

2.2 Thermodynamic Modeling

Fluid Properties

The operating mediums are modeled as semi-perfect gases. Specific heat capacity, as well as specific enthalpy and entropy are a function of static temperature. Shomate's polynomial relations are utilized to calculate their properties [30]. However, cryogenic hydrogen requires special treat-

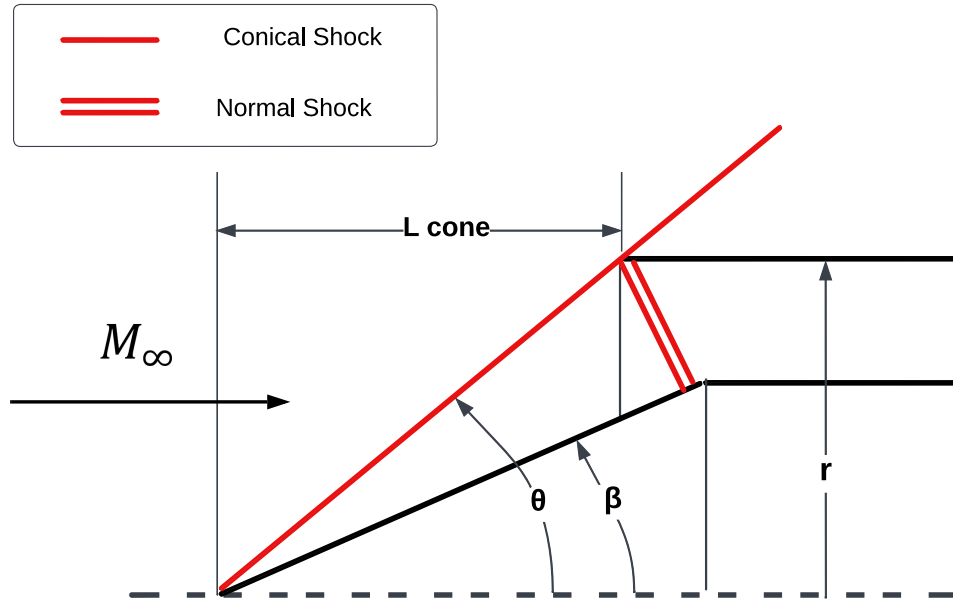


Fig. 3: Schematic of employed intake-shock topology

ment because of its liquid state. The thermodynamic properties of liquid hydrogen are acquired using CoolProp [31], which employs the equations of state from [32].

Intake

The shock structure in the intake is modeled by a conical shock attached to the lip of the nacelle, followed by a normal shock [7], as seen in Fig. 3. The half-angle of the cone is equal to 20 degrees [26]. The conical shock properties are obtained by the Taylor-Maccoll equations [29], presented in Eqs.(1)-(3), and have to be solved numerically. Normal shock properties are obtained through the application of shock-expansion theory [29]. The recovery conditions of the intake in the air-breathing operation are presented in Fig. 4.

$$\begin{aligned} \frac{\gamma-1}{2} \cdot \left[1 - V_r'^2 - \left(\frac{dV_r'}{d\theta} \right)^2 \right] \cdot \left[2V_r' + \frac{dV_r'}{d\theta} \cdot \cot \theta + \frac{d^2V_r'}{d\theta^2} \right] \\ - \frac{dV_r'}{d\theta} \left[V_r' \frac{dV_r'}{d\theta} + \frac{dV_r'}{d\theta} \frac{d^2V_r'}{d\theta^2} \right] = 0 \end{aligned} \quad (1)$$

$$\frac{V}{V_{max}} = V' = \left(\frac{2}{(\gamma - 1) \cdot M^2} + 1 \right)^{-\frac{1}{2}} \quad (2)$$

$$V_\theta = \frac{dV_r}{d\theta} \Rightarrow \frac{dV'_\theta}{d\theta} = \frac{d^2 V'_r}{d\theta^2} \quad (3)$$

where $V' = V/V_{max}$ is the non-dimensional velocity, V_r, V_θ is the radial and normal velocity components, respectively.

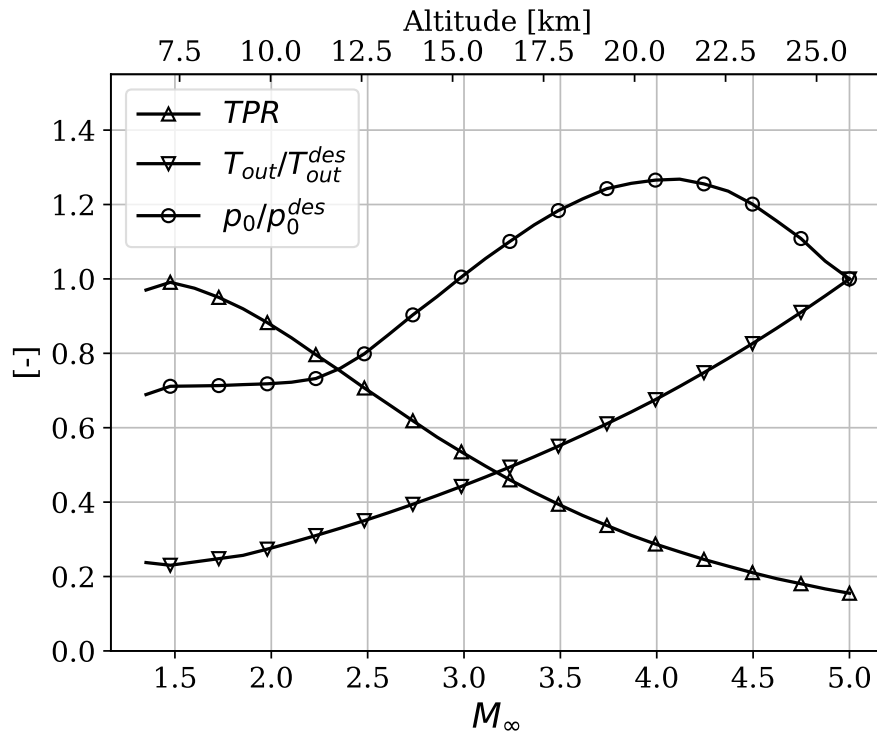


Fig. 4: Intake Recovery Conditions

Turbomachinery

Polytropic relations are applied to model the turbomachinery of the engine [33]. Constant polytropic efficiencies of 0.88 and 0.9 are employed for the compressors and turbines respectively,

adopted from [34]. Furthermore the mechanical efficiency is equal to $\eta_{mech} = 0.99$. The outlet temperature of the compressors is calculated by Eq. (4), while the outlet pressure is defined by operating principles of the engine cycle. The power of the compressors is calculated using Eq. (5).

$$T_{out} = T_{in} \cdot \left(\frac{P_{out}}{P_{in}} \right)^{R/(c_p \cdot \eta_{poly})} \quad (4)$$

$$\dot{W} = \dot{m} \cdot c_p \cdot (T_{out} - T_{in}) \quad (5)$$

Accordingly, the outlet temperature of the turbines is calculated through the work required by the coupled compressor, as seen in Eq. (6). The outlet pressure is calculated by Eq. (7).

$$T_{out} = -\frac{\dot{W}_{comp}}{\eta_{mech} \cdot (\dot{m} \cdot c_p)} + T_{in} \quad (6)$$

$$P_{out} = P_{in} \cdot \left(\frac{T_{out}}{T_{in}} \right)^{c_p/(R \cdot \eta_{poly})} \quad (7)$$

It is noted, that for increased accuracy and higher fidelity description at off-design conditions of the turbomachinery parts, the use of performance maps is typically employed. However, in the context of this work, performance maps are not considered for off-design operation. This is facilitated due to the concept itself, which allows the turbomachinery to operate at the vicinity of their design-point conditions, as a result of the employed heat exchangers [7, 6]. Additionally, the use of a higher-fidelity approach at off-design conditions for the turbomachinery components was not deemed essential based on the obtained results (see section 3.1 for validation) and served to reduce the computational cost and model complexity. The associated compression and expansion

ratios of the rotating components are allowed to vary at off-design conditions and are only partially defined by specific design requirements related to this engine concept [7, 6, 34] (e.g., helium inlet pressure for successful operation of the pre-cooler, combustion chamber pressure for rocket nozzle operation).

As discussed previously, the hydrogen pump operates with liquid hydrogen, therefore requires a different approach. From the literature [35, 36], it is known that hydrogen pumps are designed to avoid cavitation (i.e., phase change) during compression. In order to comply with this principle, it is adopted that the pump operates with constant exit conditions, these of the design point, which are known. Accordingly, the temperature of the hydrogen at exit is constant at 35 K [24]. The power consumed by the pump is calculated by applying the first thermodynamic law.

Heat Exchangers

The first thermodynamic law (Eq. (8)) is applied to model the heat transfer process. A constant total pressure drop coefficient is employed (Eq. (9)), which is calculated from available literature data at the design point [12, 18]. The heat exchanger effectiveness for HXs 3 and 4 is set to 0.96 [11, 13], while for the PC, the effectiveness is not directly involved in the solution procedure, owing to its relation to the mission-specific parameters of the air stream. In specific, at the PC, air is cooled at a constant temperature of 140 K, while the helium outlet temperature, $T_{out,He}^{PC}$, has a maximum 950 K at the design point [34]. Furthermore, the helium inlet pressure is constant at 190 bar, due to the HeC [6]. Similarly, the operation of the H2Pump imposes steady inlet conditions on the hydrogen at HX4. At HX3, the outlet temperature of helium is kept constant at a value of 1080 K, as demanded by the operation of HeT [27]. It is noted that a specific heat exchanger type is not considered in this study, to reduce model's sensitivity to heat exchanger design parameters and due to limited data available in the public domain related to HXs design specifications and materials. Besides, the obtained results do not indicate that a more detailed approach is critical.

$$\dot{Q}_{hot}^{PC} = \dot{Q}_{cold}^{PC} \quad (8)$$

$$P_{out}^{hot} = \sigma_{hot} \cdot P_{in}^{hot} \quad (9)$$

$$P_{out}^{cold} = \sigma_{cold} \cdot P_{in}^{cold} \quad (10)$$

Combustors

The engine includes three combustors, namely CC, PC, and BB. The combustion process is modeled using the same approach for all burners. Constant total pressure drop coefficient is employed of $\sigma_{Burner} = 0.95$ (Eq. (11)). The combustion is treated as perfect and the composition of the combustion products is directly obtained from the stoichiometry of the one-way reaction. It is considered that the composition of the gas mixture exiting each burner does not change. Additionally, the exhaust gas is treated as thermally perfect and variable gas properties are employed as a function of its static temperature and composition [30]. The equilibrium of the first thermodynamic law is used, utilizing the formation enthalpy of the elements as seen in equation Eq. (12). The outlet temperature is calculated by solving numerically this equation (Eq. (12)) for T_{exh} .

$$P_{0_{exh}} = \sigma_{Burner} \cdot P_{0_{in}} \quad (11)$$

$$\begin{aligned} \dot{H}_{react} = \dot{H}_{prod} \Rightarrow \\ \sum \dot{m}_i^{react} \cdot Dh_f^0(T_{react}) - \sum \dot{m}_i^{prod} \cdot Dh_f^0(T_{prod}) \\ = f(T_{exh}) \end{aligned} \quad (12)$$

Nozzles

Expansion in the convergent-divergent nozzles of the engine (CN, BN) is modeled as an isentropic and adiabatic process, while a total pressure drop coefficient of 0.95 is also employed. The composition of the combustion products does not change throughout the expansion in the nozzles. The Mach number at the inlet of the nozzle is considered to be zero [29]. Furthermore,

the criterion developed by Summerfield et al. [37] is applied in order to approximate the effects of flow separation. Additionally, at the design point the exit pressure of the nozzle is equal to the free stream pressure (i.e., nozzle operates at ideal expansion state [38]). Exit flow conditions are calculated by applying isentropic flow relations. The throat surface area (A^*) and exit area (A_e^D) are calculated at the design point. Finally, the standard gross thrust is calculated by Eq. (13)

To calculate the exit flow conditions at off-design points, initially the exit Mach number is calculated assuming no separation, by solving numerically Eq. (14), for M_e . Then, the corresponding P_e is obtained through isentropic flow relations. The Summerfield criterion is applied, as presented in Eq. (15). If the flow is separated, exit flow conditions are re-calculated for the P_e imposed by the criterion. Finally, A_e is calculated by A^* and $\frac{A_e}{A^*}$ using Eq. (16).

$$F = \dot{m}V_e + A_e(P_e - P_\infty) \quad (13)$$

$$A_e^D - A^* \left(\left(\frac{\gamma}{2} \right)^{\frac{\gamma+1}{2(\gamma-1)}} \left(\frac{(1 + \frac{\gamma-1}{2}M_e^2)^{\gamma+1}(2(\gamma-1))}{M_e} \right) \right) = 0 \quad (14)$$

$$P_e = \begin{cases} P_e & \text{if } P_e \geq 0.3 \cdot P_\infty \\ P_e = 0.3 \cdot P_\infty & \text{if } P_e < 0.3 \cdot P_\infty \end{cases} \quad (15)$$

$$\frac{A}{A^*} = \left(\frac{\gamma+1}{2} \right)^{-\frac{\gamma+1}{2(\gamma-1)}} \cdot \frac{\left(1 + \frac{\gamma-1}{2} \cdot M_e^2 \right)^{\frac{\gamma+1}{2(\gamma-1)}}}{M_e} \quad (16)$$

2.3 Control Scheme

A control scheme must be implemented to facilitate the mathematical closure of the model, as well as to establish the engine's operational framework. It is noted that the term control scheme in this study is used to describe parameters that are critical for the engine performance both in terms of modeling as well as cycle's operation. This is formed by selecting certain parameters of the engine to be constant and others to be minimized. In specific, the total mass flow that goes through the intake is decoupled from its geometry. This also allows for higher flexibility in the model through a geometry-independent approach. Furthermore, the corrected mass flow of the AC, defined as $\dot{m}_{corr} = \dot{m} \left(\sqrt{\frac{T}{T_{ref}}} / \frac{p}{p_{ref}} \right)$, experiences variations which are below 20% of the take-off value [27]. Consequently, to control the air mass flow through the core, the corrected mass flow is considered to be constant throughout the ascent. The quantity of fuel injected into the core and the ratio of air diverted in the PB are implicitly controlled by the combustion temperature of the CC. The combustion temperature for both CC and BB is kept constant throughout the operation, at 2200 K. The quantity of air that goes through the bypass duct is controlled by the stoichiometry of the combustion. Finally, the total hydrogen mass flow is minimized at each operating point.

2.4 Numerical Approach

The system of equations comprising the model is divided into two subsystems, to be solved separately. The division allows the subsystems to be studied individually under different conditions, while it does not inversely affect the synergies of the cycles. The subsystems can be seen in Figs. 5a and 5b. This results in each subsystem having two independent variables. These are, the mass flow rates of helium and hydrogen for the 1st (i.e., $\dot{m}_{He}$ and $\dot{m}_{H_2}$). Similarly, for the 2nd subsystem the independent variables are the equivalence ratio of the PB, defined as $\phi = (O/F)_S / (O/F)$, and the ratio of the hydrogen mass flow rate of the core over the total (i.e., ϕ_{PB} and $\dot{m}_{H_2}^{Core} / \dot{m}_{H_2}^{tot} \equiv \text{HFR}$). The objectives functions created, (Eqs. (17),(18)), are composed by equilibrium of the components, variables for minimization and other parameters used to aid

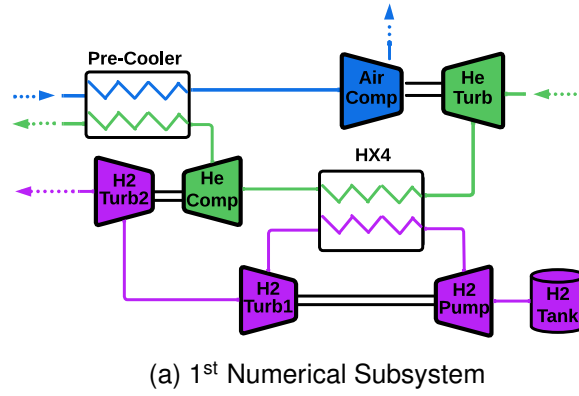
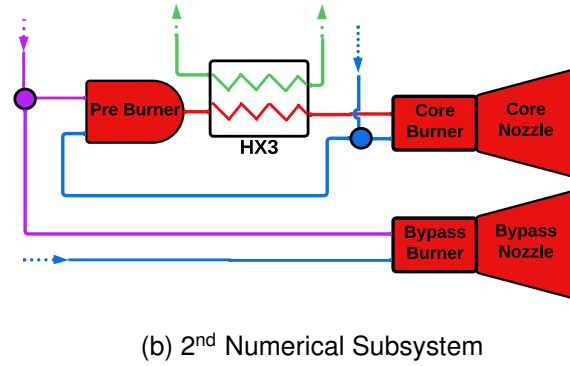


Fig. 5: Numerical subsystems employed for the solution the model

convergence.

$$F_1(\mathbf{x}) = f(E_{PC}, \dot{m}_{H_2}, T^{HX4}, T^{PC}) \quad (17)$$

$$F_2(\mathbf{x}) = f(r_{CC}, T^{PB}) \quad (18)$$

where E_{PC} is the equilibrium of the heat exchange process of the PC and r_{CC} is the residual between the target combustion temperature of the CC (2200K) with the value calculated at each iteration of solution process. The temperatures, namely T^{HX4} , T^{PC} , T^{PB} , are employed to aid in convergence, in combination with their values at the previous operating point.

The subsystems are solved sequentially. In this work, a stochastic algorithm is selected for the solution of each subsystem, because of its better performance in terms of stability and variation in the results, compared to gradient descent based algorithms. The selected algorithm is the Differential Evolution [39] method as implemented in the Optimization package of SciPy [40]. The solution process starts at the design point. Through an iterative procedure the air mass flow of the core is calculated, to produce 1050 kN of gross thrust [19]. Then, the solution proceeds to the off-design points, backtracking to $M_\infty = 1.35$.

3 RESULTS

3.1 Model Validation

In order to have confidence in the results of the model, it is critical to examine its soundness and accuracy against publicly available data. This is facilitated through a validation activity as presented in Table 1. At the design point, the specific impulse I_{sp} , is equal to 1839 s as reported in the open literature [24]. Additionally, the maximum I_{sp} during the air-breathing mode is equal to 3200 s [6]. The I_{sp} at the design point produced by the current model is equal to 1919 s, while the maximum is 3171 s. This corresponds to errors of approximately 4.3% and 0.9%, respectively. The reasonably good agreement between the model and publicly available data, in terms of design point and maximum performance metrics, is a good indication for the credibility of

Table 1: Specific impulse comparison with open literature data

I_{sp}	Model	Ref. [24, 6]	Error
Design point	1919 s	1838 s	4.30%
Maximum	3171 s	3200 s	-0.90%

Table 2: Core performance comparison with open literature data

$M_\infty = 4$	Model	Ref. [26]	Error
F_{spec}^{Core}	2.65 (kNs)/kg	2.53 (kNs)/kg	4.70%
I_{sp}^{Core}	3516 s	3370 s	4.3%

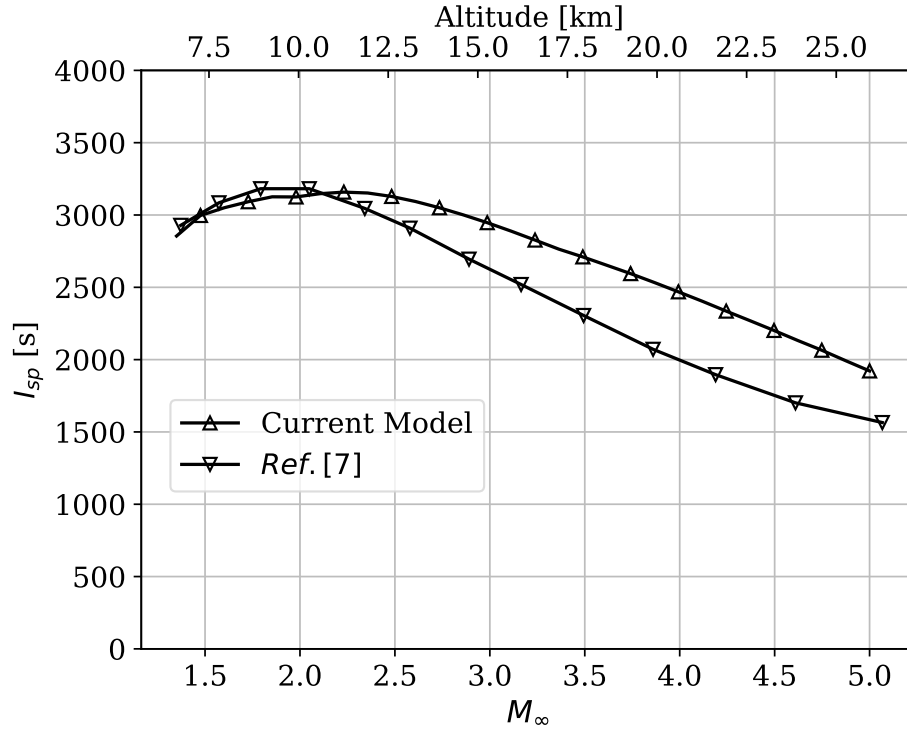


Fig. 6: Comparison of specific impulse as predicted by the current model against public data.

the model. However, the validity of the model, needs to be evaluated in the whole air-breathing ascent. In specific, it is of major importance that the overall performance trend of the system is properly captured from the developed model. This is critical to allow for confidence in any thermodynamic analysis or parametric study that is to be carried out at off-design conditions. This is depicted in Fig. 6, where the I_{sp} is shown as calculated by the current model against data found in the open literature [7], for the whole air-breathing ascent. This data is presented in a multitude of publications with the purpose to compare the efficiency of the engine with different propulsion concepts [7, 26, 27]. However, it is noted that there are inconsistencies with quantitative data as reported in other studies [24, 6]. Therefore, the comparison shown in Fig. 6 is performed to assess the credibility of the model in terms of capturing the performance trend and underlying physical mechanisms, but does not serve for a proper quantitative comparison. It is evident that the overall performance trend of the system, in terms of I_{sp} , is properly captured by the developed model. Initially there is an increase in the I_{sp} until the maximum is achieved. After this maxima, it continuously decreases until the design point is reached. Additionally, it is worth noting that the

maximum I_{sp} point predicted by the model, agrees reasonably well with the data.

A similar analysis can also be performed for the thrust of the system. The maximum gross thrust is properly captured by the model at conditions of Mach 4, as reported in previous studies [26]. Furthermore, the values of core specific thrust and specific impulse at this operating point are predicted by the model within less than 5% of the corresponding available data, as shown in Table 2. The gross thrust is used for these calculations, while no information is available for the flow rates of the bypass. The thrust components of the engine during the trajectory are shown in Fig. 7.

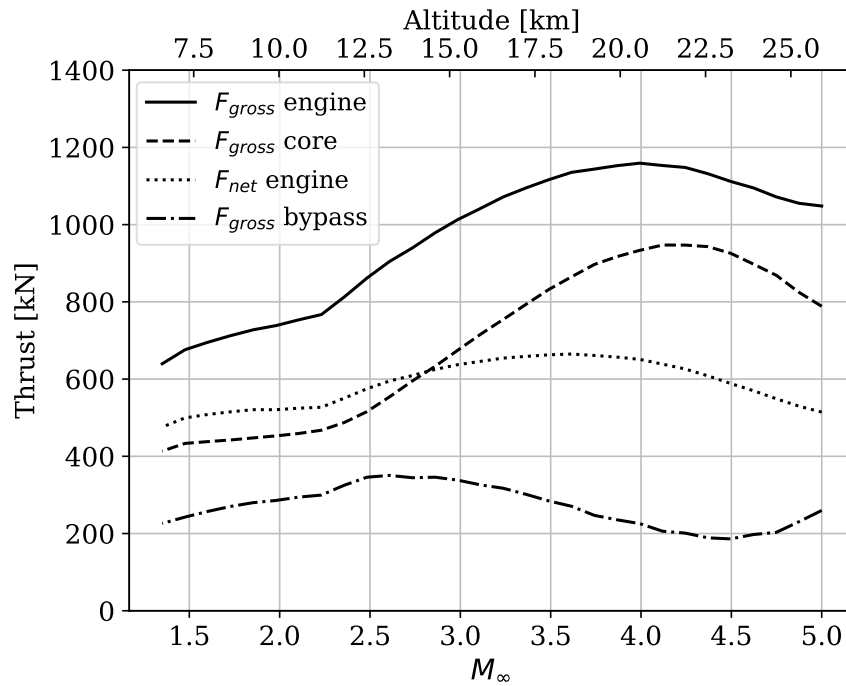


Fig. 7: Engine's thrust components

3.2 Cycle Synergies and Performance Characteristics

After the validation of the model, the cycle's operation can be described using the well-established H-S diagrams. Two different operating conditions are presented, corresponding to off-design and design operation, namely Mach 2 and 5. In the diagrams free stream conditions are adopted as a

reference point, thus enthalpy and entropy are equal to zero.

The H-S diagram of the open air cycle of the engine's core is depicted in Fig. 8. The conditions at the exit of the corresponding cycle component have been annotated in the graph to facilitate the comparison. During the transition from Mach 2 to Mach 5, the total enthalpy and entropy change is increased, as can be observed from the total area under the curves of each cycle. This corresponds to increased mass flow rates and heat loads generated at the design point. In specific, the entropy increase at the intake is larger due to increased shockwave intensity, subsequently enthalpy also increases due to the higher recovery temperature. The resulting larger temperature difference in the PC, results in larger entropy rise during the heat transfer process. This direct relation of heat load generation with the increased levels of entropy generation is the reason why the PC is typically designed at the end of the air-breathing mode (Mach 5) [6]. In the compression process of the AC and the combustion taking place in PB, minor differences are observed. When examining HX3, the enthalpy drop is more pronounced at Mach 2. The larger contributor to the enthalpy increase of the open air cycle is the combustion process of the CC. Finally, the expansion through the core nozzle is more efficient at Mach 5, in which the exhaust gases are expanded almost to ambient conditions.

The H-S diagram of the closed helium cycle is illustrated in Fig. 9. The total enthalpy exchange of the cycle is much larger at $M_\infty = 5$, due to the significantly increased thermal load of the PC. The enthalpy and entropy change of HX3 at $M_\infty = 5$, is a small part of the cycle, however at $M_\infty = 2$, it comprises a large part of the closed helium cycle. In specific, Fig. 9 shows that at off-design conditions of Mach 2, the entropy generation in the cycle is higher at HX3 compared to the PC. This indicates that different components of the cycle have to be designed and/or optimized at different operating conditions. Therefore, the use of a model which can properly represent the engine performance at both design and off-design conditions is necessary.

The actual flow conditions of the working mediums at each station of the engine are shown in Fig. 10a for the design point and Fig. 10b for conditions of $M_\infty=2$, where the temperature, mass flow rate and total pressure have been annotated. The numbers are rounded to the nearest integer. These are the cycle conditions to which the previously shown H-S diagrams correspond,

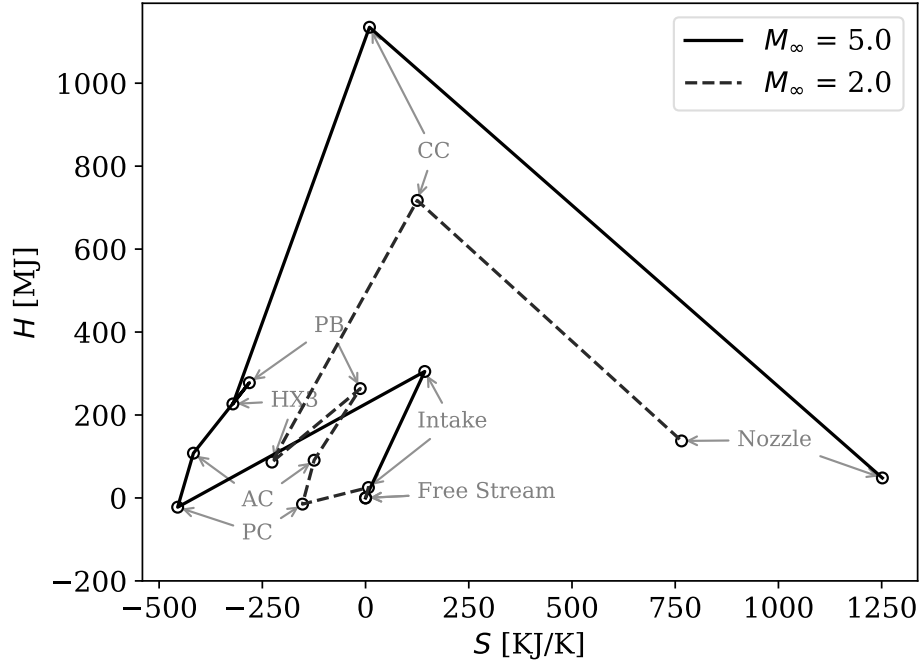


Fig. 8: H-S diagram of open air cycle of core

and give a quantitative view of the different operation of the engine's cycle at off-design state. The reduced intensity of the intake shock wave system can be observed in the lower temperature levels downstream of the intake at Mach 2 (Fig. 10b). Additionally, one can notice the notable difference in the operation of the system PC-HX3 at conditions of Mach 5 and 2. In specific, owing to the cycle's principles (i.e., constant HeT conditions), the operating temperature difference of HX3 is increased at $M_\infty=2$, owing to the reduced thermal load in the PC. In contrast, at the design point, the operating temperature difference of the PC is maximum, and therefore, the requirement of HX3 is minimized (Fig. 10a).

In the operation of the engine there is a balance between the heat loads generated by the underlying processes of the system and the control scheme which affects the mass flow rates of the working mediums. The calculation of the air flow rate through the core has been described in section 2, and so has the minimization of hydrogen flow through the control scheme. In Fig. 11 it is shown that air flow rate is a function of the total pressure recovery of the intake. Furthermore, the flow rate of hydrogen follows the trend of the heat load of HX4, as depicted in Fig. 12. As for the flow rate of helium, which is not part of the engine's control scheme, it follows the trend

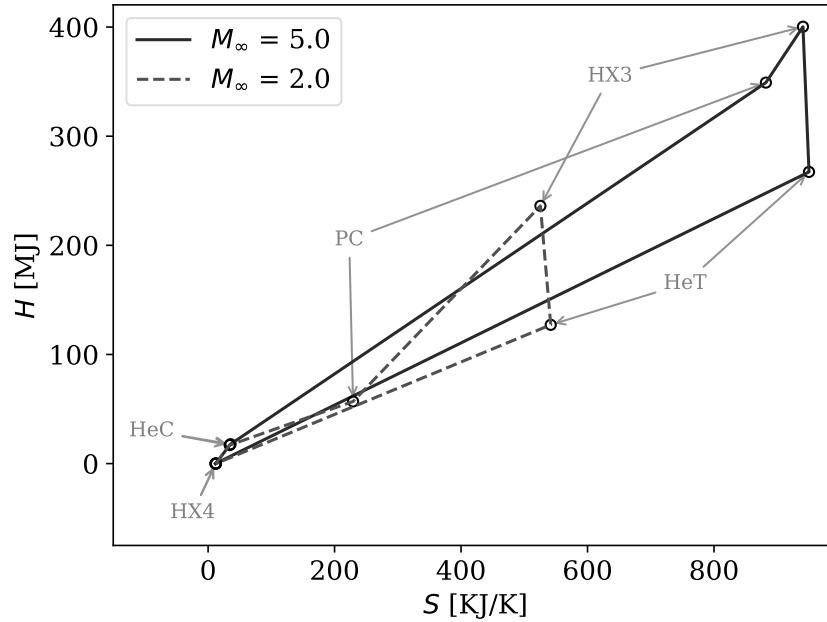


Fig. 9: H-S diagram of closed helium cycle

of the hydrogen flow rate. The changes in the trend of the thermal and work loads of the other components of the closed helium cycle do not have a pronounced effect on the flow rate. Carrying-on with the analysis of the thermal loads of Fig. 12, the maximum load of HX3 appears near Mach 2.5. Through HX3, helium has to be heated to 1080 K, subsequently the energy required is a function of the flow rate and the inlet temperature (i.e., the outlet temperature of the PC). Both parameters are affected by the control scheme of the engine, thus the maximum point of $\dot{Q}_{HX3}$ and its value, are closely related to the latter. Subsequently, the design of HX3 is coupled to the development of the control scheme of the engine. However, the maximum heat load dissipation requirements for HX3 are in the supersonic range below Mach 3 as it will be shown later on. Additionally, except from maintaining steady inlet conditions of the HeT, HX3 has the function to add sufficient energy to the closed helium cycle in order for HeT to extract the work needed for the compression of air through AC. However, there is a point during the ascent (~ 3.25 Mach) in which the energy added to the cycle through the PC is adequate to power the HeT. At the start of the ascent, the energy added by the HX3 is more than 60% of the work demanded by HeT, as shown

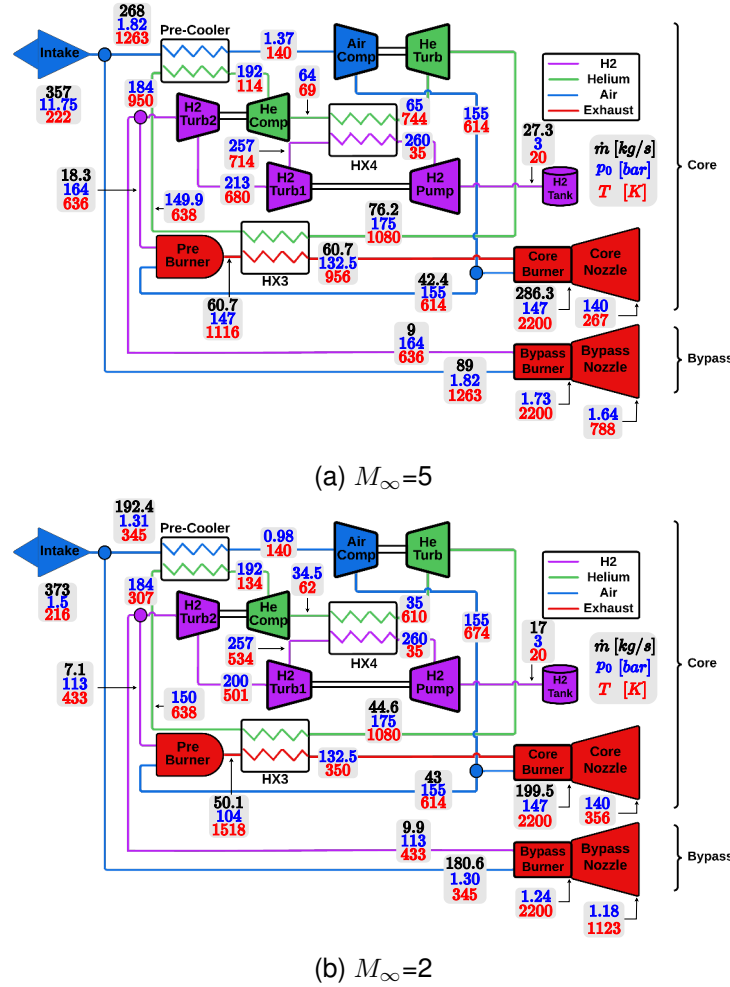


Fig. 10: Flow conditions of the cycle at (a) design and off-design (b) operation

in Fig. 12. The excess energy must be extracted through HX4 which leads to reduced efficiency. To compensate this, an optimization process could be applied on the helium flow rate and/or on the HeT inlet temperature. For the latter, which is a design-driven parameter and is set constant, the case of varying its value could be examined, for this phase of the ascent. Additionally, the maximum thermal loads for PC and HX4 appear at the design point of the engine. Finally, $\dot{Q}_{HX4}$ increases continuously, despite the different trend of $\dot{Q}_{HX3}$ at start of the ascent, highlighting the underlying complexity of the cycle synergies.

There is a strong interaction between the combustion process of the PB and the heat transfer process through HX3. This interaction affects the ratio of air diverted into the PB and the resulting

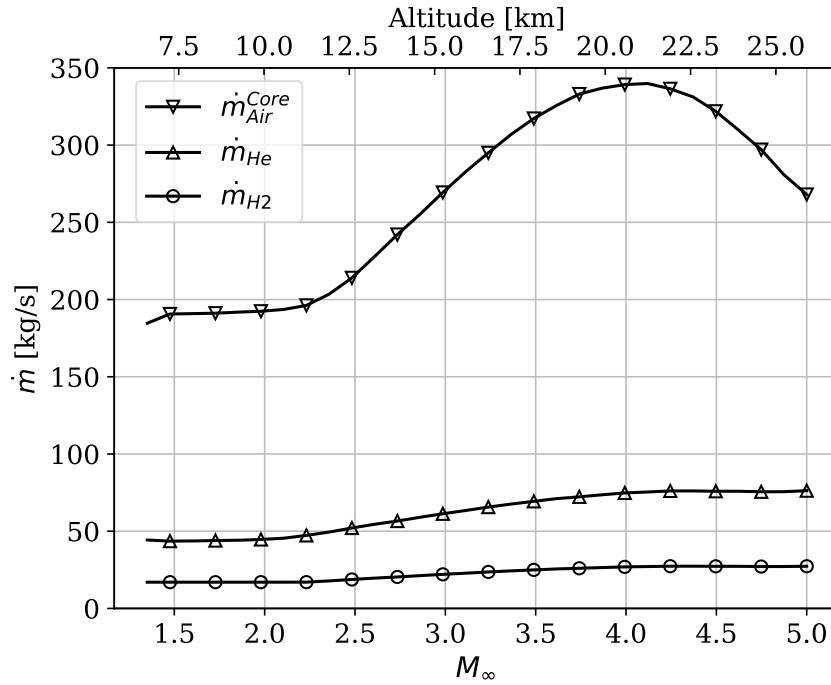


Fig. 11: Core mass flows

equivalence ratio. In the next subsection it will be demonstrated that the hydrogen flow rate through the core (Fig. 13, top), thus through PB, is imposed only by the operation of the CC. Consequently, the air that flows through the PB controls both the total flow rate and the equivalence ratio, ergo the PB combustion temperature. Additionally, the flow rate is constrained by the heat transfer of HX3. In Fig. 13 (bottom) it can be seen that the combustion is fuel-rich throughout the entire ascent for both PB and CC, as mentioned in section 2. The numerical results produced by the model for the behavior of the equivalence ratio of the two burners, shown in Fig. 13 (bottom) are validated against an analytical expression which is derived from the mass conservation of the 2nd subsystem, as shown in Eq. (19). Additionally, although the ratio of air diverted into the PB remains under 25% for the entire trajectory (see top of Fig. 13), it is not constant. Therefore, a constant value assumption which would simplify the model cannot be adopted.

$$\phi_{CC} = \frac{\phi_{PB} \cdot \dot{m}_{H_2}^{Core} \cdot (O/F)_S - 0.232 \cdot \dot{m}_{Air}^{AC}}{(\phi_{PB} - 1) \cdot (O/F)_S \cdot \dot{m}_{H_2}^{Core}} \quad (19)$$

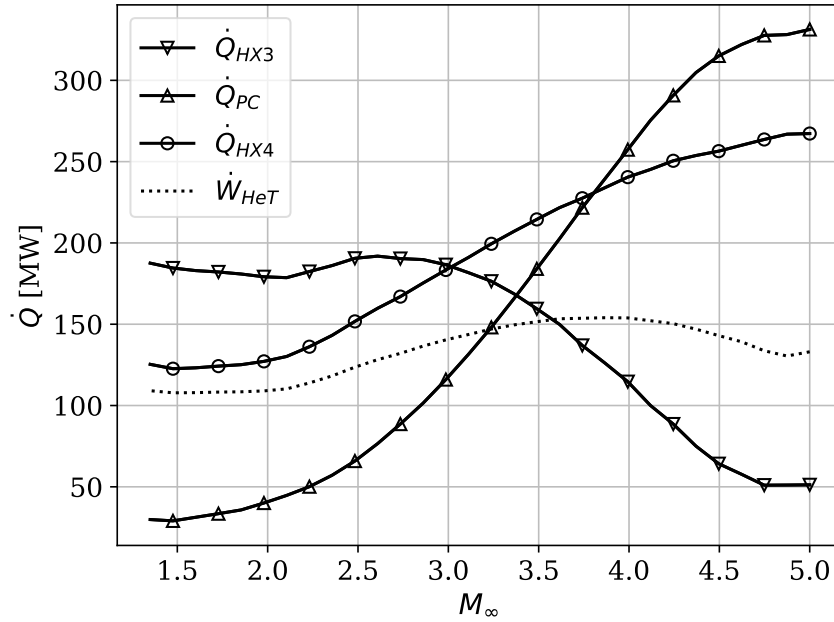


Fig. 12: Heat exchangers thermal loads

where $\phi = (O/F)_S / (O/F)$ is the equivalence ratio the combustors, and 0.232 the mass ratio of oxygen to air.

Furthermore, the effect of flow separation phenomena inside the exhaust nozzles on the overall performance are also investigated, through the implementation of the Summerfield criterion as described in section 2. In Fig. 14 the gross thrust of the engine is compared to the thrust that would be generated assuming ideal expansion to the ambient static pressure along the whole air-breathing ascent. The flow fully attaches to the nozzle, very close to the design point of the engine, approximately at $M_\infty = 4.9$. The maximum decrease of thrust is 10%, which decreases the I_{sp} by 13%, while the average reduction of gross thrust is 4.6% and 3.1% for the I_{sp} . This analysis indicates that the flow separation within the nozzles should be taken into account for a more accurate estimation of the performance metrics, especially at lower altitudes.

3.3 Core Exhaust System Parametric Analysis

To further analyze the synergetic characteristics of the components comprising the exhaust system of the core (Fig. 5b), a parametric analysis is carried-out. This is facilitated due to the

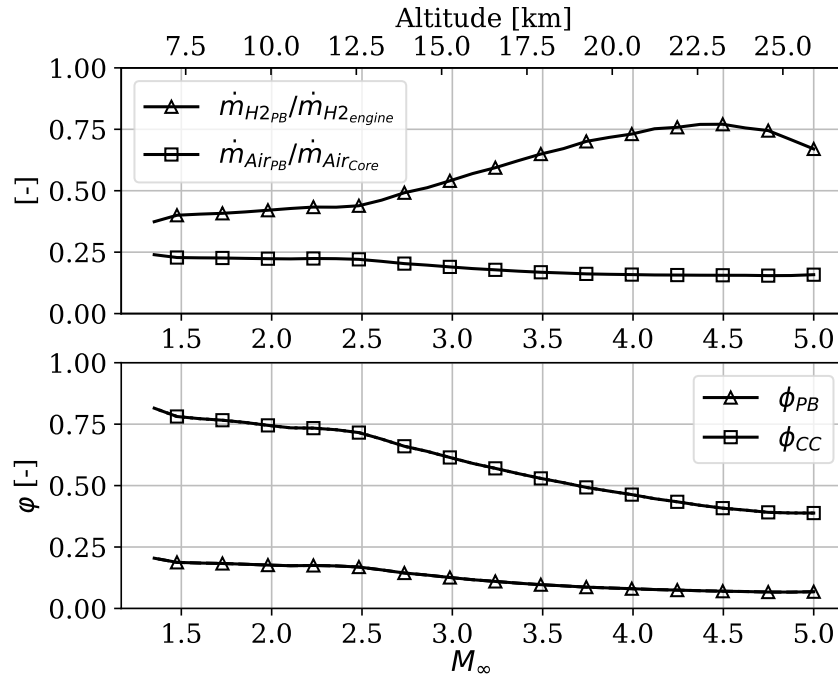


Fig. 13: Top: Fuel and air flow rate ratio diverted into the PB; Bottom: Equivalence ratio of PB and CC

employed numerical approach, which allows for the 2nd subsystem to be investigated separately at any point of the trajectory. The operating point selected is at Mach 4.5 and altitude of 22.6 km. In this analysis, initially the 1st subsystem is evaluated and then its parameters remain constant. These are $\dot{Q}_{HX3}$, $\dot{m}_{Air}^{Core}$ and $\dot{m}_{H2}^{Tot}$, $\dot{m}_{He}$. As described in section 2, the independent variables of the 2nd subsystem are HFR, ϕ_{PB} and their effect is examined in the range of $0.15 \leq \text{HFR} \leq 1$ and $0.1 \leq \phi_{PB} \leq 1$.

In Fig. 15a the variation of temperature of the main combustor, T_{exh}^{CC} , with the relative hydrogen mass flow through the core and the PB equivalence ratio is presented. It is clear, that from the independent variables of the 2nd subsystem, the temperature is invariant to the equivalence ratio of the PB and only a function of the quantity of hydrogen in the core. The maximum combustion temperature of approximately 2600 K is located near values of $\text{HFR} = 0.37$. Consequently, the selected operating temperature of 2200 K (annotated in the graph) can be achieved at two values of relative hydrogen core flow rate. By superimposing the two possible scenarios of combustion temperature (Fig. 15a) on the operating limits of the PB exhaust temperature, we can identify the

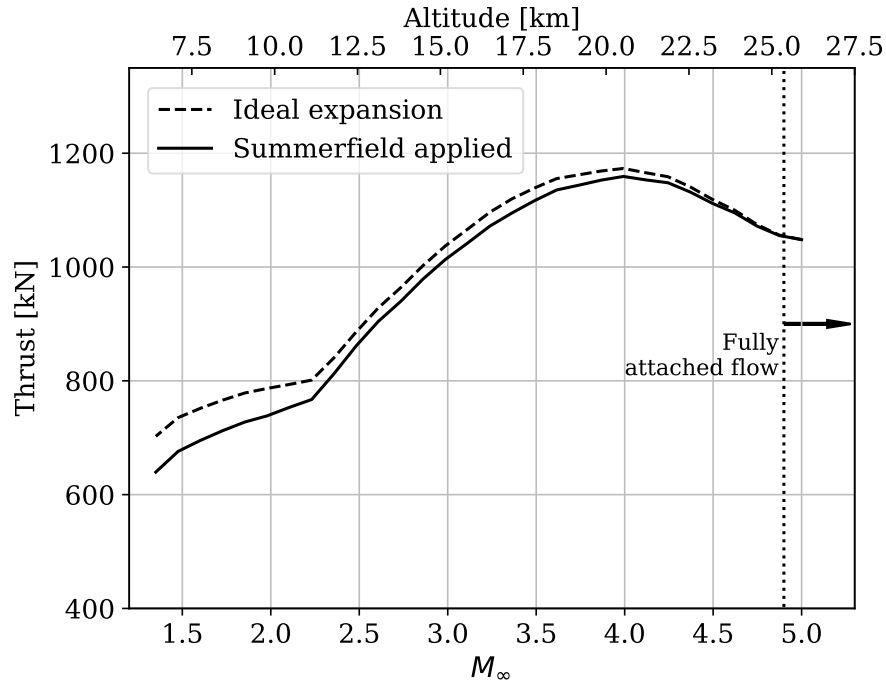


Fig. 14: Effect of flow separation on the thrust produced by the engine's nozzle

feasible operating conditions in terms of HFR and ϕ_{PB} , for a physically sound and numerically stable solution. This is depicted in Fig. 15b. The upper limit for the PB is imposed by the materials used in HX3 [25] and the lower limit by the lowest theoretical temperature allowed, which is the HeT inlet temperature [27]. The feasible solutions of the system are along the T_{exh}^{CC} lines and inside the upper and lower limits of the PB. It is revealed that the feasible solution for the lower value of the fuel ratio is narrow and could become non feasible in other points of the trajectory. In this case, for the combustion temperature to remain constant, a numerical spike would be created in the model. Therefore, this analysis defines the allowable region of operating conditions of the 2nd subsystem. Finally, it is observed that the temperature of the PB is affected both by the equivalence ratio and the quantity of fuel added to the system.

The effect of the independent variables on the gross thrust and specific impulse of the engine's core is also examined. The gross thrust generated by the core nozzle at any operating point, with constant inlet pressure, is affected by the total mass flow and inlet temperature. This is depicted in Fig. 15c, in which one can notice that the thrust is increasing for increasing values

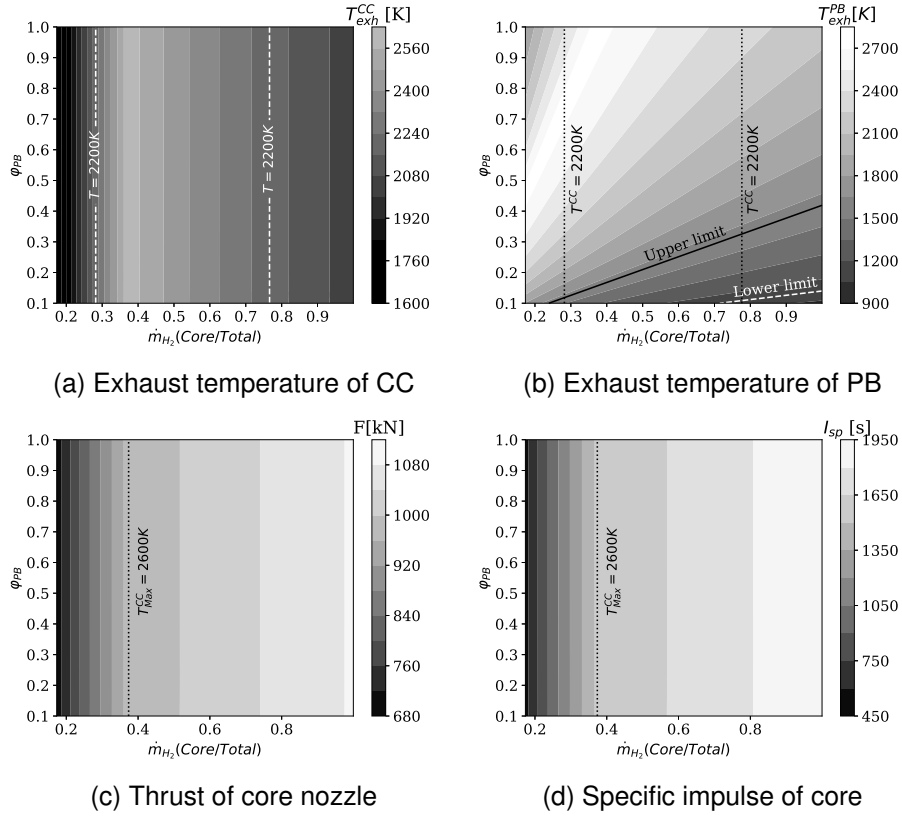


Fig. 15: Core exhaust system parametric analysis on its independent variables

of $\dot{m}_{H_2}^{Core}$. The steeper gradient at lower values of core hydrogen flow rates corresponds to the simultaneous increase of mass flow and the steeper increase of the core combustor temperature at that region (Fig. 15a). Once the maximum combustion temperature is reached (annotated in Fig. 15c), the effect of increasing hydrogen flow rate on the generated thrust dominates over that of the combustion temperature, which decreases. The same analysis and behavior applies also on the specific impulse, as seen in Fig. 15d.

3.4 Effect Of Core Mass Flow on the Engine Performance

The effect of the core air mass flow rate on the operation of the engine is also examined to evaluate its impact on the leading performance metrics and to assess the validity of alternative modeling considerations which are sometimes found in other studies [11, 41]. In this analysis the air flow rate is set constant and equal to that of the design point throughout the ascent. It is then

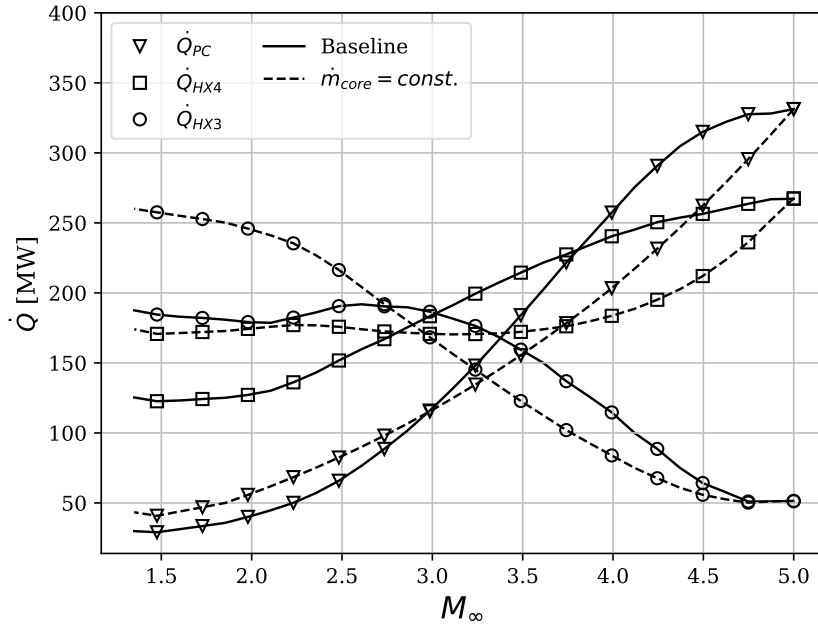


Fig. 16: Comparison of thermal loads distribution for the baseline model against the assumption of constant air mass flow through the core

compared to the original approach, which will be referred to as baseline. Consequently, at the design point, helium and hydrogen flow rates, as well as the thermal loads of the heat exchangers are identical to the baseline case. However, they are expected to be different both in quantity and trend for the off-design operating points. The comparison is depicted in Fig. 16, in terms of the thermal loads of the heat exchangers. By setting the air mass flow constant, the trend of $\dot{Q}_{PC}$ is identical to that of the recovery temperature of the intake (Fig. 4). It is observed that by assuming constant air mass flow rate the distribution of the thermal loads generated at HX3 and HX4 is notably different compared to the baseline model. In specific, the maximum heat load of HX3 appears at the start of the ascent ($M_\infty \sim 1.4$), while in the baseline case this is point is at $M_\infty \sim 2.6$. Additionally, for values of $M_\infty \leq 3$, the heat load at HX3 in the baseline model, is approximately constant at about 190 kW, while when assuming constant core air mass flow is continuously decreasing until the design point of Mach 5. A similar behavior can be observed for the heat loads of HX4, where in the baseline case the heat load is continuously increasing until Mach 5, while in the constant mass flow approach the heat load starts to increase at values above

Mach 3.5. However, the behavior of the heat load of the PC is similar to the baseline. The results shown in Fig. 16 demonstrate that although the assumption of constant air mass flow through the core offer advantages in terms of simplicity, it might lead to erroneous conclusions regarding the synergetic characteristics of the cycle, especially at the early ascent stage ($M_\infty \leq 3$).

The effect of the constant air mass flow through the core on the specific impulse of the combined system (core and bypass), as well as on the core alone with respect to the baseline model, is depicted in Fig. 17. The modified model slightly overestimates the specific impulse by up to 5% in the initial ascent phase ($M_\infty \leq 2.8$) compared to the baseline. If only the core of the engine is examined, the I_{sp} is higher up to values of $M_\infty = 4.2$. The overall trend of the engine's specific impulse is similar in the two approaches examined. However, it is stressed that the simplified approach of constant mass flow can be adequate to capture leading performance metrics such as the specific impulse (although slightly overestimated), but it might adversely affect the synergies of the components of the cycle, especially for Mach numbers below 3.

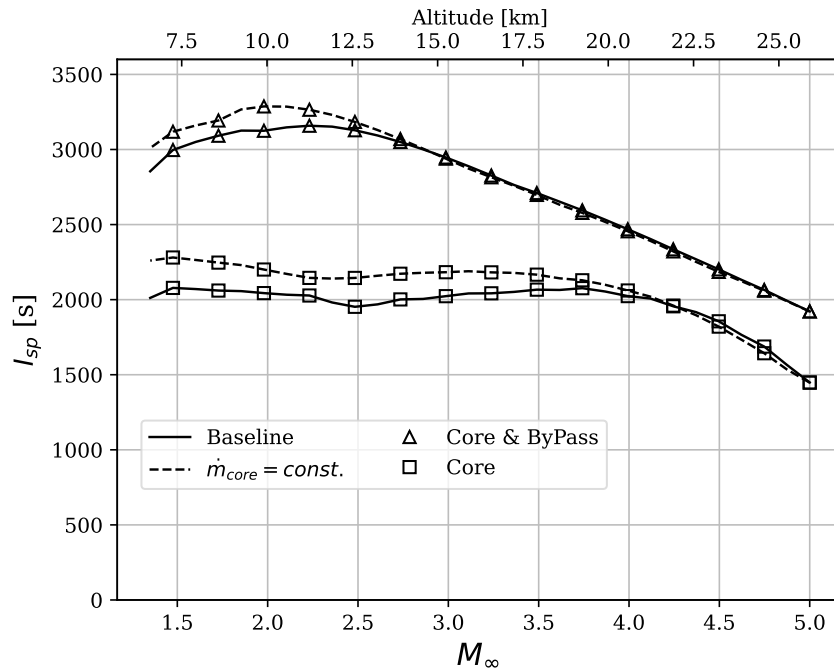


Fig. 17: Comparison of specific impulse for the baseline model against the assumption of constant air mass flow through the core

3.5 Sensitivity of Performance on PC and HeT Parameters

In the closed helium cycle the operation of the PC is limited by the maximum temperature limit imposed by the materials used. An improvement of the materials could raise this temperature, which could affect the overall performance. Furthermore, the constant inlet temperature of the HeT is a design-driven parameter. Therefore, it is of interest to identify the sensitivity of the cycle's performance on these variables. Specifically, the analysis is performed in the range of $900 \leq T_{out,max}^{PC,He} \leq 1000$ and $1000 \leq T_{in}^{HeT} \leq 1160$, for free stream conditions of Mach 5, since by definition the analysis is physically sound when the maximum heat load of the PC is reached (i.e., at the design point). The results are presented in Fig. 18, in which the relative change of I_{sp} and F_{spec} is shown with the respect to baseline case. The conditions corresponding to the baseline case are shown in the graph with black circles. It can be seen that the maximum PC outlet temperature of helium has a larger impact on the F_{spec} and I_{sp} compared to the inlet temperature of the HeT. In specific, raising the PC outlet temperature increases both performance parameters up to 3%. This behavior is in line with previous studies performed at the design point [19]. On the contrary, increasing T_{in}^{HeT} for fixed values of $T_{out,max}^{PC,He}$ slightly reduces the performance metrics compared to the baseline. This could be attributed to the excess levels of introduced energy into the helium cycle which need to be dissipated at HX4, leading to increase in fuel consumption (Fig. 12), considering that for a specific operating point the work of the HeT is coupled to the requirements of the AC.

3.6 Sensitivity of Performance on AC and CC Parameters

The AC of the investigated cycle can operate under large compression ratios (~150) due to the pre-cooling throughout the air-breathing ascent. This ratio is an important parameter for the design of the component, although its relative effect on the leading performance characteristics of the cycle is not apparent. Achieving this pressure ratio requires extending the state of the art technology and could introduce further complexity in the design process. Furthermore, in the open literature, there is a large variation for the values selected [12, 18, 27, 26]. In this section, its relative effect will be compared against another design-driven parameter of the core's open air cycle. The combustion temperature of the CC is selected, since its impact on the performance of

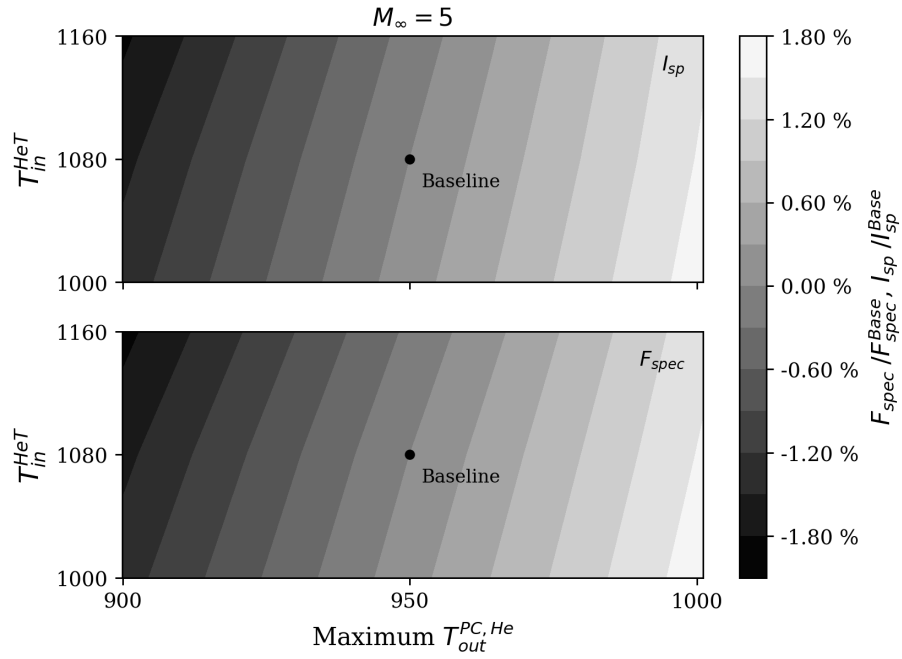


Fig. 18: Variation of specific impulse with the maximum PC outlet and turbine inlet temperature of helium

the engine has been demonstrated earlier in this study (Fig. 15). This analysis is carried out in the range $2150 \leq T_{exh}^{CC} \leq 2250$ and $130 \leq P_{0,out}^{AC} \leq 180$. The results show that despite the significantly larger range of the AC total pressure outlet ($\pm 16\%$) compared against the combustion temperature ($\pm 2.5\%$), both of their affect is less than 4% on the specific impulse and specific thrust (Fig. 19). In detail, at the design point (Mach 5), increasing the total pressure outlet of the AC increases both the specific thrust and the specific impulse of the engine. Furthermore, increasing the combustion temperature, increases the specific impulse of the engine but decreases the specific thrust. This behavior is consistent at the off-design point examined (Mach 2 in Fig. 19). Consequently, while the total outlet pressure of the AC improves both performance metrics and should be optimized from a thermodynamics perspective, the associated engineering challenges might not justify the resulting minor relative gains when compared against other parameters of the cycle, such as the combustion temperature of the CC.

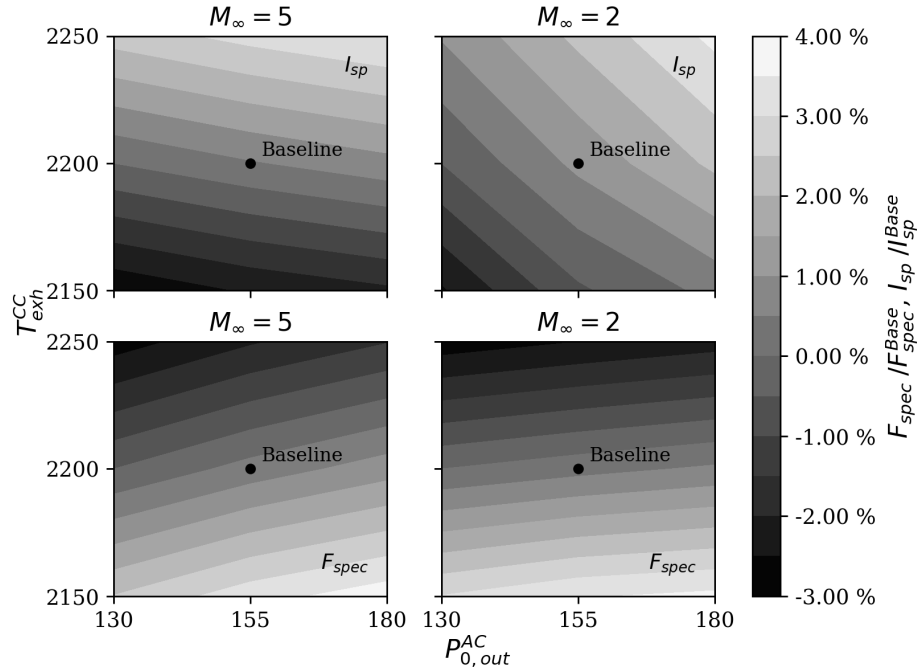


Fig. 19: Variation of specific impulse and net thrust with the total outlet pressure of the AC and combustion temperature of the CC

3.7 Impact of the Bypass Duct

To investigate the effects of the bypass duct on the performance characteristics of the engine, the current model is modified and the bypass duct is totally removed. In the modified model all hydrogen must flow through the core, thus the assumption of constant T_{exh}^{CC} cannot be kept (Fig. 15). The effect is examined based on two different scenarios. First the inlet conditions of the intake are matched throughout the mission profile (Fig. 2). This results in identical thermal loads of heat exchangers and total mass flow rates of the working mediums. Subsequently, the effects of the bypass duct can be isolated from the synergies of the helium with the hydrogen cycles. Specifically, only the 2nd subsystem is affected (Fig. 5b). In the second investigated scenario, the net thrust of the two models is matched, with the goal to compare the engines while performing the same mission, therefore this scenario is more relevant from an engineering perspective. It is noted that aerodynamic effects (e.g., parasitic drag, base drag, etc.) are not included in the models.

The results for the specific impulse are presented in Fig. 20. The total specific impulse of the engine (core and bypass) is significantly reduced when removing the bypass duct, in both cases.

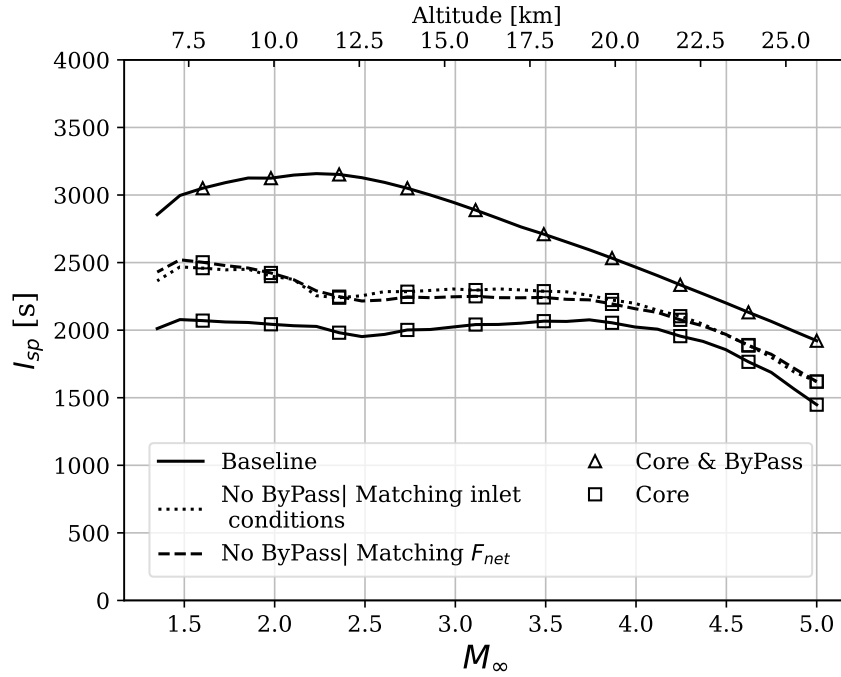


Fig. 20: Specific impulse comparison of model with and without the bypass duct

Furthermore, the trend is notably different, owing to the absence of the ramjet-like characteristics, and clearly follows the trend of the core. Consequently, for the proper estimation of the performance of the cycle, it is necessary for the model to include the bypass duct. The impact of the bypass on the net thrust of the system is also examined and presented in Fig. 21, only for the case of matching the inlet conditions, since the same analysis cannot be performed by definition in the second scenario, in which the two systems have identical net thrust. Initially, it is observed that the net thrust produced by the no-bypass model is larger for Mach 3.5 and above, while the maximum increase appears at Mach 4.25 and is equal approximately to 6%. However, this corresponds to a simultaneous increase of fuel flow rate of 21%, which results to a decrease of I_{sp} equal to 14.5%. Consequently, the no-bypass model can generate higher net thrust, for a phase of the trajectory although, with significantly decreased efficiency. This phase also coincides to the operating region of reduced efficiency of a ramjet cycle ($\sim M_{\infty} > 3$), which the bypass duct operates similarly. Furthermore, the trend of net thrust generation of the core is not affected by the removal of the bypass duct. It is reminded that the combustion temperature, T_{exh}^{CC} , of the baseline model is kept constant

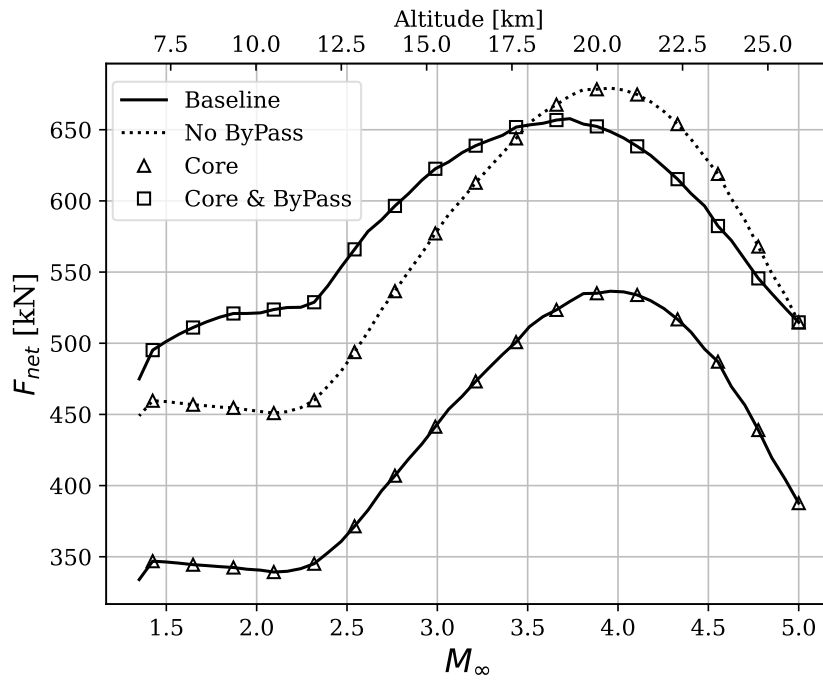


Fig. 21: Net thrust comparison with engine without the bypass duct when matching the inlet conditions

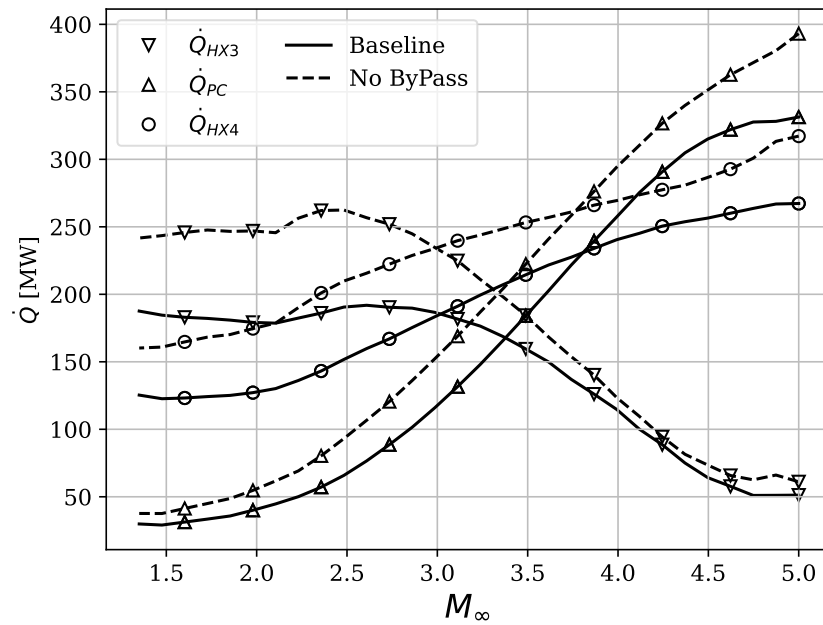


Fig. 22: Heat exchanger thermal load comparison when matching the net thrust

throughout the trajectory, while in the case without the bypass duct the combustion temperature is varying, as a result of the entirety of the hydrogen flow passing through the core. Regardless, in the scenario of matching the inlet conditions, the two systems (with and without the bypass) feature identical thermal loads at the heat exchangers and total mass flow rates of the working mediums. The identical trend in the net thrust of the no-bypass system and that of the core of the baseline model (Fig. 21) demonstrates that the net thrust generation process of this cycle's core, is driven by the introduced heat exchanger synergies. Additionally, in the case of matching the net thrust, the heat exchangers thermal loads have similar trends but are notably higher compared to the baseline as shown in Fig. 22. The increased thermal loads would mean that for the same mission the engine without the bypass would have to carry larger and heavier heat exchangers or have increased mass flow rates, both leading to increased weight. Finally, the results of Fig. 22 can also be used to confirm the conclusions of the analyses presented previous subsections. In specific, the maximum point of the $\dot{Q}_{HX3}$, coincides with the local minimum of the core specific impulse approximately at $M_\infty = 2.5$, suggesting that the improvement of HX3 characteristics could result in increased specific impulse in that region of the trajectory ($1.35 < M_\infty < 3$), 3.2. Furthermore, the increased I_{sp} and F_{net} , due to the higher fuel flow rate through the core, agrees with the results of the analysis on the core exhaust system (Fig. 15), and demonstrates that the conclusions apply to the entire ascent.

Table 3: Comparison of model without bypass duct

	$(\Delta I_{sp})_{max}$	$(\Delta \dot{m}_{H_2})_{max}$
Matching inlet conditions	-33%	+18.8%
Matching net thrust	-32%	+48%

The results are quantified and presented in Table 3. For the case of matching inlet conditions, the maximum reduction of I_{sp} is 33%, while the maximum increase of $\dot{m}_{H_2}$ is 18.8%. In the case of matching the net thrust, the maximum reduction of I_{sp} is 32%, while the maximum increase of $\dot{m}_{H_2}$ is 48%. As mentioned previously, from a pure engineering perspective, it is the analysis of identical

net thrust that is of primary interest. In this case, removal of the bypass would increase by a factor of about 1.5 the fuel flow rate and decrease the efficiency by approximately 32%. Consequently, the bypass duct has a significant impact on the efficiency of the engine, from a thermodynamics point of view.

4 CONCLUSIONS

A model based on an air-breathing pre-cooled cycle was developed, by applying one-dimensional thermodynamic relations within a component level approach. A control scheme is employed and the resulting numerical system was divided in two subsystems. These are solved sequentially using a stochastic algorithm. The model is validated using open literature data, and the maximum error of the model is found to be $< 5\%$ for the F_{spec} and I_{sp} . Furthermore, the trend of I_{sp} is predicted correctly, as well as the operating points of maximum I_{sp} and F_{gross} . After the successful validation, the synergies of the cycle were examined and several parametric analyses were carried-out. Finally, a model without the bypass duct was created and compared to the original. The most significant conclusions are as follows:

- The effect of nozzle flow separation on the leading performance metrics was quantified. The net thrust decreased up to 10% which decreases the specific impulse by 13%, while the flow fully attaches to the nozzles approximately at Mach 4.9. Therefore, flow separation must be taken into account for a more accurate estimation of the performance metrics, especially at lower altitudes.
- The simplified approach of constant core air flow is sufficient to capture the trend of the specific impulse, but it may lead to erroneous conclusions regarding the synergetic characteristics of the cycle, such as the heat load distribution of heat exchangers, especially at the early ascent stage ($M_{\infty} \leq 3$).
- The combustion temperature of the CC, is a function only of the ratio of the fuel flow rate diverted into the core. Higher values of this parameter increase the value of both the net thrust and the specific impulse of the core, while it has no impact on their observed trend throughout the trajectory.

- Removing the bypass duct while matching the inlet conditions of the core (i.e., identical thermal loads and mass flows), the trend of the performance metrics of the core is not affected. This suggests that the net thrust generation and the specific impulse behavior of the core are driven by the synergies of the cycle's heat exchangers.
- The design of the HX3 is dominated by the inlet conditions of the core at the early phase of the ascent ($M_\infty < 3$). At this phase, its process comprises a large part of total enthalpy and entropy exchange of the helium cycle, while in the air cycle it is more pronounced at Mach 2 compared to Mach 5. In addition, the maximum thermal load of the HX3 coincides with a local minima of the specific impulse of the core, which appears for $M_\infty < 3$. Consequently, its design and/or optimization should be focused on this part of the ascent, contrary to the PC, which is at Mach 5.
- The effect of the bypass was quantified for off-design conditions throughout the air-breathing ascent. This impact has not been reported in previous studies. When matching the net thrust of the original model with the no-bypass model (i.e., the same mission is performed by the models), the specific impulse would decrease up to 33% and the fuel mass flow rate would increase up to 48%. The trend of the specific impulse of model without the bypass duct is notably different, which follows the trend of the core of the original model. Consequently, for the correct estimation of the I_{sp} for both design and off-design operating points of the engine, it is imperative for the bypass duct to be included in the model.

REFERENCES

- [1] Ingenito, A., 2021, *Subsonic Combustion Ramjet Design* Springer.
- [2] Curran, E. T., and Murthy, S., 2001, *Scramjet propulsion*, Vol. 189 AIAA.
- [3] Snyder, L., Escher, D., DeFrancesco, R., Gutierrez, J., and Buckwalter, D., 2004, "Turbine based combination cycle (tbcc) propulsion subsystem integration," In 40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, p. 3649.
- [4] Oike, M., Kamijo, K., Tanaka, D., Yamada, H., and Yoshida, H., 1999, "Lace for rocket-based combined-cycle," In 37th Aerospace Sciences Meeting and Exhibit, p. 91.

- [5] Sato T., T. N., Naruo Y., Omi J., T. J., and T., N., 2000, "Development study on atrex engine," *Acta Astronautica*, **47**(11), pp. 799–808.
- [6] H., W., A., B., and M., H., 2007, "The sensitivity of precooled air-breathing engine performance to heat exchanger design parameters," *JBIS - Journal of the British Interplanetary Society*, **60**, 05, pp. 188–196.
- [7] Varvill, R., and Bond, A., 2003, "A comparison of propulsion concepts for ssto reusable launchers," *Journal-British Interplanetary Society*, **56**(3/4), pp. 108–117.
- [8] Jivraj, F., Bond, A., Varvill, R., and Paniagua, G., 2007, "The scimitar precooled mach 5 engine,".
- [9] Wang, C., Dang, C., Ha, C., Qin, J., and Huang, H., 2022, "Thermodynamic analysis for a novel steam rankine cycle based indirect chemical precooled engine used for supersonic flight," *Fuel*, **321**, p. 123956.
- [10] Yao, Z., Guo, Y., Liu, M., and Zhou, S., 2022, "Performance study of a pre-cooled turbo-rocket combined engine under a wide mach number of 0~ 5," *Case Studies in Thermal Engineering*, **38**, p. 102307.
- [11] Wei, X., Jin, F., Ji, H., and Jin, Y., 2022, "Thermodynamic analysis of key parameters on the performance of air breathing pre-cooled engine," *Applied Thermal Engineering*, **201**, p. 117733.
- [12] Zhang, J., Wang, Z., and Li, Q., 2017, "Thermodynamic efficiency analysis and cycle optimization of deeply precooled combined cycle engine in the air-breathing mode," *Acta Astronautica*, **138**, pp. 394–406.
- [13] X., Y., C., W., and D., Y., 2020, "Minimization of entropy generation of a closed brayton cycle based precooling-compression system for advanced hypersonic airbreathing engine," *Energy Conversion and Management*, **209**, p. 112548.
- [14] Yu, X., Wang, C., and Yu, D., 2019, "Configuration optimization of the tandem cooling-compression system for a novel precooled hypersonic airbreathing engine," *Energy Conversion and Management*, **197**, p. 111827.
- [15] Yu, X., Wang, C., and Yu, D., 2020, "Thermodynamic design and optimization of the multi-

- branch closed brayton cycle based precooling-compression system for a novel hypersonic aeroengine,” *Energy Conversion and Management*, **205**, p. 112412.
- [16] Fernández-Villacé, V., and Paniagua, G., 2010, “Simulation of a combined cycle for high speed propulsion,” In 48th AIAA Aerospace Sciences Meeting Including the New Horizons Forum and Aerospace Exposition, p. 1125.
- [17] Z., Z., W., Y., D., P., Z., L. Y. S. Y. W., and L., J., 2022, “A novel simplified precooled air-breathing engine cycle: Thermodynamic performance and control law,” *Energy Conversion and Management*, **258**, p. 115472.
- [18] Zhang, D., Chen, C., and Yu, X., 2023, “Control law synthesizing for an innovative indirect precooled airbreathing engine under off-design operation conditions,” *Energy*, **263**, p. 126110.
- [19] Tsentis, S., Gkoutzamanis, V., Gaitanis, A., and Kalfas, A., 2021, “Multi-platform app-embedded model for hybrid air-breathing rocket-cycle engine in hypersonic atmospheric ascent,” *The Aeronautical Journal*, **125**(1291), pp. 1631–1653.
- [20] Wang, C., Eri, Q., Wang, Y., Kong, B., and Ding, W., 2023, “Novel three-dimensional simplified numerical simulation method of a compact precooler for hypersonic precooled air-breathing engine,” *Applied Thermal Engineering*, **230**, p. 120827.
- [21] Wang, C., Eri, Q., Wang, Y., and Ding, W., 2023, “Flow and heat transfer characteristics of intake-precooler system for hypersonic precooled aero-engine,” *Applied Thermal Engineering*, **229**, p. 120596.
- [22] Wang, C., Eri, Q., Wang, Y., Kong, B., and Ding, W., 2023, “Multi-objective aerodynamic optimization of an axisymmetric variable-geometry inlet with a mach 5 design point,” *Aerospace Science and Technology*, **136**, p. 108189.
- [23] Hemsell, M., Longstaff, R., and Bond, A., 2010, “Skylon users’ manual,” *Reaction Engines Limited*, **1**.
- [24] Verstraete, D., and Hendrick, P., 2011, “Hydrogen fueled precooled airbreathing engines for hypersonic aircraft and spaceplanes,” *ISABE-1329*.
- [25] Varvill, R., 2010, “Heat exchanger development at reaction engines ltd.,” *Acta Astronautica*, **66**(9-10), pp. 1468–1474.

- [26] R., L., and A., B., 2011, “The skylon project,” In 17th AIAA International Space Planes and Hypersonic Systems and Technologies Conference, p. 2244.
- [27] Agency, E. S., 2011, Skylon assessment report.
- [28] Sutton, G. P., and Biblarz, O., 2016, *Rocket propulsion elements* John Wiley & Sons.
- [29] Anderson, J. D., 1990, *Modern compressible flow: with historical perspective*, Vol. 12 McGraw-Hill New York.
- [30] Chase, M. W., and (US), N. I. S. O., 1998, *NIST-JANAF thermochemical tables*, Vol. 9 American Chemical Society Washington, DC.
- [31] Bell I., Wronski J., Q. S., and V., L., 2014, “Pure and pseudo-pure fluid thermophysical property evaluation and the open-source thermophysical property library coolprop,” *Industrial & Engineering Chemistry Research*, **53**(6), pp. 2498–2508.
- [32] Leachman, J. W., Jacobsen, R. T., Penoncello, S., and Lemmon, E. W., 2009, “Fundamental equations of state for parahydrogen, normal hydrogen, and orthohydrogen,” *Journal of Physical and Chemical Reference Data*, **38**(3), pp. 721–748.
- [33] Wilson, D. G., 1984, “Design of high-efficiency turbomachinery and gas turbines,”.
- [34] H., W., S., F., and A., B., 2009, “Heat exchanger design in combined cycle engines,” *Journal of the British Interplanetary Society*, **54**, 04, pp. 122–130.
- [35] Kamijo, K., Sogame, E., and Okayasu, A., 1982, “Development of liquid oxygen and hydrogen turbopumps for the le-5 rocket engine,” *Journal of Spacecraft and Rockets*, **19**(3), pp. 226–231.
- [36] Douglass, H., 1973, “Liquid rocket engine centrifugal flow turbopumps,” *NASA SP-8109*.
- [37] Summerfield, M., Foster, C. R., and Swan, W. C., 1954, Flow separation in overexpanded supersonic exhaust nozzles.
- [38] Zucrow, M. J., and Hoffman, J. D., 1977, “Gas dynamics. volume 2-multidimensional flow,” *New York*.
- [39] Storn, R., and Price, K., 1997, “Differential evolution—a simple and efficient heuristic for global optimization over continuous spaces,” *Journal of global optimization*, **11**, pp. 341–359.
- [40] Virtanen, P., Gommers, R., Oliphant, T., and et al., 2020, “SciPy 1.0: Fundamental Algorithms

for Scientific Computing in Python,” *Nature Methods*, **17**, pp. 261–272.

- [41] Grimaldi, G. “Development of a conceptual design tool to predict performance of hydrogen fuelled precooled air-breathing rocket engines,” In 33rd Congress of the International Council of the Aeronautical Sciences, ICAS 2022.

LIST OF FIGURES

1	Air-breathing Pre-cooled Cycle	7
2	Air-breathing mode trajectory [23]	8
3	Schematic of employed intake-shock topology	10
4	Intake Recovery Conditions	11
5	Numerical subsystems employed for the solution the model	17
6	Comparison of specific impulse as predicted by the current model against public data.	19
7	Engine's thrust components	20
8	H-S diagram of open air cycle of core	22
9	H-S diagram of closed helium cycle	23
10	Flow conditions of the cycle at (a) design and off-design (b) operation	24
11	Core mass flows	25
12	Heat exchangers thermal loads	26
13	Top: Fuel and air flow rate ratio diverted into the PB; Bottom: Equivalence ratio of PB and CC	27
14	Effect of flow separation on the thrust produced by the engine's nozzle	28
15	Core exhaust system parametric analysis on its independent variables	29
16	Comparison of thermal loads distribution for the baseline model against the assumption of constant air mass flow through the core	30
17	Comparison of specific impulse for the baseline model against the assumption of constant air mass flow through the core	31
18	Variation of specific impulse with the maximum PC outlet and turbine inlet temperature of helium	33
19	Variation of specific impulse and net thrust with the total outlet pressure of the AC and combustion temperature of the CC	34
20	Specific impulse comparison of model with and without the bypass duct	35
21	Net thrust comparison with engine without the bypass duct when matching the inlet conditions	36

22	Heat exchanger thermal load comparison when matching the net thrust	36
----	---	----

LIST OF TABLES

1	Specific impulse comparison with open literature data	18
2	Core performance comparison with open literature data	18
3	Comparison of model without bypass duct	37



DSPACE

<https://dspace.org/>

Synergetic and performance characteristics of a high-speed pre-cooled propulsion concept

Chatzistefanou, Athanasios; Tsentis, Spyros; Kalfas, Anestis I.

2025-02-01

ASME

<https://doi.org/10.1115/1.4066352>

Chatzistefanou A, Tsentis S, Kalfas AI. (2024) Synergetic and performance characteristics of a high-speed pre-cooled propulsion concept. Journal of Engineering for Gas Turbines and Power, Volume 147, Issue 2, February 2025, Article number 021016