DOI: 10.13009/EUCASS2022-6164

Multidisciplinary design analysis and optimization of launch vehicles including environmental impact

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Abstract

The space economy dynamics given by the New Space ecosystem are changing at a fast rate which currently is not left unnoticed. Launch activities are thriving and the need to assess environmental impacts of an exponential increase on the amount of launches per year has emerged. In particular, the future number of launches per year raises concerns about launching systems' emissions and their impact on the environment. Different types of environmental impacts of launch vehicle design can be considered. This may include impacts on human health (toxicity), climate change (greenhouse gases effects, ozone depletion), resources depletion, *etc.* These effects arise throughout the entire launchers' life-cycle, including their design, manufacturing and transportation, operational and disposal phases. The present study is focused on the operational phase of a launcher and focuses particularly on its emissions and consequent radiative forcing (RF). Notably, the paper is focused on the effect of launchers' flight emissions which are significant in the stratosphere, such as: water vapour, carbon dioxide and carbon soot. Previous works have already shown how those chemical species damage the Earth's thermal balance, especially in an area in the stratosphere where they accumulate. Launcher vehicles' emissions result in a set of large environmental phenomena and their relation to climate change are scientific fields scarcely investigated. Simultaneously, launchers' emissions are a complex outcome that depends not only on the chemical dynamics during the combustion, but is also strongly influenced by the design and the mission analysis. This paper is focused on the inclusion of environmental impact analysis within the launch vehicle design process. Indeed, several design variables such as the total amount of propellant, the type of propellant, its mixture ratio and the guidance influence directly both performances and the stratospheric impact. The radiative forcing is the physical parameter chosen to assess the impact on the thermal balance. This work proposes to include environmental impact as a new discipline in the multidisciplinary design process coupling legacy disciplines such as propulsion, trajectory, aerodynamics and structure. The proposed multidisciplinary design analysis enables to provide decision-making regarding several criteria such as launch vehicle performance (*e.g.*, total mass of vehicle) and environmental impact (*e.g.*, RF). It is included an exploration of different design options in a tentative of consolidating the compromise between more sustainable and reliable types of propellants for reusable launch vehicles, propelled on LOX/LH2, LOX/CH4 and LOX/RP1. The optimizations are analyzed with the environmental footprint of the mission, in particular the total RF produced, which is expected to depend in particular on the mixture ratio.

1 Introduction

In the recent years, the space industry has increasingly attracted the attention of private investors. Once private operators realized how to use space services and applications to bring value to their own value chains, new PPPs (Public-Private Partnerships) appeared, as well as the provision of private investment and venture capital to space startups. Space R&D is on the rise and, in particular, has become intertwined with innovative Information Technology domains such as Artificial Intelligence and Machine Learning. All sectors of the space economy are under pressure, seeking higher performance and disruptive solutions. Not only governments, but also companies and institutions have started to promote the commercialization of the space sector. As a result, reducing production and operating costs has never been more important. The space services with the highest demand, embedded in the Telecommunications and Earth Observation markets, include secure data transmission, real-time tracking and the provision of big data. Market studies show that an impressive increase in production and operational activities is expected to meet the needs of organizations, business operators and private end-users in these markets. In particular, mega constellations are in the spotlight. Ultimately, after commercial satellites, launch systems is becoming one of the largest space production sectors, focusing efforts on cost reduction, as the number of launches per year is expected to increase exponentially over the next decade.

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Associated with the paradigm shift within the launcher industry, environmental concerns arise from a potential increase in the total number of launches per year. Indeed, the interest in the sustainability of space activities, including the production and operation of launchers, is growing strongly: clean space and green launchers are the focus of both space launcher industries and agencies. It is clear that efforts are needed to ensure the sustainability of access to space. Today, there are no explicit environmental regulations covering these topics. At the same time, as changes in the environmental policies of space systems are expected in the near future, the development of environmental modeling methodologies that can be used as decision-making tools in the early stages of launcher design becomes crucial. In fact, the objective of this paper is to introduce a methodology to include the environmental impact of a launcher as a constraint in the preliminary stage of launcher design.

The environmental impact of a product, and in particular of a launch system in this case, must be considered as a set of impacts of different types. Among the most important impacts are those on human health (*e.g.*, soil and air toxicity), climate change (*e.g.*, greenhouse gas effects, ozone depletion), and resource depletion (*e.g.*, fossil resource depletion). These effects occur during the entire life cycle of launchers, including the design, production, transport, operation and disposal phases.

Only a limited number of studies on the environmental impact of launchers are available in the literature. Of particular interest is the paper by Maury *et al.* [18] on the life cycle assessment (LCA) of space systems. LCA is a well-defined standardized procedure (ISO 14040/14044) to assess the environmental impact of a product throughout its life cycle [15], *i.e.* from the beginning of its design to the end of its life cycle (disposal or recycling). For a few years now, ESA and national agencies have been studying LCA for space and launchers, a very new field of investigation [26]. For example, LCA was applied by Stergiou *et al.* [25] to compare the impact of using carbon-fibre reinforced plastics or stainless steel for the VEGA launcher. Among the various environmental impacts of launchers, several studies have been devoted in particular to the pollution associated with combustion product emissions and their interaction with the atmosphere, particularly with regard to solid propulsion [19]. With regard to climate change, and in particular the effects of greenhouse gases, the work of Ross and Sheaffard [21], and DeSain and Brady [9] on direct radiative forcing (RF), comparing and estimating the impact of the operational phase of different launchers, is particularly interesting. It is worth mentioning that, regarding RF estimates, further progress has recently been made in relation to aircraft systems, summarized in the review by Lee *et al.* [17]. Furthermore, as a launcher is approximately 90 % propellant, it is expected that the propellant, including all phases of its life cycle (production, handling and operation), plays an important role in the overall impact of the launcher. This is highlighted for example in [20], where a comparison is made between the use of green hydrogen, methane and bio-methane considering the entire production chain. Launcher activities also have a great impact on the ecology of the region surrounding the spaceport, as investigated for example by Koroleva *et al.* [16] in their study of Russian launchers. Overall, apart from system studies, there is a lack of studies and models able to characterize the impact of specific launchers. For example, the assessment of exhaust emissions from launch systems or even the short-/long-term effects on a launch site and local/global atmospheric perturbations after a launch lack scientific research. However, as models are developed and knowledge about the LCA of launchers increases, there is a need to develop methodologies that aim to use this information for the design of more sustainable launch systems.

In the present paper, a methodology is proposed in order to include the environmental impact associated to a launcher into the design process of a launcher. More precisely, the impact during the launch due to the rocket engine emissions of chemical species into the atmosphere is considered in the design process. The ultimate objective should be to take into account all the life cycle of a launcher, that is considering all the impacts through an LCA. As a starting point, the impact on climate change is only addressed, through direct radiative forcing, of the operation phase of an expendable launcher. Particularly, this work focuses on the total Earth's change in the RF balance as a direct consequence of launchers' emissions during flight phase. The exhaust products studied include among others water, carbon dioxide, alumina particles and carbon soot, chemicals which were shown in previous studies [9] to have the biggest impact in the stratosphere. The consideration of other impacts will be the objective of future works. This type of approach is under development for aeronautics applications [14] but, to the authors knowledge, there are no published available papers showing the application to launchers systems. Specifically, in the present work the environmental impact is introduced as a new discipline in a MultiDisciplinary Analysis and Optimisation (MDAO) tool for the design of a launcher. MDAO approaches have being successfully applied in the last years for the description of different space systems including launchers [1, 3, 4]. MDAO allows to carry out an optimization of a complex system, considering in parallel the different disciplines that are needed in order to design the system. In the present work, the FELIN code [5], based on the opensource framework OpenMDAO [11], will be used. In the design process, four disciplines are used in order to describe the launcher: propulsion, structures, aerodynamics and trajectory. The launcher is optimized with respect to a certain number of design variables (including pressure chamber, oxydizer to fuel ratio, guidance laws, *etc.*) in order to minimize the Gross Lift-Off Weight (GLOW). A new discipline is added in order to evaluate the direct consequence of launchers' emissions during the flight phase in terms of RF balance, allowing to include a constraint in terms of RF balance in the optimization process.

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In the following, the proposed approach to account for environmental impact in the MDO of launch vehicles is developed. In particular, attention is given to the RF estimation of launch vehicles during their flight phase (Section 2.1) and the estimation of rocket engines emissions (Section 2.1.1). Then, the steps to integrate the RF as a discipline in the MDO process are detailed (Section 2.2). Finally, the developed tool is used in order to investigate and compare different design options for launch vehicles propelled with different types of propellants (Section 3). The results analysis includes the environmental footprint of the mission which is translated by the total RF produced and its influence on the design parameters.

2 Methodology to account for the environmental impact resulting from the launch vehicle flight phase in the design process

As discussed in the introduction, the environmental impact of launch vehicles may include broad aspects such as climate impact, biodiversity, pollution, resource depletion, *etc.* Including all these impacts in the early design phases of the launch vehicle design to carry out multi-criterion design is a challenging task as it is necessary to first implement physical models for all these phenomena and then to integrate them into a design process. In order to lay the foundation of the introduction of environmental impact in multidisciplinary process for launch vehicle design, in this paper, it is chosen to focus on the launch impact resulting from the release in the atmosphere of chemical product from the rocket engine combustion. The proposed design approach is illustrated with this impact considering that it could be extended to other environmental impacts in future works.

In order to account for the impact of the rocket emissions during the launch vehicle flight through the atmosphere, a new discipline has to be integrated in the MultiDisciplinary Analysis (MDA) of launch vehicle design process. Indeed, in the early design phases, classical MDO formulations (for instance the coupled MultiDiscipline Feasible - MDF formulation [7]) involves four coupled disciplines: propulsion, structure (mass & sizing), aerodynamics and trajectory (Figure 1) to estimate the performance of a launcher. Considering a particular value of the design variables (and the trajectory control variables), the evaluation of the performance and constraints for the launch vehicle results from the coupled simulation of these four disciplines through the solving of a non-linear system of equations (referred as MultiDisciplinary Analysis) to ensure interdisciplinary feasibility.

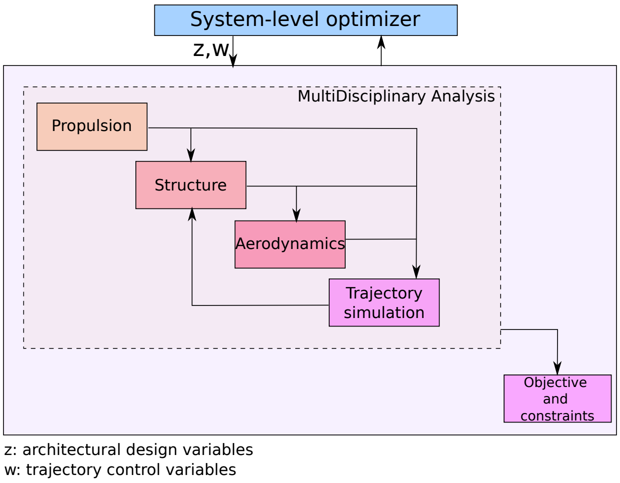

Figure 1: Classical MDO formulation for launch vehicle design

To model such a new discipline related to the atmospheric impact of the rocket emissions, it is necessary to go deeper in the involved physical phenomena. The chemical emissions resulting from the combustion of the propellants induce a modification of the atmosphere composition and therefore may lead to climate change effects. The physical phenomena due to rocket engine emissions are discussed in the next section.

2.1 Estimation of the radiative forcing during launch vehicle flight

To estimate the climate change resulting from the emissions of rocket engines during the launch vehicle flight, it is possible to estimate the radiative forcing (Figure 3). Radiative forcing (RF) is "*a measure of the influence a factor has in altering the balance of incoming and outgoing energy in the Earth-atmosphere system and is an index of the importance of the factor as a potential climate change mechanism*" [24].

The estimation of radiative forcing is a challenging task and different RF quantities have been defined [24] to ease the estimation and to provide an order of magnitude of the potential of climate change of any anthropogenic emissions. One may distinguish the instantaneous RF, the stratospheric-adjusted RF, the zero-surface-temperature-change RF and the effective RF. Considering the radiative forcing defined as the net flux imbalance at the tropopause, as illustrated in Figure 2, the instantaneous RF considers the atmospheric temperatures are fixed everywhere, the stratospheric-adjusted RF allows stratospheric temperatures to adjust but the tropopause and Earth surface are considered fixed, the zero-surface-temperature-change RF allows atmospheric temperatures to adjust everywhere with Earth surface temperatures fixed and effective RF (also called equilibrium climate response) allows the atmospheric and surface temperatures to adjust to reach equilibrium (no tropopause flux imbalance), giving a surface temperature change (ΔT_s).

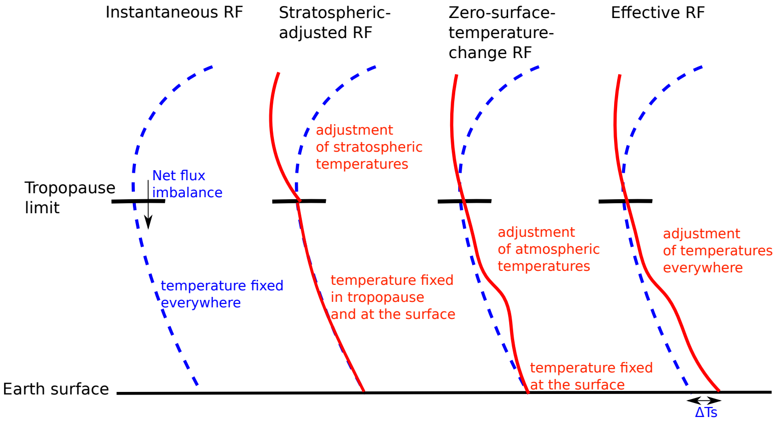

Figure 2: Definitions of radiative forcing (adapted from [24]). Blue dashed line: unperturped temperature profile. Red solid line: perturbed temperature profile.

The estimation of radiative forcing requires to compute the modification of the atmospheric concentrations due to the rocket emissions during the flight (Figure 3). However, most of the time, RF estimation involves a global climate model which is a complex mathematical model of the major climate system components (atmosphere, land surface, ocean, and sea ice), and their interactions. In practice, it relies on complex numerical simulation models which are too computationally intensive to be integrated into a MDAO framework for launch vehicle design in early design phases.

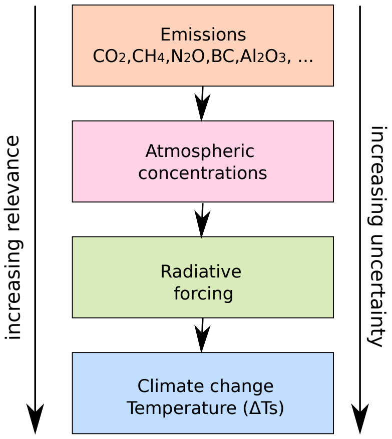

Figure 3: Steps for the estimation of climate change resulting from rocket emissions

To provide a first order magnitude of the climate change impact of launch vehicle flight, Ross and Sheaffer [21] proposed a simplified estimation of the Instantaneous Radiative Forcing (IRF). This accounts for the main chemical emissions of rocket engines during the flight: CO_2, H_2O, black carbon (BC) and Al_2O_3 (alumina). It is an analytical model that does not involve a complex numerical simulation and therefore is compatible with the MDAO framework. Mainly, two bricks are required to estimate the preliminary IRF: the rocket emissions during the flight and the IRF analytical formula. These bricks are discussed in the next sections.

2.1.1 Estimation of rocket emissions

On the contrary of aircraft which emit mainly in the tropopause, rockets during their flights emit through all the layers of the atmosphere and are the only direct source of human-produced compounds in stratosphere and mesosphere. Rocket engine combustion products are expelled through the nozzle into the atmosphere. The

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Table 1: Major primary chemical emissions for liquid rocket engines

Propellant	Major Primary Emissions
LOx/LH$_2$	H_2O, H_2
LOx/LCH$_4$	H_2O, CO_2, CO, H_2
LOx/RP-1	H_2O, CO_2, CO, H_2

chemical species present at the nozzle exit plane are called the primary emissions of the rocket engine. However, as the temperature of the rocket engine exhaust at the nozzle exit plane is very high, the chemical species in the engine plume may continue to react with each other and with the surrounding atmosphere (corresponding to the afterburning). The resulting products formed by afterburning are referred as secondary emissions. Table 1 provide the major primary chemical emissions for liquid rocket engines.

In addition to the primary emissions listed in Table 1, other minor chemical species may be produced due to different combustion phenomena such as incomplete combustion and non-equilibrium processes inside the rocket engine. For instance, Black Carbon (BC, also referred as soot) is produced inside the rocket engine combustion chamber by incomplete combustion of carbon-based propellants such as LOx/RP-1 (or solid propellant). It is expected that LOX/CH_4 rocket engines will also produce some black carbon. The non-equilibrium chemistry and inhomogeneous mixing processes involved in the formation of BC are still not fully understood and is currently an active field of research. Therefore, little is known about the amount of BC that is produced by different types of rocket propellants. However, as pointed out in [21], BC has a large impact in terms of radiative forcing, therefore a better estimation of the emission index is required in the future to reduce the uncertainty associated to that chemical product. Furthermore, combustion inefficiencies in hydrocarbon-based propellants could result in trace amounts of a complex mix of hydrocarbon emissions and impurities in the fuel and could also have as impact the residual emissions of other species that depend on the chemical composition of the impurities. Even in small quantities, these minor combustion products may have significant climate impacts.

In addition to primary emissions produced inside the combustion chamber and the nozzle, secondary emissions are formed outside of the nozzle due to afterburning and reactions with the atmosphere. Chemical species such has nitrogen oxides (NOx), hydrogen molecules (H, H_2) or carbon monoxide (CO) may be produced.

The rocket emissions are very difficult to estimate and are still an active research field. In the literature [9, 13, 21] rocket emissions are estimated either by numerical simulations (with various model fidelities) or by in-situ measurements. Measurements on a full rocket engine are difficult [13] due to complex operational conditions (*e.g.*, pressure, temperature) and therefore the estimation of rocket emissions mainly rely on numerical simulations. In the literature, mainly two types of models for rocket emission estimations may be distinguished: low fidelity models (using a simplified chemical equilibrium and mainly dedicated to primary emissions) and high fidelity models (involving complex Computational Fluid Dynamics simulation which may involve some afterburning reactions). In the studies of the literature [9, 21], the low fidelity models are either based on stoichiometric estimation of combustion products or on simple thermochemical equilibrium simulations. An estimation of the composition of the exhaust plume for a launch vehicle can be determined by converting the available propellant on-board of the launch vehicle to their nozzle exhaust products using a stoichiometric combustion. The stoichiometric approach relies on the principle of matter conservation to estimate the combustion products based on the propellant masses. The mass of exhaust product $m_{ex}(X)$ of a chemical species X is given by:

$$m_{ex}(X) = EI(X) \times m_p$$

with $EI(X)$ the emission index for the chemical species X and m_p the propellant mass. The emission index corresponds to the mass of the species emitted per kilogram of propellant burnt. A point of attention, the emission indices for aircraft and spacecraft are different: emission indices for aircraft are defined relative to fuel mass, whereas emission indices for spacecraft are defined relative to propellant mass (the fuel plus the oxidizer). The main emission differences among the various rockets depend on the propellant type. Consider for instance the emitted species CO_2, the amount of carbon dioxide emission produced depends heavily on a combination of the fuel and oxidizer used for the launch vehicle. Hydrocarbon fuels like kerosene or RP-1 (Rocket Propellant - 1) are composed of long chains of hydrocarbons that oxidize to produce carbon dioxide and water vapor. One way to estimate the amount of carbon dioxide is to assume a combustion of a series of CH_2 groups (or $C_nH_{1.953n}$ for refined kerosene as RP-1):

$$CH_2 + \frac{3}{2}O_2 \rightarrow CO_2 + H_2O \tag{1}$$

Therefore, in the stoichiometric estimation approach, the total kerosene mass is converted to mass of CO_2 in the immediate exhaust plume by the following (by accounting for the molecular mass of the different involved species):

$$m_{CO_2} = m_{RP1} \times \frac{12.01g + 16.00g * 2}{12.01g + 1.008g * 2} \tag{2}$$

Table 2: Emission Index in the literature (in kg per kg of burned propellant)

Propellant	CO	CO_2	H_20	H_2	OH	BC	Reference
LOx/RP-1	0.456	0.222	0.250	0.006	0.029	0.020	[22]
	0.240-0.399	0.335-0.470	0.250-0.284	0.006-0.015	0.000	0.025	[13]
	0.319	0.265	0.397	0.019	-	-	[23]
	-	0.600	0.350	-	-	0.010 -0.040	[21]
LOx/LCH_4	0.344	0.187	0.422	0.009	0.024	0.000	[22]
	0.051-0.189	0.360-0.492	0.439-0.452	0.002-0.011	0.000-0.002	0.000	[13]
LOx/LH_2	0.000	0.000	0.907	0.032	0.027	0.000	[22]
	0.000	0.000	0.959-0.965	0.035	0.000	0.000	[13]
	0.000	0.000	0.980	0.035	0.000	0.000	[23]
	0.000	0.000	1.0	-	-	-	[21]

Therefore the emission index $EI(CO_2)$ is given by:

$$EI(CO_2) = \frac{12.01 + 16.00 * 2}{12.01 + 1.008 * 2} \tag{3}$$

This approach may be applied for all the exhaust product species for kerosene (and RP-1) such as water vapor or carbon monoxide depending on the assumed combustion reaction. Similarly, the emission index may be defined for all couples of propellants classically used in rocket propulsion such as LOx/LH_2, LOx/LCH_4, LOx/RP-1, and solid propellants, *etc.*

Such an approach is easy to implement and provides a first order of magnitude but presents important approximations. An alternative low-fidelity approach is to rely on tools such as Chemical Equilibrium with Applications (CEA) [10] to simulate the combustion of gas in a rocket engine chamber and the gas expansion in a nozzle. It computes the chemical equilibrium compositions and properties of complex mixtures. The conditions for chemical reaction equilibrium are stated in terms of Gibbs (or Helmholtz) energy or the maximization of the entropy. The system of equations to solve the equilibrium and to obtain the chemical composition are non linear and iterative methods are used (*e.g.*, Newton-Raphson algorithm).

Considering the three main liquid propellant types: LOx/RP-1, LOx/LCH_4 and LOx/LH_2, Table 2 provides a summary of the values of Emission Index for different chemical products due to rocket engine emissions found in the literature [13, 21, 22] considering similar low fidelity models as described above.

From Table 2, it can be seen that the uncertainty associated to the emission indices for the different propellant types is large. Such a level of uncertainty is important to notice as it has a strong impact on the assessment of the climate impact of rocket engine emissions. Further researches related to the characterization of the emission indices are required to better describe the impact of chemical releases in the atmosphere.

As the purpose of the current paper is to propose a methodology for the inclusion of the climate impact of rocket emissions in the design process of launch vehicles, an affordable computational approach as to be used. In the following, the approach proposed by DeSain [9] (based on stoichiometric estimation approach of the primary emissions) is adopted. The proposed methodology could be used with other emission estimation techniques (as long as the computational cost is affordable in a design process).

2.1.2 Preliminary estimation of radiative forcing

Based on the estimation of rocket emissions as discussed in the previous section, a measure of climate impact of such emissions in the atmosphere is required for integration in the design process. Indeed, it is not possible to only consider the emissions as the different released chemical species into the atmosphere do not have the same influence on the climate impact. A preliminary estimation of Instantaneous Radiative Forcing dedicated to launch vehicle has been proposed by Ross and Sheaffer [21] and is adopted in the present work. IRF does not include the atmospheric response that brings the atmosphere back into thermal steady state (which would require a global climate model) but it allows to get a first order of magnitude of the climate impact induced by launch vehicle emissions. The IRF is calculated differently for each chemical component, corresponding to the unique radiative behavior in the atmosphere of each species. In the proposed approach by Ross and Sheaffer [21], IRF is determined using a mass-specific scattering or absorption factor σ of exhaust (for a gas or particles). It is also assumed that the scattering and absorption are separable into short-wave (SW) and long-wave (LW) components, as for the incoming and outgoing fluxes. The IRF is assumed to be defined by:

$$IRF = \left(\int I(\lambda)_{LW} \sigma_a(\lambda)_{LW} \mathrm{d}\lambda - \int I(\lambda)_{SW} \sigma_s(\lambda)_{SW} \mathrm{d}\lambda \right) MA^{-1} \tag{4}$$

with $\sigma_a(\lambda)_{LW}$ and $\sigma_s(\lambda)_{SW}$ the wavelength-dependent mass-specific absorption and scattering coefficients, respectively. The IRF is measured in W.m^{-2}. The fluxes $I(\lambda)_{LW}$ and $I(\lambda)_{SW}$ are respectively the mean (temporal and spatial) solar SW and terrestrial LW flux spectra. In addition, M is given as the steady state burden propellant mass, and, finally, A as the surface of the exhaust accumulation region, defined as the region with altitudes between 10 and 30km and a surface area of $1.2 \times 10^{14}\text{m}^2$. In practice, for each chemical species considered in the rocket emissions, a derivation of Eq.(4) is carried out including the integration and some simplifications (*e.g.*, assuming that the cross section carries the spectral and spatial integrations) to lead to an analytical equation easy to compute. For instance, for H_2O species, the resulting IRF is given by:

$$IRF_{H_2O} = \sigma_{H_2O} I_{LW} M_{H_2O} A^{-1} \tag{5}$$

with σ_{H_2O} the mass specific absoption coefficient of H_2O, I_{LW} the terrestrial LW radiation and M_{H_2O} is the stratospheric H_2O burden. The influence of the increase of water vapor in the atmosphere is not fully understood due to different aspects such as the complex nature of H_2O absorption bands or the potency of stratospheric H_2O to generate IRF that varies strongly with altitude. Indeed, H_2O in the lower stratosphere causes greater IRF than in the middle and upper stratosphere because it absorbs terrestrial LW radiation from the warm troposphere and reemits at relatively cooler temperatures in the lower stratosphere. Similarly, Ross and Sheaffer [21] provided expressions of IRF for the main chemical species for the different types of liquid rocket engine propellants such as CO_2 and black carbon. The resulting full IRF induces by a rocket launch is the sum of the IRF of the considered chemical releases into the atmosphere. It is possible to compare different launch vehicle technologies, propellant types and architectures in terms of climate impact due to their emissions during flight by comparing the preliminary estimation of the IRF. The lower the IRF, the lower climate impact may be assumed for the considered launch vehicle. It is important to notice the different simplifications in the IRF modeling and derivations providing only an order of magnitude of the impact. For more details on the estimation of IRF for the different chemical release of rocket engines, please refer to [21].

Ross and Sheaffer [21] estimated that the IRF induced by the rocket emissions into the atmosphere since the beginning of rocket launch is an order of magnitude below the aircraft. However, considering the growth of rocket launch in the last decade and the dynamic created by the New Space, it is important to develop methodologies and tools to be able to carry out trade-off in terms of launch vehicle design considering these impacts. A proposed approach is discussed in the next section.

2.2 Launch vehicle design accounting for the impact of rocket emissions

In order to integrate the climate impact of rocket emissions during launch, it is necessary to modify the classical MultiDisciplinary Analysis and Optimization (MDAO) design process (Figure 1). It is possible to include the model for preliminary estimation of Instantaneous Radiative Forcing (IRF) resulting from chemical releases in the atmosphere of the rocket engines presented in the previous section.

Different alternatives exist to transcribe the design problem into an optimization problem accounting for rocket emissions. One possible approach is to define a multi-objective optimization problem in which traditional performance such as Gross-Lift-Off-Weight or cost is optimized along with the radiative forcing representing a measure of the impact of rocket emissions on the climate. Another formulation consists in having an incremental approach and starting from a baseline, trying to reduce the radiative forcing by a certain factor, leading to a new constraint into the optimization problem.

In the following, the incremental formulation is adopted to illustrate a possible first attempt to account for such a new environmental constraint in a design process. The resulting multidisciplinary design optimization problem may be formulated as follows:

$$\begin{aligned}
\text{min} \quad & f(\mathbf{z}, \mathbf{y}(\mathbf{z}), \mathbf{w}) && (6)\\
\text{w.r.t.} \quad & \mathbf{z}, \mathbf{w} && (7)\\
\text{s.t.} \quad & \mathbf{g}(\mathbf{z}, \mathbf{y}(\mathbf{z}), \mathbf{w}) \leq 0 && (8)\\
& g_{IRF}(\mathbf{z}, \mathbf{y}(\mathbf{z}), \mathbf{w}) \leq 0 && (9)\\
& \mathbf{h}(\mathbf{z}, \mathbf{y}(\mathbf{z}), \mathbf{w}) = 0 && (10)\\
& \mathbf{z}_{\min} \leq \mathbf{z} \leq \mathbf{z}_{\max} && (11)\\
& \mathbf{w}_{\min} \leq \mathbf{w} \leq \mathbf{w}_{\max} && (12)
\end{aligned}$$

with $\mathbf{z}$ the design variables, $\mathbf{w}$ the trajectory control variables, $\mathbf{y}$ the interdisciplinary coupling variables, $f(\cdot)$ the objective function, $\mathbf{g}(\cdot)$ the inequality constraint function vector related to the traditional disciplines (*e.g.*, propulsion, aerodynamics, structure and trajectory), $g_{IRF}(\cdot)$ an inequality constraint related to the radiative forcing resulting from the rocket engine emissions during the flight and $\mathbf{h}(\cdot)$ the equality constraint function vector. Considering a coupled MDO formulation such as MultiDiscipline Feasible (MDF), the interdisciplinary coupling variables are determined by a MultiDisciplinary Analysis (MDA) by solving a system of non-linear equations ensuring the consistency of the coupling variables between the different disciplines.

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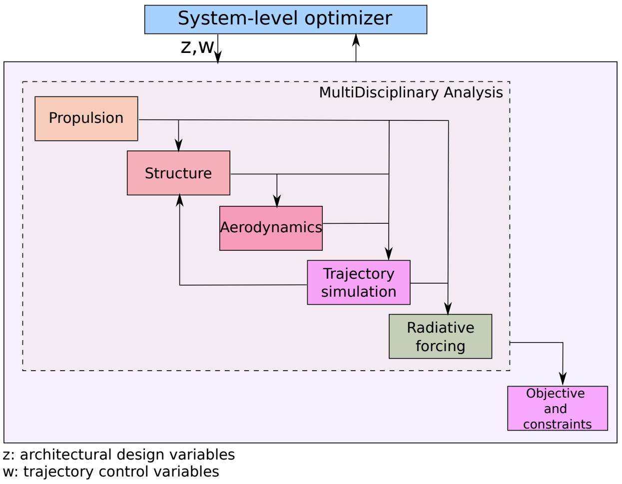

Figure 4: Introduction of radiative forcing discipline in MDF formulation

In the multidisciplinary design process, an additional discipline is introduced (Figure 4), here called radiative forcing. It takes input variables coming from the trajectory discipline (*e.g.*, amount of propellant consumed in the stratosphere), computes the corresponding chemical species emissions in the stratosphere during the flight and estimates the resulting radiative forcing. This estimate is directly compared to a maximum allowed target set as a specification threshold in the constraint $g_{IRF}(\cdot)$.

With this integrated design process, design and trajectory modifications directly have an impact on the radiative forcing. Indeed, design variables such as propellant mixture ratio have a strong impact on the rocket engine performance but also on the emissions of chemical releases contributing to the climate change and therefore a trade-off between performance and environmental impact is required. Similarly, the trajectory control of the launch vehicle has an impact on its overall performance but also directly on the amount of chemical release into the stratosphere.

In more advanced design process, additional environmental criteria could be including in the MDAO process such as resource depletion resulting from a parametric Life Cycle Analysis assessment. The main difficulty lies in the development of suited physical phenomenon modeling in an MDAO context.

3 Application

3.1 Two-Stage-To-Orbit test case

In order to illustrate the introduction of climate impact of the rocket engine emissions during the flight through the atmosphere, a Two-Stage-To-Orbit (TSTO) test case is considered. The reference mission is the injection of a 7 tons payload into a circular low Earth orbit at the altitude of 700km. Liquid rocket engines are considered for the two stages. Different types of liquid propellants are assumed and compared in the following: LOx/RP-1, LOx/LCH_4 and LOx/LH_2. The design problem consists in minimizing the Gross-Lift-Off-Weight (GLOW) of the launch vehicle while ensuring the injection of the payload into the target orbit and ensuring physical integrity of the launcher and the payload (*e.g.*, maximal axial load).

A MultiDiscpline Feasible formulation is implemented as follows:

$$\begin{aligned}
\text{min} \quad & GLOW(\mathbf{z}, \mathbf{y}(\mathbf{z}), \mathbf{w}) && (13)\\
\text{w.r.t.} \quad & \mathbf{z}, \mathbf{w} && (14)\\
\text{s.t.} \quad & \mathbf{g}(\mathbf{z}, \mathbf{y}(\mathbf{z}), \mathbf{w}) \leq 0 && (15)\\
& g_{IRF}(\mathbf{z}, \mathbf{y}(\mathbf{z}), \mathbf{w}) \leq 0 && (16)\\
& \mathbf{h}(\mathbf{z}, \mathbf{y}(\mathbf{z}), \mathbf{w}) = 0 && (17)\\
& \mathbf{z}_{\min} \leq \mathbf{z} \leq \mathbf{z}_{\max} && (18)\\
& \mathbf{w}_{\min} \leq \mathbf{w} \leq \mathbf{w}_{\max} && (19)
\end{aligned}$$

$\mathbf{z} = [m_{p1}, m_{p2}, D, OF_1]$ are the design variables corresponding to the propellant masses of stages 1 and 2 (m_{p1}, m_{p2}), the first and second stage diameters $D_1 = D_2 = D$ and the mixture ratio of the first stage OF_1. The trajectory control variables $\mathbf{w} = [\theta_i, \theta_f, \xi, \Delta t, \Delta\theta, t_v]$ are the duration of vertical lift-off (t_v), the linear pitch-over duration and angle $(\Delta t, \Delta\theta)$, and three parameters for the bi-linear tangent law for the second stage

DOI: 10.13009/EUCASS2022-6164

Table 3: Rocket engine characteristics for the different types of propellant - * denotes design variables

Propellant	Stage	Nb engines	OF	Pc (bar)	q (kg/s)	ϵ
LOx/RP-1	1	6	*	108	300	21
	2	1	2.3	97	200	117
LOx/LCH$_4$	1	4	*	108	325	15
	2	1	2.7	100	340	150
LOx/LH$_2$	1	2	*	90	330	20
	2	1	4.	40	35	80

phase $(\theta_i, \theta_f, \xi)$. Both the design vector $\mathbf{z}$ and the trajectory control vector $\mathbf{w}$ are controlled by the system-level optimizer [6]. The constraints related to the payload injection (*e.g.*, altitude, velocity and flight path angle) and to the physical integrity of the launcher and payload (*e.g.*, maximal allowed axial load, maximal dynamic pressure, maximal heat flux) are components of the inequality constraint function vector $\mathbf{g}(\cdot)$. Eventually, the climate impact of the rocket engine emissions during the flight through the atmosphere is defined with $g_{IRF}(\cdot)$. The characteristics for the considered rocket engines are given in Table 3.

Two scenarios are studied:

- the design of three launch vehicles with three types of propellant (LOx/RP-1, LOx/LCH$_4$ and LOx/LH$_2$) without constraint on the climate impact to compare the resulting IRF and define baselines IRF_b,
- the design of three new launch vehicles including a constraint on the required reduction of IRF compared to the obtained baseline in the previous study. The purpose is to analyze the resulting modifications in the launch vehicle design and trajectory introduced by IRF constraint in the design process. The constraint $g_{IRF}(\cdot)$ is formalized as follows: $g_{IRF}(\mathbf{z}, \mathbf{y}(\mathbf{z}), \mathbf{w}) = IRF(\mathbf{z}, \mathbf{y}(\mathbf{z}), \mathbf{w}) - (100 - \%r) IRF_b \leq 0$ with $IRF(\cdot)$ the estimated IRF, IRF_b the IRF of the baseline (resulting IRF of the optimal launcher of the first study) without considering the climate impact and $\%r$ the target reduction percentages. To illustrate, a reduction percentage of 5% is considered for the three type of propellants. MDO problems are solved with this reduction percentage.

Covariance Matrix Adaptation - Evolutionary Strategy algorithm (CMA-ES) [12] is used to solve the MDO problem in order to ensure global convergence (population size of 12 individuals and 2500 iterations). In the present work, the FELIN code [5], based on the opensource framework OpenMDAO [11], is used. In FELIN, four disciplines are coupled in order to determine the launch vehicle performance: propulsion, structures, aerodynamics and trajectory. In the next paragraphs, details on the disciplinary models are provided.

3.2 Disciplinary models

In this section, the disciplinary models that are used in the MDO process are described.

Propulsion: In order to take the propulsive uncertainty into account, a propulsion module has been derived from Rocket Chemical Analysis Equilibrium (CEA) model [10]. This code computes the performance of the rocket engine (specific impulse, thrust) from several inputs (chamber pressure, oxydizer to fuel ratio, *etc.*). CEA performs theoretical performance of rocket engine calculations that are adapted to conceptual and preliminary design phases. The specific impulse and the mass flow rate computed by CEA are coupling variables that are transmitted to the mass & sizing and the trajectory disciplines.

Mass and sizing: The mass and sizing module aims at computing the dry mass of the different stages from the propulsion, geometry and trajectory variables. To this end, Mass Estimation Relationships (MERs) have been used from [6] for expandable launch vehicles. All the different components of the launch vehicle (tanks, engine, nozzle, turbopumps, thrust frame, intertank *etc.*) have been modeled and the masses of these elements are computed using analytical relationships. This module allows to provide a rapid estimation of the dry mass of the launch vehicle depending on the design variables and other disciplines outputs.

Aerodynamics: The aerodynamics discipline consists in computing the aerodynamics coefficients such as the drag and lift coefficients required to compute the aerodynamics loads during the launcher atmospheric flight. Estimating the aerodynamic performance of this type of vehicles for all the different phases (subsonic, transonic, supersonic and hypersonic) is a quite challenging task. The calculations of the drag and lift coefficients are based on the ONERA code MISSILE [8] which relies simplified aerodynamics theory and on an experimental data base to determine the aerodynamics forces and coefficients of complex launcher geometries. Drag and lift coefficients tables as a function of the Mach number and angle of attack are directly given to the trajectory discipline allowing to remove the feedback loop between the trajectory and the aerodynamics. This model is generally sufficient in the early design studies.

Trajectory: The trajectory discipline consists in integrating the system of the ordinary differential equations (equations of motion) according to the time. Then, a single shooting method is used in order to define the optimal control law (parameterized pitch and azimuth angles profiles). The parameters $\mathbf{w}$ that define the trajectory control law are then optimized using a direct single-shooting method [2]. The system of equations of motion giving the state variables along the trajectory are defined by $\dot{\mathbf{x}}(t) = f_{\text{ode}}(\mathbf{z}, \mathbf{y}(\mathbf{z}), \mathbf{x}(t), \mathbf{w})$ is integrated using a 5^{th} order Runge-Kutta method involving the handling of events (fairing jettisoning, change of control law profile as a function of flight conditions, *etc.*). The control law of the pitch angle is decomposed into different phases [6]: lift-off, pitch-over maneuver, gravity turn and bi-linear tangent law. A discontinuity between gravity turn and bi-linear tangent law is allowed since no more aerodynamics forces are undertaken by the launch vehicle at this altitude. Each phase is parameterized by a set of design variables that are optimized. Constraints are involved in order to ensure that the reached apogee and perigee match the target ones and the mass of propellant that has been used during the flight is consistent with respect to the rocket architecture.

Radiative forcing: The discipline corresponds to the preliminary model for the estimation of radiative forcing presented in Section 2.1.2.

3.3 Results analysis

First, a comparison of the results obtained for the three types of propellant without considering the climate impact is carried out.

Table 4: Results of MDO problems for three different types of propellant without considering rocket emissions

Propellant	Optimal GLOW	IRFb	Optimal Design Variables	Values
LOx/RP-1	348.2t	0.0171mWm^{-2}	mp_1	252t
			mp_2	59t
			OF_1	2.43
			D	5.0m
LOx/LCH$_4$	270.3t	0.0115mWm^{-2}	mp_1	171t
			mp_2	65t
			OF_1	3.09
			D	3.9m
LOx/LH$_2$	154.0t	0.0051mWm^{-2}	mp_1	110t
			mp_2	17t
			OF_1	4.07
			D	3.0m

In Table 4, the results from the MDO problem solving for the three types of propellant. In this problem, the environmental impact induced by the rocket engine emissions is not considered in order to provide a baseline for comparison. First, it can be seen that in terms of ranking, LOx/RP-1 has a higher IRF than LOx/LCH$_4$ that has a higher IRF than LOx/LH$_2$. The low IRF for LOx/LH$_2$ compared to the two other types of propellant is mainly due to the absence of black carbon emissions. These results in terms of impact of the different types of propellant are in line with previous study results [21].

Then, the launch vehicle design problems are solved with the radiative forcing constraints. This constraint requires a reduction of 5% of the baseline radiative forcing found in the previous optimizations. Table 5 presents the results of MDO problem solving with radiative constraint. First, all resulting launchers are able to decrease by 5% the radiative forcing. Moreover, for the three obtained launchers, an increase in GLOW (below 1%) is observed due to the addition of the climate impact constraint. Moreover, in order to be able to deal with this new constraint, the mixture ratio has been modified (increased) to reduce the amount of fuel burnt and to modify the rocket engine emissions. However, because of the modification of the mixture ratio, the rocket engines are less efficient in terms of specific impulse (Isp). For instance, for the LOx/RP-1 propellant, the Isp decreases from 308.4s for the baseline to 304.9s for the constrained launcher (see Table 6 for all the results).

In addition, as the mixture ratio of the second stage is fixed in the optimization problem, in order to decrease the IRF compared to the baseline, the propellant mass of the first stage is increased and the propellant mass of the second stage is decreased (see Tables 4 and 5). Therefore, the staging between the two stages is modified. By doing that, it is possible to act on the mixture ratio of a larger quantity of propellant for the first stage and therefore to modify the chemical products of a larger amount of propellant. Figures 5, 6, 7, 8 display the altitude (with and without the coast phase), velocity and flight path angle for the optimal launcher for the

DOI: 10.13009/EUCASS2022-6164

Table 5: Results of MDO problems for three different types of propellant with a reduction of 5% of IRF compared to the baseline

Propellant	Optimal GLOW	IRF	Optimal Design Variables	Values
LOx/RP-1	350.4t	0.0162mWm^{-2}	mp_1 mp_2 OF_1 D	258.1t 55.5t 2.65 4.8m
LOx/LCH$_4$	271t	0.0109mWm^{-2}	mp_1 mp_2 OF_1 D	178t 60t 3.37 3.44m
LOx/LH$_2$	154.5t	0.0048mWm^{-2}	mp_1 mp_2 OF_1 D	111.2t 16.1t 4.41 3.87m

Table 6: Comparison of the specific impulse for the baseline and the constrained launchers

Propellant	Isp baseline	Isp constrained
LOx/RP-1	308.4s	304.9s
LOx/LCH$_4$	323.8s	322.0s
LOx/LH$_2$	422.1s	419.7s

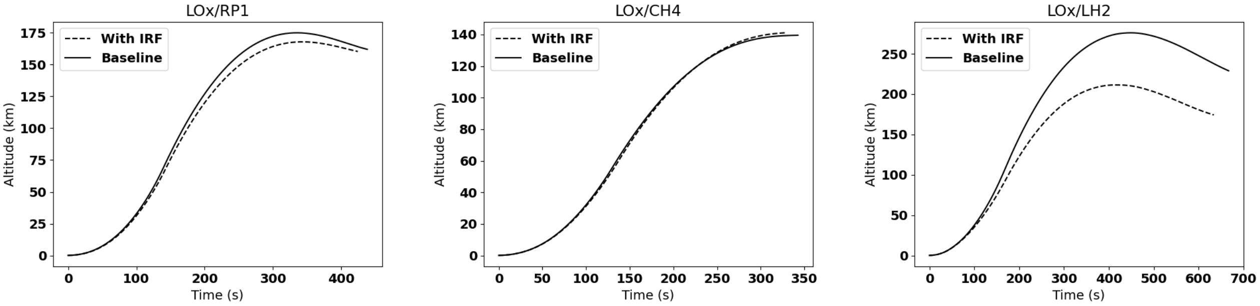

Figure 5: Altitude (without coast phase) comparison between baseline and with IRF constraints

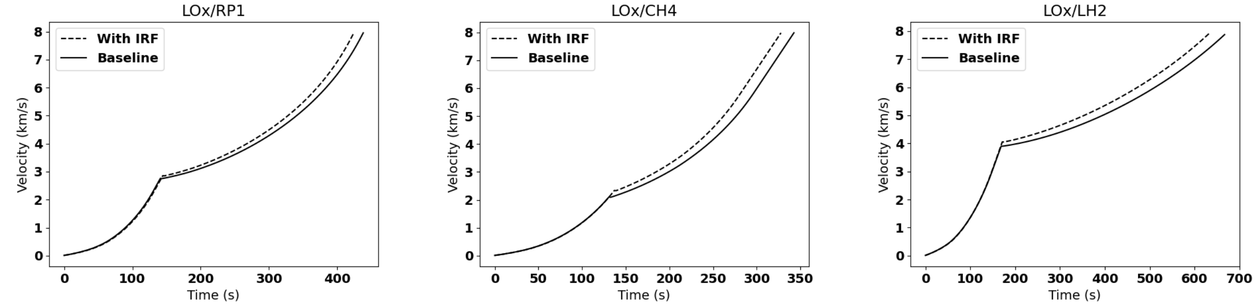

Figure 6: Velocity comparison between baseline and with IRF constraints

three different propellant types. Therefore, the integration of a climate change constraint in the design process of launch vehicle design leads to modification of the design (*e.g.*, modification of mixture ratio, modification of staging) in order to meet the new constraint. Such design modifications can only be assessed by using a multidisciplinary design process.

4 Conclusions

In this paper, a methodology to include the impact on climate change of the rocket engine emissions into a multidisciplinary design process at early design phase has been proposed. A new discipline estimating the radiative forcing caused by the emissions of chemical products into the atmosphere during the flight has been

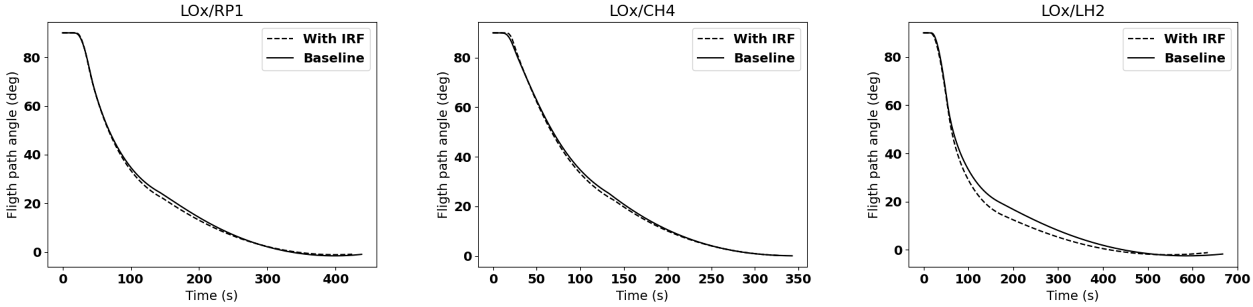

Figure 7: Flight path angle comparison between baseline and with IRF constraints

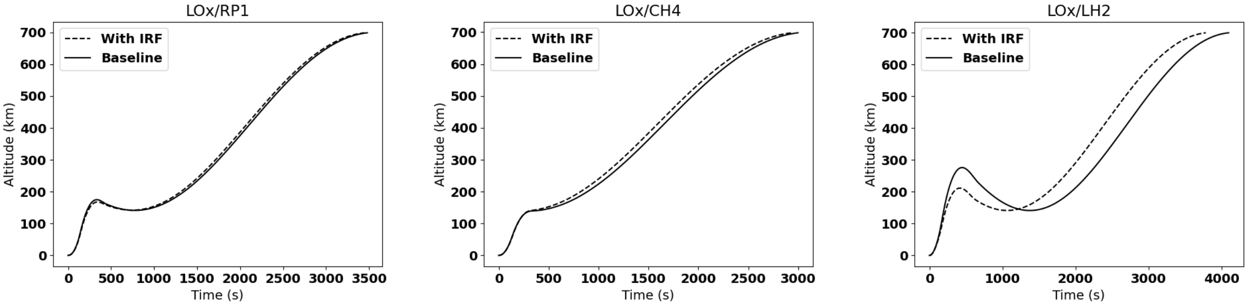

Figure 8: Altitude with coast phase comparison between baseline and with IRF constraints

added into the design process. The rocket emissions are estimated based on low fidelity models to be compatible with the computational cost induced by the MDAO framework. Then, an estimation of the instantaneous radiative forcing (IRF) is carried out using simplified modeling in order to provide an order of magnitude of the climate impact of rocket emissions. A new MDO problem is defined accounting for this new discipline under the form of a constraint to decrease the IRF compared to a baseline.

This approach has been applied to a Two-Stage-To-Orbit design problem by comparing three types of liquid rocket engine propellants: LOx/RP-1, LOx/LCH$_4$ and LOx/LH$_2$. The introduction of a new climate change constraint leads to the modification of the launch vehicle characteristics (*e.g.*, mixture ratio, staging) and the associated trajectory in order to deal with the required IRF reduction.

The proposed methodology is generic and could be adapted to other environmental impacts (such as resource depletion, biodiversity, human health, pollution, *etc.*) by introducing new disciplinary models estimating these impacts. If a large number of criteria is considered, a new multi-criteria design process has to be defined. As discussed in the paper, a high level of uncertainty is present in the modeling of rocket engine emissions and in the estimation of radiative forcing. Further works are required to improve the modeling fidelity and to include them in a MDAO framework (using surrogate models for instance). Moreover, the introduction of modeling uncertainty into the design process could help to carry out reasonable trade-off between different technologies for launch vehicle design.

Acknowledgments

The design process has been built using OpenMDAO library [11].

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