F. Castellini, M. Lavagna, A. Riccardi, C. Buskens
Quantitative Assessment of Multidisciplinary Design Models for Expendable Launch Vehicles
Journal of Spacecraft and Rockets, Vol. 51, N. 1, 2014, p. 343-359
doi:10.2514/1.A32527

JOURNAL OF SPACECRAFT AND ROCKETS
Vol. , No. ,

Quantitative Assessment of Multidisciplinary Design Models for Expendable Launch Vehicles

Francesco Castellini* and Michèle R. Lavagna†
Politecnico di Milano, 20156 Milano, Italy
and
Annalisa Riccardi‡ and Christof Büskens§
University of Bremen, 28359 Bremen, Germany

1 DOI: 10.2514/1.A32527

The research effort described in this paper is centered on the development and quantitative assessment of a multidisciplinary design optimization environment for the early design phases of expendable launch vehicles. The focus of the research is on the engineering modeling aspects, with the goal of evaluating in detail the accuracy of engineering-level methods for launch vehicle design, both in terms of disciplinary errors (e.g., engine specific impulse evaluation) and of system-level sensitivities, to assess their applicability to industrial early design. Although aerospace applications of multidisciplinary design optimization can be found in literature, the systematic assessment of the models' accuracies to the extent described in the present research is a rather new endeavor, which is critical for the advancement of this field. In fact, the widespread industrial application of multidisciplinary design optimization has often been obstructed by the difficulty of finding a suitable compromise between the analysis fidelity and computational cost. Considerable effort was therefore spent on a careful, incremental modeling process, with the purpose of overcoming such an obstacle. As a result, although it is clear that the development and tuning of a reliable multidisciplinary environment is a particularly complex and challenging task, detailed investigations showed that a good compromise can indeed be achieved for the expendable launch vehicles application. In particular, the 1σ accuracy on the payload performance was assessed to be in the order of 12% for a computational time <2 s per design cycle, allowing one to obtain physically sound design changes through the multidisciplinary design optimization, even exploiting only fast engineering-level methods.

Nomenclature

A_b	=	burn area, m^2
$C_{D/L/m}$	=	drag/lift/pitching moment coefficient
I_{sp}	=	specific impulse, s
L/D	=	length-over-diameter ratio
M	=	mass, kg
M_{inert}	=	inert mass, including unused propellants, kg
M_{prop}	=	propellant mass, kg
n_{ax}	=	axial acceleration, g
n_{lat}	=	lateral acceleration, g
p_{cc}	=	combustion chamber pressure, bar
p_{tanks}	=	tanks pressure, bar
q_{dyn}	=	dynamic pressure, Pa
q_{heat}	=	heat flux, W/m^2
r_b	=	burning rate, cm/s
T	=	=thrust, kN
α	=	angle of attack, rad
Δt_{PO}	=	pitch pushover duration, s
$\Delta\theta$	=	pitch pushover entity, rad
$\Delta\theta_{BTL,i}$	=	bilinear tangent law pitch initial value, rad
θ	=	pitch angle, rad
$\theta_{BTL,f}$	=	bilinear tangent law pitch final value, rad
ξ	=	bilinear tangent law shape parameter
ψ	=	yaw angle, rad

I. Introduction

SINCE the dawn of the space era, the development, production, and operations of launch vehicles have been very costly businesses. Although the past two decades have witnessed several development efforts focused on the reduction of the launch prices, the most remarkable being constituted by Space-X Falcon vehicles, the cost for access to space is still a major obstacle to the growth of space exploration and exploitation.

It is commonly recognized that most of the life cycle costs (LCC) of launch vehicles and space programs in general, approximately around 80% [1], is determined early in the conceptual phase, whereas detailed design decisions have much smaller effects. The quality of the early design process is for this reason critical to reduce space programs costs, for which quality is intended as the capability to produce design solutions as close as possible to the optimum, defined with respect to one or more key design drivers, such as the LCC. Multidisciplinary design optimization (MDO) was therefore chosen as the topic of a wide Ph.D.¶ research effort, undertaken in collaboration with the ESA, in light of its potential for improving the initial design processes of complex systems. In line with European background and objectives, classical unmanned expendable launch vehicles (ELVs) were defined as the applicative scenario to be investigated. Other classes of space transportation systems were also studied and represent natural extensions of the work presented here, but the engineering models were not implemented. The MDO architecture, optimization algorithms, and engineering models were developed in a C++ software tool named Space Vehicles Analysis and Global Optimization (SVAGO).

MDO is a relatively new design methodology, realizing the coupling together of two or more analysis disciplines with numerical

Received 14 October 2012; revision received 16 May 2013; accepted for publication 29 May 2013; published online XX epubMonth XXXX.

*Ph.D., Aerospace Engineering Department, Via La Masa 34; castellini@aero.polimi.it. Student Member AIAA.

†Assistant Professor, Aerospace Engineering Department, Via La Masa 34; lavagna@aero.polimi.it. Senior Member AIAA.

‡Ph.D., Zentrum für Technomathematik, Bibliothekstraße 1; riccardi@math.uni-bremen.de.

§Professor, Zentrum für Technomathematik, Bibliothekstraße 1; bueskens@math.uni-bremen.de.

¶Within ESA's Ph.D. Program in Education for Space, Technology, Innovation and Knowledge; information available online at http://www.esa.int/esaMI/Education/SEMBTFO4KKF_0.html [retrieved 29 July 2013].

optimization methods, which has the potential of drastically improving both the efficiency of the design process and the quality of the design solutions. It was defined as "a methodology for the design of complex engineering systems and subsystems that coherently exploits the synergy of mutually interacting phenomena" by NASA's Langley Research Center's Multidisciplinary Design Optimization Branch [2] or in more simple words by AIAA's MDO Technical Committee as "how to decide what to change, and to what extent to change it, when everything influences everything else" [3]. A traditional engineering practice usually involves time-consuming design iterations of disciplinary experts, with all associated costs. Moreover, it is likely to result in suboptimal design, as illustrated in [4]. The MDO approach allows one to drastically reduce the design cycle times through the automation of the multidisciplinary design analysis (MDA) and is theoretically capable of achieving the global optimum by concurrently varying all the involved design variables [4]. If successfully applied in the initial design phases of complex engineering systems, MDO should therefore ensure significant monetary advantages [5] by both increasing the design process efficiency and improving the design quality (e.g., identification of a solution with the lowest LCC).

In summary, MDO software environments can be tremendously useful tools for designers, reducing cycle times and identifying the best design concepts, if sufficient interactivity allows them to effectively introduce the human experience in the automated loop, actively steering the optimization process toward the most promising regions of the search space. Nevertheless, successful industrial applications reaching the hardware development phase are still extremely rare (e.g., NASA/Boeing's Blended-Wing-Body research aircraft [6]). This is due in part to the resistance of design offices to the introduction of MDO, which requires a large initial investment in software and personnel, and in part to the technical challenges in the areas of engineering modeling, hardware and software infrastructure, mathematical analysis, and optimization, which come with the practical implementation of MDO. In particular, one specific aspect was identified as the most critical: the tradeoff between the engineering models' simplicity and accuracy. In fact, it is still not clear how to obtain reliable design information for industrial-level early project phases when accepting the compromises in design fidelity that are necessary to limit the computational efforts. For instance, computational fluid dynamics (CFD) and finite elements modeling (FEM) approaches are not well suited to MDO deployment unless very large resources of modern computer clusters are invested and the design freedom is limited to one or a few specific configurations.

In this context, the present research work is targeted at assessing the industrial applicability of MDO for early ELV design by quantitatively evaluating the accuracy of the engineering models both in terms of disciplinary errors and of system-level sensitivities. Note that the first relevant examples of multidisciplinary models for launch vehicles date back to the Ph.D. efforts of Olds [7] and especially Braun [8]. These seminal works were focused on problem decomposition techniques and MDO architectures, introducing novel methodologies and demonstrating the potential of MDO for large-scale practical problems. However, the application was limited to a single-vehicle configuration, the disciplinary models were tailored for the specific case, and the accuracy of the engineering analyses was not extensively evaluated. Many subsequent launch vehicle applications of MDO are documented in literature, as for example [9–12], introducing more complex engineering-level models but still lacking a rigorous assessment of the global accuracy of the MDA and hence not conveying a precise feeling of the reliability of the MDO design solutions.

Having identified this gap, the idea at the basis of the present work was to study the complex relations among the different disciplines, identify the main drivers for the global accuracy of the MDA process, and quantitatively assess the suitability of the developed models for real-world industrial applicability, both in terms of accuracy and of computational requirements. To derive considerations of general validity for all ELVs, models capable of tackling any kind of unmanned configuration were implemented, introducing a level of complexity that was not considered in the previously mentioned literature sources. Because of the limitation in the available computational resources and to the excessive technical complexities, the exploitation of high-fidelity analyses such as FEM or CFD was not considered, targeting instead traditional engineering-level models.

The focus of the paper is entirely on the engineering modeling aspects, with the necessary global and local optimization infrastructure only briefly described as the mathematical tool enabling MDO. The main question that is proposed in the paper can be formulated as the following: Is it possible to reduce the complexity of the design models for ELVs to the point that a full MDA can be executed within a few seconds on a single processor, without losing the accuracy necessary to place confidence on the achieved design solutions and tradeoffs, i.e., around 10%.

The availability of a relatively accurate MDA process requiring only seconds to execute, in contrast with the minutes or hours typical of CFD/FEM, would solve the previously mentioned critical issue of finding a good compromise between analysis fidelity and simplicity. This would allow for large search space exploration within the MDO loop, considering arbitrary vehicle configurations, in reasonable computational times and possibly without the need for high-performance computing (HPC).

To answer the previous question, a great deal of technical sophistication had to be introduced in the engineering models, which were developed in two steps. First, a conceptual-level modeling environment was developed, implemented, and tested. A thorough validation procedure and critical analysis of the results, together with an independent review from ESA, highlighted the key weaknesses of these models. A wide range of upgrades spanning all disciplines was identified, enabling higher fidelity and larger functionality at a reasonable price in terms of computational effort. The enhancements were implemented in a second modeling step, targeted to the early preliminary design, with a further validation campaign assessing the improvements in accuracy. No hard constraint was a priori imposed on the design cycle times, but an order-of-magnitude target was set to 1 s for the conceptual models, to be possibly relaxed for the new early preliminary analyses in front of significant improvements.

This incremental strategy founded on the critical analysis of the validation results obtained at each step was at the basis of the engineering modeling effort and resulted in the achievement of good global performance accuracy with limited computational times, as will be presented in the paper. Such a rigorous procedure for the development and assessment of the MDA constitutes the most innovative aspect of the present research work since no literature source provides quantitative evaluations of the engineering methods to the extent described here, to the authors' knowledge. In a research field that seems mature for more important applications, the lack of accuracy and reliability evaluations represents a "showstopper" for the industrial applicability since large initial investments in MDO technology can only be justified in front of a certain confidence in the results that can be achieved. Hence, the information obtained within this research represents a relevant original contribution to the field of MDO, providing useful hints for the adoption of this design approach and for the evaluation of its suitability to different scenarios. In particular, results related to the disciplinary models errors, their system-level sensitivities and criticalities, and the effects of optimization when applied to different typologies of analysis models hold the general validity for the design of ELVs.

In the continuation of the paper, Sec. II presents a brief description of the optimization methods and MDO architecture enabling the application of MDO. Sections III and IV then detail the conceptual-level step, describing the modeling and the validation procedure, respectively. Stemming from a critical analysis of the results, the enhanced early preliminary design environment is the subject of Secs. V and VI, in which the comparison of the accuracies of both steps is also reported. Finally, the most relevant lessons learned from the research are given in Sec. VII, together with insight for further improvements in the modeling of ELVs and other space transportation systems.

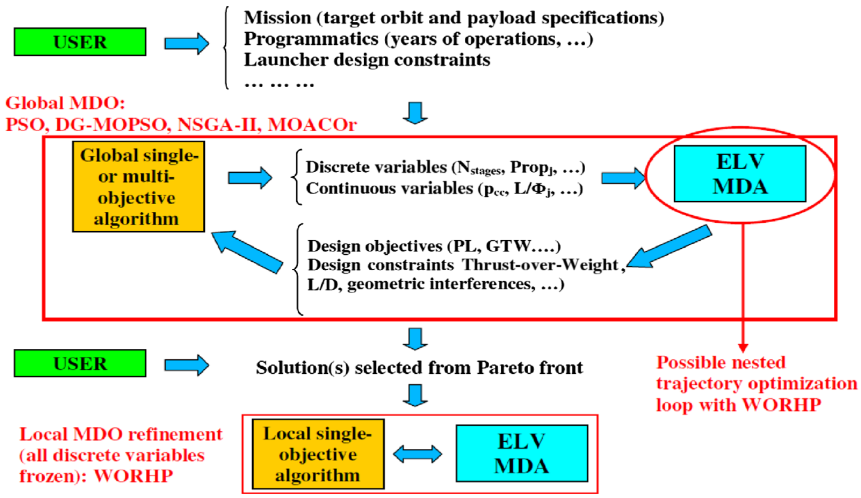

Fig. 1 Overview of the single-level BBO architecture defined for the MDO process.

II. MDO Architecture and Optimization Approach

As mentioned in the introduction, the MDO approach was implemented in a software environment named SVAGO. Although the paper is focused on the engineering modeling aspects, a brief overview of the MDO architecture and optimization approach is given here, whereas more details can be found in [13]. The software environment is constituted of an extensible markup language (XML) central database, an input/output infrastructure particularly suited for MDO, a general-purpose optimization framework, and multidisciplinary design models for ELVs. Its implementation was carried out focusing on a few key concepts. First, the optimization architecture was kept as general as possible, including stochastic methods for search space pruning, tradeoffs involving categorical and integer variables and multi-objective problems, and a gradient-based algorithm for efficient subproblem optimization and solutions refinement. Particular attention was then paid to the modularity of the object-oriented code and to the flexibility of the data storage structure to improve maintainability and extendibility beyond the realm of ELVs. Computational efficiency was also among the development drivers, due to the large central processing unit (CPU) resources required by the MDO approach. Finally, strong user interactivity was implemented, with the purpose of allowing the introduction of human expertise in the MDO loop by active user control of the process. Through the modification of optimization variables, constraints and objectives, multiple restarts with different settings, real-time results inspection, and selective local refinement, the designer can, in fact, effectively steer the optimization toward the most promising regions of the search space.

The overall MDO architecture is based on a straightforward black-box optimization (BBO) approach, in which all disciplines are integrated in a single MDA block constituting a black-box function, which exchanges information with the optimization layer only through the design variables (inputs), objectives, and constraints (outputs). BBO represents the most simple and effective approach when the problem's size and computational requirements are not excessive. A single system-level optimization process is employed, capable of global exploration and handling all design variables, constraints, and objectives, as shown in Fig. 1. The process flow starts at the user, who provides all the required input parameters at the system and discipline levels. A global MDO run is then executed with one of the available optimizers, which recursively calls the ELV MDA block until the convergence criteria are met. Finally, the user can inspect results and, if deemed necessary, select one or more of the Pareto-optimal solutions for a successive local refinement run (the same MDA model, all discrete variables constrained to the optimal value from the global run).

The system-level optimizer can be chosen among several stochastic global algorithms: a simple single-objective particle swarm optimization (SO-PSO) [14] or one of the double-grid multi-objective PSO (DG-MOPSO, proposed in [15]), NSGA-II [16], MOACOr [17], or their hybridization [18], which was developed ad hoc for ESA's Program in Education for Space, Technology, Innovation and Knowledge research. A previous research work [19] describes in detail the use of such global optimization strategies for the application to trajectory and multidisciplinary designs, including an extensive quantitative comparison on mathematical benchmarks and representative test problems; hence, no further details are reported here. The optimization architecture is completed by a state-of-the-art gradient-based algorithm called WORHP [20], which was developed by the University of Bremen and Universität Würzburg and externally linked to the MDO environment. WORHP is a sequential quadratic programming method, designed to robustly solve large-scale sparse nonlinear programming problems but is also suited for smaller and denser problems and can hence be employed within SVAGO for efficient subproblem optimization (e.g., trajectory optimization) and for the local refinement of previously obtained global solutions.

The complex high-level software architecture of the MDO design environment is schematically represented in Fig. 2. It is worth noting that the input XML file generated by a graphical user interface (GUI) is automatically parsed to C++ objects defining a central system database (CSD), which stores all data regarding user input selections, mission parameters, vehicle design, optimization settings, and output results. A core class named central system intelligence (CSI) manages all the flow of information, handling the optimization process and defining the MDA's call sequence. All communications among subsystems only occur through the CSD so that no discipline-to-discipline interface has to be defined. This hierarchic structure greatly enhances the maintainability and extendibility of the MDO environment since only the interfaces with the CSD and the execution rules for the CSI are affected by the modification, replacement, or addition of any subsystem.

III. Multidisciplinary Modeling for Conceptual Design of ELVs

The engineering modeling of launch systems is a particularly complex task, even when restricting the target to classical (i.e., simple cylindrical stages and boosters with no wings), expendable, and unmanned vehicles. In the first step of the research described here, the models are kept simple enough to allow the execution of a full MDA on a single processor in about 1 s or less. When this constraint is combined with the need of exploiting only freely available tools, the choice of the engineering models is rather limited. For this reason, many researchers in the past (e.g., [9,10,21,22]) have independently converged toward common codes, such as Chemical Equilibrium with Applications (CEA) for propulsion performance and Missile DATCOM for aerodynamics, or analogous in-house developed software. This common approach was complemented by the use of

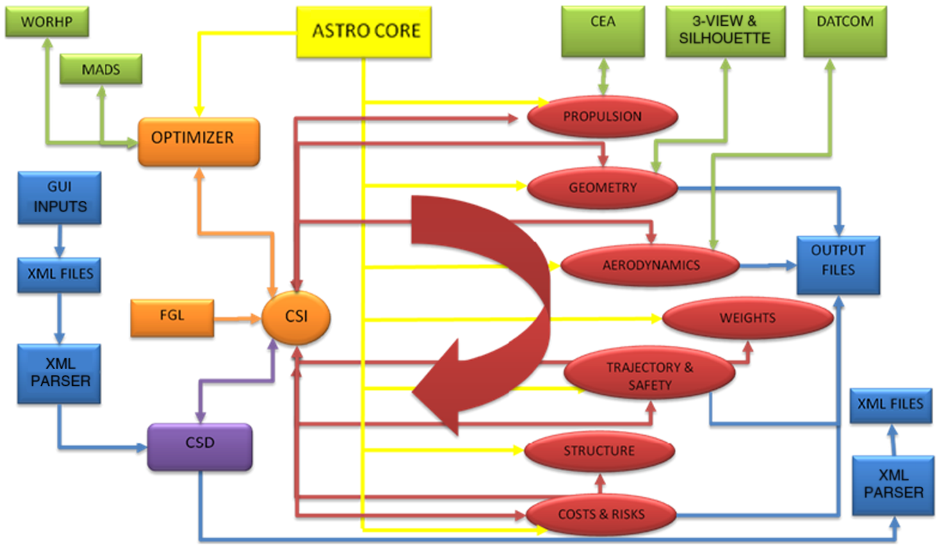

Fig. 2 High-level software architecture for the developed MDO environment. The optimizers WORHP and MADS (Mesh-Adaptive Direct Search) are external codes linked to SVAGO. A First Guess Layer (FGL) provides an initialization of all optimization variables to the CSI, which then proceeds to executing the MDA and/or MDO processes.

different modeling fidelities according to the anticipated impact of each discipline on the global performance. Several assumptions were taken for this purpose, such as largely favoring the propulsion system analysis with respect to other less relevant disciplines as aerodynamics and weights. Such assumptions were a posteriori verified with sensitivity studies, aimed at identifying the areas more critically requiring model enhancements. Stemming from these considerations, the following paragraphs trace an overview of the MDA process and the related disciplinary models.

A. System Level

A graphical tool very well suited for representing engineering design cycles is the design structure matrix (DSM), which is shown in Fig. 3 for the conceptual MDA of ELVs. The DSM shows the execution flow and exchange of data among the different disciplines, with the design proceeding sequentially along the diagonal from left to right, and terms above/below the diagonal constituting feedforward/-backward information. The designed MDA cycle is constituted by propulsion, geometry, aerodynamics, weights, trajectory, costs, and reliability, with the involved vectors of user parameters, optimization variables, cross-coupling variables, and disciplinary outputs qualitatively detailed in Fig. 4. The common practice (e.g., [10,21,23]) in launchers design is to set up a feedback of the structural, thermal, and possibly control loads from the trajectory module, requiring an iterative loop to close the design cycle. However, due to the simplicity of the weights estimation process (see Sec. III.E), the only trajectory parameters to be fedback are the maximum encountered axial acceleration, heat flux, and dynamic pressure (below-diagonal term X_{54}). To eliminate such feedback and avoid time-consuming iterations, the three load parameters are introduced instead as system-level optimization variables, to be simultaneously used both in the weights module as mass estimation inputs and in the trajectory module as thresholds for the path constraints.

The launcher design and trajectory optimization variables are divided into categorical, integer, and continuous variables. The integer variables include the number of stages, boosters, and engines. Categorical variables include architectural parameters [e.g., boosters configurations, the common core boosters (CCBs) option, etc.], technological tradeoffs for propulsion [off-the-shelf (OTS) vs new

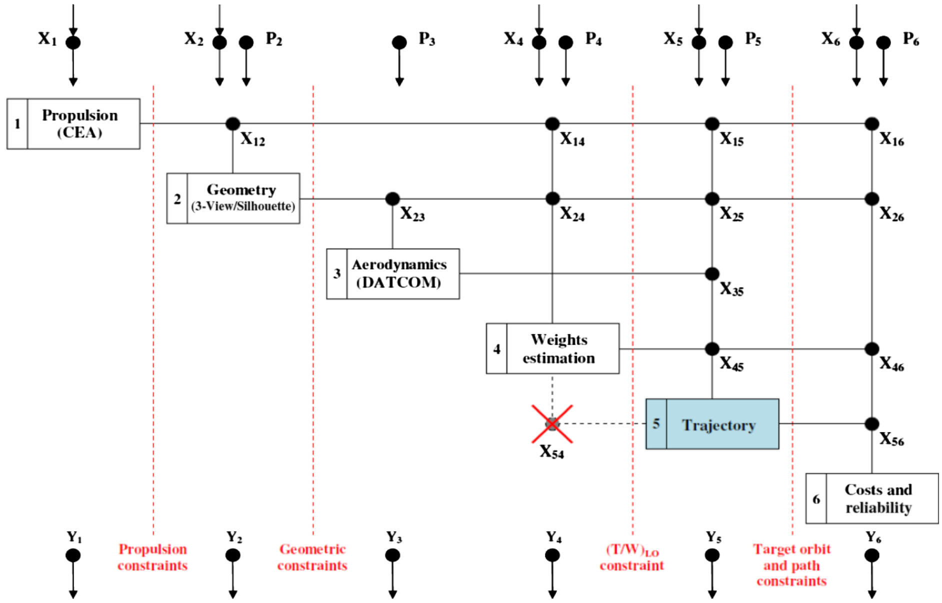

Fig. 3 ELVs' conceptual-level DSM, with P_j, X_j, X_{jk}, and Y_j vectors qualitatively described in Fig. 4.

P_2	Payload (PL) length and diameter	X_{12}	Propulsion, system lengths and diameters	X_{46}	All nonpropulsion masses		
P_3	Aerodynamics, database discretization settings	X_{14}	Propulsion, system masses and COG positions	X_{56}	Flight phases durations		
P_4	PL maximum, heat and structural loads	X_{15}	I_{sp}, minimum, operative altitude, exhaust area	X_{54}	Encountered loads: this feedback is eliminated by using maximum, heat and structural, loads as optimization, variables, input to weights and constraints to trajectory.		
P_5	PL mass and target orbit (a, e, i)	X_{16}	Propulsion, system masses				
P_6	Program and cost factors	X_{23}	Complete launch vehicle geometry	Y_1	Propulsion constraints, all propulsion, system, specifications		
X_1	Architecture, propulsion, design variables	X_{24}	Lengths for all stages and boosters	Y_2	Geometry constraints, complete LaWGS geometry		
X_2	Architecture	X_{25}	Aerodynamic reference area	Y_3	Complete vehicle aerodynamic database		
X_4	Architecture, nonpropulsion, techonology, tradeoffs, maximum, heat and structural loads	X_{26}	Fairing length and volume	Y_4	Detailed weights breakdown structure, total mass at launch, takeoff T/W constraint		
X_5	Architecture, propulsion design variables, trajectory control parameters	X_{35}	Complete vehicle aerodynamic database C_L, C_D, C_m (Mach, AOA)	Y_5	Nominal trajectory data, final orbit data, path constraints profiles		
X_6	Architecture, all techonology, tradeoffs, cost & reliability oriented variables	X_{45}	All nonpropulsion masses and COG longitudinal positions	Y_6	Detailed cost breakdown structure, mission success probability vs mission time, LCC, global reliability		

Fig. 4 Qualitative definition of the DSM's parameters for each discipline j: P_j (fixed parameters), X_j (optimization variables), X_{jk} (coupling variables with discipline k), and Y_j (disciplinary outputs).

design, propellants, feed system, nozzle and thrust vector control types, and restart and throttle capabilities], geometry (constant vs variable diameter) and weights (tanks arrangements and types, materials concept, and smarts redundancy level), and cost/reliability-oriented variables (horizontal/vertical processing, number of qualification tests for the engines, low-cost engine option, engine-out capability). Continuous variables describe instead stages'/boosters' geometry (length over diameters) and the propulsion system (propellant mass, nominal thrust, chamber pressure, area ratios, mixture ratio, etc.), and trajectory load parameters (heat flux, axial acceleration, and dynamic pressure) and trajectory control variables (pitch, yaw and thrust laws, ignition times, and coast durations).

The resulting number of launcher design variables equals $13 + 26 \cdot (N_{\text{stages,max}} + N_{\text{BoosterSets,max}})$, where $N_{\text{stages,max}}$ and $N_{\text{BoosterSets,max}}$ represent the maximum number of stages and of booster sets to be included in the design. The number of trajectory optimization variables instead largely varies depending on the controls discretization settings. However, feasible trajectories with near-optimum payload capacity can be obtained with very small problems (i.e., one parameter per each control per each flight phase, summing up to 10–15 optimizable parameters), allowing for the best robustness and efficiency of the process at the price of a very small performance loss. All optimization variables (discrete or continuous) can either be fixed by the user or allowed to be decided by the system-level optimization algorithm within user-provided boundaries, ensuring flexibility to the optimization framework.

A number of important constraints is imposed in the trajectory module, in the form of final errors on the orbital parameters, and path constraints are imposed on the heat and structural loads, static controllability, and ground-track direction. Design-related constraints are also imposed on figures such as the liftoff thrust to weight, geometric interferences, and the thrust range for each technology. Finally, the failure of one of the external design tools (CEA or DATCOM) is also accounted as a constraint violation. The objectives of the optimization can be selected from four available criteria, or any of their combinations through weighting factors, for single- or multi-objective optimization. These are the gross takeoff weight (GTW), the payload (P) excess with respect to the required performance on the reference target orbit, the cost per launch (CPL), and the mission success probability (MSP).

B. Propulsion

The propulsion analysis for each stage/booster is performed by either picking up an OTS system from a database of 38 currently flying liquid rocket engines (LREs), collected from the International Guide to Space Launch Systems [24] and several web sources,** or by designing a new liquid propellant (LP) or solid propellant (SP) system. The choice of an OTS or new engine can be either imposed by the user or optimized through a dedicated categorical variable. For new designs, the chamber pressure, mixture ratio (LP only), and expansion ratio are optimized in different ranges depending on the propellants and feed system type, and NASA's CEA [25,26] is executed to compute the theoretical performance. Empirical corrections derived from standard propulsion sources [27,28] and further calibrated with the engines' databases are applied for I_{sp} losses, whereas the inert masses are obtained through weight estimation relationships (WERs) developed from the previously mentioned database. Finally, simple models are implemented for the estimation of the geometric dimensions (following the scaling based on a nominal thrust level that can also be optimized) and for the minimum operational altitude due to shock waves in the nozzle.

C. Geometry

All stages and boosters are assumed to be cylindrical, with either cylindrical or conical interstages, whereas fairing ogives follow power laws. The only optimization variables are therefore Boolean variables for the continuity of the diameter from one stage to the next and, in the case of discontinuities, the length-to-diameter ratios. The external geometry is mainly determined by the propulsion system's dimensions plus additional volumes accounting for interstages, intertanks, and equipment compartments. All length and diameter parameters are translated into geometry files in the Langley Wireframe Geometry Standard (LaGWS) [29], which was selected as the geometry format due to its simplicity and common interfaces with many analysis codes. Additionally, Gnuplot-based visualization tools are linked to the MDO environment for visual inspection. These are 3-View and Silhouette from the Public Domain Aeronautical Software,†† respectively, for two-dimensional and three-dimensional plots.

D. Aerodynamics

The largest influence of aerodynamics on the global performance of ELVs is in the subsonic and low supersonic regimes (i.e., Mach = [0.6–3]), for which even linear aerodynamics panel codes involve rather high computational loads, certainly not compatible with the 1 s design cycle time requirement. Nevertheless, this influence is still fairly limited, as will be confirmed by the sensitivity analyses described in Sec. IV.B; therefore, Missile DATCOM [30] was selected as an analysis tool to determine aerodynamic coefficients as a function of Mach and total angle of attack (AOA). Being a collection of semi-empirical methods with a components buildup approach, DATCOM allows a database of 20 by 5 Mach/AOA points to be generated in about 0.2 s and can therefore be efficiently

**Information available online at http://www.astronautix.com, http://www.russianspaceweb.com, http://space.skyrocket.de [retrieved 29 July 2013].

††Information available online at http://www.pdas.com [retrieved 29 July 2013].

executed within the MDA loop, with the purpose of obtaining C_L and C_D for trajectory integration as well as C_m (the pitching moment) for the static controllability verification.

E. Weights

No structural analysis model is implemented in the conceptual-level environment due to the excessive computational load associated with FEM and the complexity increase related to simplified methods such as beam approximations. Hence, simple WERs are implemented for both structural and nonstructural weights, mainly taken or adapted from a comprehensive published collection [31]. The only exception is related to SP systems, for which the grain case's mass is computed with pressure vessel sizing relations, which, despite their simplicity, have shown a very good correlation against a database of solid rocket motors (SRMs).

F. Trajectory

A 3 degrees of freedom (DOF) dynamics and limited environmental models (zero-order gravity, US 76 atmosphere (U.S. Standard Atmosphere 1976 model), and no wind) were considered appropriate for this step, with a Runge–Kutta–Fehlberg 45 integrator used to simulate the launcher trajectory from the launch to the orbit insertion. Parameterized pitch and yaw constitute the control parameters, and a set of standard guidance laws (vertical launch, linear pitchover, target inclination, gravity turn, bilinear tangent law, plus a final circularization burn) defines a first guess for the optimization. For LP, the throttle level at the control nodes can be added to the optimization problem. Three simple thrust profiles can instead be defined for SP motors (constant, linearly decreasing, or two-level thrust) to match the dynamic pressure and axial acceleration constraints. Additional models are included to account for the propulsion performance variation with altitude, in-flight ignitions, and path constraints evaluation (heat flux, axial and lateral accelerations, dynamic pressure, static controllability, and geographic heading). More details about the trajectory models and their validation can be found in [19].

An accurate and robust evaluation of the ELV's P performance is particularly critical for MDO since a fair comparison of different concepts can only be ensured if errors in the payload assessment are small. A separate work [32] describes in detail the trajectory optimization process, with a focus on the specific aspects that ensure a robust performance assessment with both global and local algorithms.

G. Costs and Reliability

Particular attention was paid to nonperformance disciplines, which are often neglected in MDO studies but provide driving criteria in today's design-to-cost and design-to-reliability approaches. Hence, the cost and reliability models were implemented in order to estimate the LCC and MSP of ELVs to be used as MDO objectives together with classical performance criteria such as GTW minimization or P maximization.

The LCC is estimated through cost estimation relationships (CERs) from the transparent TRANSCOST model [33], building up the cost breakdown structure shown in Fig. 5. The TRANSCOST approach was adapted to fully reflect all technological tradeoffs defined by the selected optimization variables and was complemented with additional CERs and tunings internally available at ESA. The MSP of an ELV is instead computed as a function of mission time, with a cumulative reliability profile in each mission phase computed from the multiplication of exponential functions for the active components. Failure rates with values available at ESA define the decay rate for each component and each phase, with a risk breakdown structure shown in Fig. 6. The resulting overall profile shows exponential branches corresponding to different mission phases (ground storage, takeoff, boosters, and stages flights), connected by discrete events (separations and payload release) that are modeled as instantaneous drops in reliability.

IV. Conceptual Modeling Validation

One of MDO's main challenges is the verification and especially validation of the overall software system. Verification requires the execution of all mathematical routines and all branches of the engineering analysis, first standing alone and then combined in MDA and MDO. Although time consuming, this process is not particularly critical and is not reported here. In fact, the real challenge lies in the validation, intended as the quantitative assessment of both the MDA's suitability to represent the actual system design and of the MDO's capability to improve such a design. Moreover, the validation procedure also has the goal of identifying the most relevant weaknesses of the multidisciplinary model in order to improve the analysis cycle in further steps.

In light of this criticality, a complex validation procedure was set up, consisting of four parts. First, each disciplinary analysis was validated standing alone, using known data for existing subsystems (e.g., engines and structures) to determine the expectable errors on the main outputs of each discipline. Second, sensitivity analyses (SA) were run starting from the disciplinary errors with the aim of both identifying the most critical disciplines and of statistically estimating the expectable global error. The optimized P mass was assumed as a global performance figure, with two European launchers [Ariane-5 Evolution Cryotechnique type A (ECA) and Vettore Europeo di Generazione Avanzata (VEGA)] taken as test cases. Third, for the same vehicles, full MDA processes including P optimization were aimed at verifying the capability of the models to correctly assess the performance, cost, and reliability of ELVs on real-world scenarios. Fourth, small MDO problems were defined for Ariane-5 ECA and VEGA, with the goal of verifying the optimization's capability to improve ELVs' designs in simplified MDO cases. A summary of the results from this validation process is reported in the next paragraphs. Although any kind of unmanned ELV with classical cylindrical

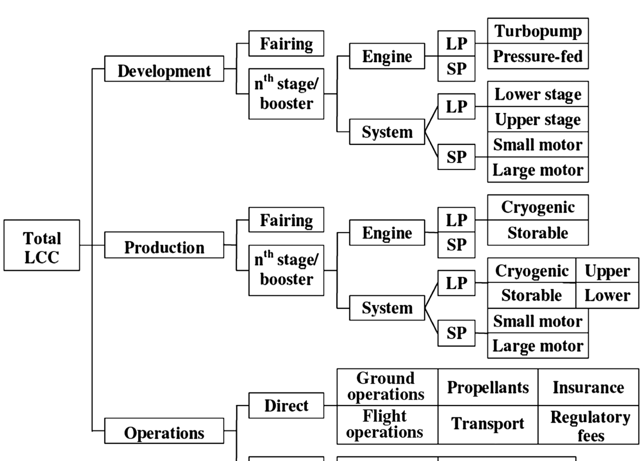

Fig. 5 CBS developed for the estimation of the total LCC of ELVs.

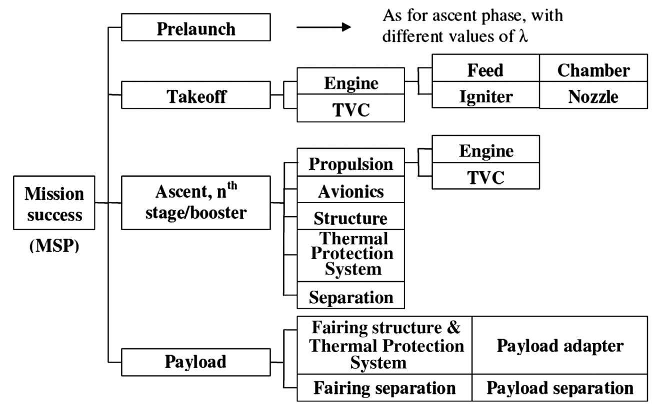

Fig. 6 Risk breakdown structure developed for the estimation of the total MSP of ELVs .

Table 1 Statistical figures of the errors in the estimation of the most relevant disciplinary output parameters

Discipline	Parameter	Description	E, %	M, %	μ, %	σ, %
Propulsion	$I_{\mathrm{sp,vac}}$, s	Vacuum specific impulse	1.02	3.09	−0.59	1.27
Propulsion	A_e, m^2	Nozzle exhaust area [for I_{sp}(h)]	14.03	31.19	−0.85	15.03
Aerodynamics	C_D	Drag coefficient	9.35	81.80	4.28	9.27
Aerodynamics	C_L	Lift coefficient	10.40	98.47	9.10	14.27
Weights	$M_{\mathrm{inert},SP,BS}$, kg	SP boosters total inert mass	10.64	21.09	−0.04	13.50
Weights	$M_{\mathrm{inert},SP,LS}$, kg	SP lower stages total inert mass	22.46	36.06	8.31	16.07
Weights	$M_{\mathrm{inert},LP,LS}$, kg	LP lower stages total inert mass	10.06	37.60	5.63	13.47
Weights	$M_{\mathrm{inert},LP,US}$, kg	LP upper stages total inert mass	10.02	21.24	−3.30	14.18
Weights	M_{PF}, kg	Payload fairing mass	15.04	33.62	−8.68	16.40

NOTE: LS, Lower stage; US, Upper stage; BS, Boosters set.

stages and boosters configurations can be tackled with the developed models, results are shown here only for Ariane-5 and VEGA, for which rather detailed design information was available for comparison.

A. Disciplinary Models Validation

To assess the accuracy of the disciplinary analyses of the DSM of Fig. 3, stand-alone executions were repeated for sets of existing components for which the input/output data were collected from publicly available sources, determining the average [$E = \mathrm{mean}(|e|)$], maximum [$M = \max(|e|)$], mean [$\mu = \mathrm{mean}(e)$], and standard deviation [$\sigma = \mathrm{stdev}(e)$] of the errors on relevant output parameters. During this phase, several tuning parameters specific to each disciplinary model were also calibrated, such as specific impulse loss coefficients or WER and CER parameters. The complete results are too extensive to be shown here, but full details are available in Castellini's Ph.D. dissertation [34]. A qualitative overview of the disciplinary accuracies is, however, given in the following text, with summarizing figures reported in Table 1. Several important variables are not included in the table (e.g., propellant mass, thrust, etc.) since they are treated as inputs given by the user or the optimizer.

1. Propulsion

The validation was performed against the database of LREs used also for OTS selection, with the addition of several SRMs. Through the experimental calibration of the specific impulse losses, extremely good accuracies on the vacuum I_{sp} could be reached (∼1% average error), whereas larger discrepancies cannot be avoided on other parameters such as the estimated nozzle's exhaust area and inert mass (∼14 and 19% average errors).

2. Aerodynamics

The validation against ESA's databases for VEGA and Ariane-5 showed average errors in the 15% range for C_D and 20–25% for C_L and C_m, even though errors as high as 100% occur for several flight conditions. Although not critical in terms of global performance, as justified in Sec. IV.B, such discrepancies are inevitable with a DATCOM-based approach.

3. Weights

Detailed weight breakdown structures (WBSs) of ELVs are not easily available; hence, only the total inert mass of stages/boosters was used as a validation figure, showing most errors being in the 10–25% range.

4. Trajectory

The validation of the dynamics integration and guidance strategies performed against ASTOS[‡‡] resulted in negligible errors on simulated trajectories. However, trajectory optimization indicated that the developed 3 DOF ascent trajectory description tends to overestimate the payload performance. In particular, Ariane-5 ECA's maximum payload to standard Geostationary Transfer Orbit (GTO) is assessed at 10,944 kg (+8.9% with respect to the reference 10,050 kg from the launch vehicle's user manual[§§]), and VEGA's payload to 700 km polar Low Earth Orbit is of 1715 kg (+14.3% with respect to the reference 1500 kg from manual[¶¶]). Although these discrepancies may be partially due to uncertainties in the launcher parameters (inert masses, aerodynamic coefficients, specific impulses, and exhaust areas), several modeling weaknesses were identified: a) the roughness of SP thrust model, only approximately following actual motors profiles; b) the lack of steering losses due to aerodynamic moment compensation, maneuvers, or wind; c) the simplicity of the controllability verification models, neglecting dynamic effects or wind; and d) overestimation of the I_{sp}, which is assumed constant in spite of the degradations occurring due to throat erosion or pressure variations.

5. Costs and Reliability

The cost and reliability models were validated against both European and non-European ELVs, even though only global figures on the overall launch cost and launch success probability could be retrieved. In particular, the CERs were shown to generically overestimate the launch cost of Ariane-5, VEGA, Soyuz, Delta, and Falcon families, although their reciprocal ranking is well predicted. Similarly, MSPs obtained with SVAGO are in line with the historical failure rates for Ariane-5, Soyuz, and Delta, whereas VEGA's foreseen reliability (ESA estimate) is accurately matched. Finally, the reliability advantages coming from features such as Falcon's engine-out capability or CCB configurations are clearly reflected in the model results.

B. Sensitivity Analyses

To evaluate the suitability of the engineering models for the conceptual design of ELVs, discipline-level accuracies are not sufficient since the different errors combine to determine the overall error on the global performance of the vehicle. For this reason, a detailed analysis of the sensitivity of the global performance to the disciplinary errors is the next logical validation step. Only the global performance measured in terms of optimized P mass was considered since no realistic terms of comparison for cost and reliability of launchers exist. Local trajectory optimizations with WORHP were therefore executed for Ariane-5 ECA and VEGA, perturbing the main disciplinary output parameters with respect to the actual launcher design by percentages reflecting the disciplinary errors in Table 1. Two types of SA were considered with different objectives. First, one-variable-at-a-time analyses were considered, in which only one parameter was perturbed by a percentage equal to $\pm E$, with the goal of identifying the critical discipline(s) that is (are) most likely to determine the largest errors in global performance, on the basis of both the parameters' relevance and modeling accuracy. Then, Monte Carlo analyses were considered, in which all parameters were

[‡‡]Information available online at http://www.astos.de [retrieved 29 July 2013].

[§§]Arianespace, "Ariane-5 User's Manual", Issue 5, Revision 1, July 2011.

[¶¶]Arianespace, "VEGA User's Manual", Issue 3, Revision 0, March 2006.

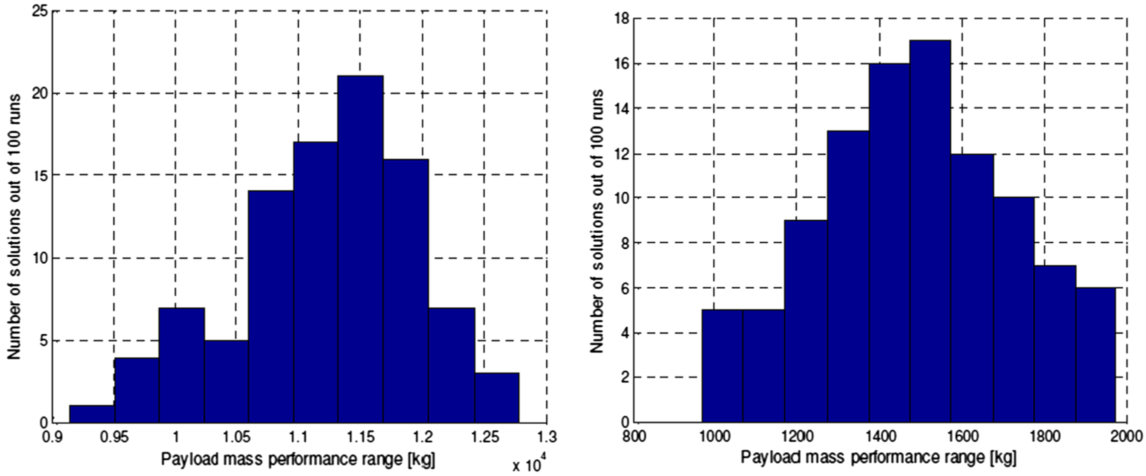

Fig. 7 Monte Carlo SA results: *P* performance distributions for Ariane-5 ECA (left) and VEGA (right).

simultaneously varied according to a Gaussian (μ, σ) distribution, with the goal of obtaining payload performance distributions (μ_P, σ_P) representative of the system-level accuracy of the models. From these, in fact, the probable bias of the models toward higher or lower payloads can be determined (μ_P) as well as the even more important $1\sigma_P$ or $3\sigma_P$ ranges, which should represent the deviations in performance that can be expected to be due to modeling errors rather than to actual design changes.

The results of the one-variable-at-a-time analyses showed that the most critical discipline is by far the weights analysis, resulting in P errors up to $\pm 10\%$ when varying the inert mass of the upper stage. Although the sensitivity to other stages/boosters was found to be much more limited (e.g., for Ariane-5, a payload reduction of 1 kg results from a mass growth of the booster or core of 8.3 or 2.6 kg, respectively, against the 1 : 1 ratio for the upper stage), this suggested to invest significant effort in the improvement of the structural models for the second modeling step. On the contrary, the propulsion and aerodynamics resulted to be accurate enough for the purpose of conceptual and even early preliminary design, with the largest P errors in the order of 3 and 1%, respectively. This is due to the high accuracy achieved in the LP/SP engines performance analysis and the very low sensitivity of P to C_D and C_L.

Monte Carlo SA results are visually presented in Fig. 7, showing P distributions for 100 Monte Carlo runs. The numerical values μ_P and σ_P are compared in Table 2 to the reference mass from manual and the estimated performance with all launcher data fixed to the actual figures (i.e., only trajectory optimization). In addition to the already mentioned bias of the trajectory models, a tendency toward overestimating Ariane's performance is clear, whereas a pessimistic evaluation of VEGA's design balances the bias from the trajectory. This different behavior can be again traced back mainly to the weight models, since VEGA's modern structures and nozzle design result in low structural ratios, not reflected in the implemented WERs, developed from more traditional launchers. In regard to the standard deviations, $\sigma_P = 8\%$ for Ariane-5 and $\sigma_P = 16\%$ for VEGA are reasonable figures for expected 1σ launcher performance errors in a conceptual-level design environment employing such simplified engineering models. As a final note, μ_P and σ_P combine in P distribution ranges for the two launchers with a worst-case 1σ error of $+19\%$.

C. Multidisciplinary Design Analysis

As a further step of the validation process, the MDAs of European launchers were aimed at comparing the performance, cost, and reliability provided by the implemented design cycle to the actual values. Although the results are too lengthy to be shown here (see [34] for all the details), a few remarks are provided that are then directly used as inputs for the model enhancements described in the next section. On a disciplinary level, the MDA showed that the geometry calculations sensibly underestimate the lengths, especially for VEGA interstages, with nonnegligible effects on the weights. Moreover, although the GTWs of both Ariane and VEGA are rather accurately estimated, the actual WBSs are not well matched, confirming the impression that weights models should be improved. In particular, an almost -50% error on Ariane's upper stage is the major cause for a sensible 24.1% overestimation of its P performance with respect to the user manual's value, representing a further 1.5 tons P increase with respect to the figure obtained with frozen launcher parameters. The MDA instead sets VEGA's payload mass to -6.5% with respect to the reference by manual, a drastic reduction (315 kg) with respect to the fixed design P assessment, again mostly explained by the errors in inert masses, partially offset by a sensible overestimation of the I_{sp} of the Zefiro-9 third stage.

The actual cost breakdown structure (CBS) and reliability data were not available for either launcher; hence, a quantitative validation of nonperformance disciplines was not possible. However, the CPL of 172 and 35 M€ in fiscal year (FY) 2009 for Ariane-5 ECA and VEGA, respectively, are comparable to the current advertised launch prices, and the CBSs appear reasonable, as shown, for example, for VEGA in Table 3. The assessed MSP, shown to be rather sensible for VEGA in Fig. 8, is instead less accurately reproduced by the models for Ariane. In fact, its reliability profile is driven by the cryogenic propulsion's high failure rate, which comes from historical evaluations, determining a low 92.7% probability of mission success, definitely pessimistic if compared to the remarkable 32 successes out of 33 ECA version launches.

D. Multidisciplinary Design Optimization

The last validation step is constituted by the full MDO process, which was tested with the purpose of assessing the capability of the

Table 2 Ariane-5 ECA and VEGA's *P* masses from manual, fixed design, and Monte Carlo distributions

	Ariane-5 ECA		VEGA	
Payload mass from manual, kg	10,050	— —	1500	— —
Payload mass for fixed actual design, kg	10,944	+8.9%	1715	+14.3%
Payload mass distribution mean value μ_P, kg	11,217	+11.6%	1488	−0.2%
Payload mass distribution stdev σ_P, kg	762	7.6%	239	+15.9%
Payload mass expectable ($1\sigma_P$) range, kg	[10,455;11,979]	[+4.0; +19.2]%	[1249;1727]	[−15.7; +16.1]%

Table 3 CBS from VEGA MDA (all costs in FY 2009 M€), assuming 120 launches in 20 years

	Development system	Development engine	Production system	Production engine
PLF	25.8	0	— —	0.7
P80	177.1	0	— —	8.1
Z23	34.8	0	— —	3.1
Z9	26.9	0	— —	2.2
AVUM	113.9	114.8	— —	3.6
Operations	Ground	Propellant	Flight	Other
Per flight	8.1	0.1	1.6	1.1
LCC	Development	Production	Operations	Total
Total	740.0	2132.6	1300.9	4173.5
Per flight	6.2	17.7	10.8	34.8
k€/kg	4.1	11.8	7.2	23.2

optimization to produce design improvements. To both simplify the task of the optimizer and allow a more straightforward interpretation of the achieved solutions, the investigated problems were kept small, fixing the P mass to the values by manual[§§] (10,050 kg and 1500 kg for Ariane-5 ECA and VEGA, respectively) and minimizing the GTW with the SO-PSO. All technological and architectural tradeoff variables as well as the propulsion systems designs were thus frozen in the MDO setup, with a ±30% allowed range for propellant masses, length-over-diameter ratios, and trajectory loads. The resulting test MDO problems foresee 7 and 9 design optimization variables for Ariane and VEGA, respectively, with $X_{\mathrm{opt,A5ECA}} = \{M_{\mathrm{prop,EPC}}, M_{\mathrm{prop,ESC\text{-}A}}, M_{\mathrm{prop,P241}}, (L/D)_{\mathrm{P241}}, q_{\mathrm{dyn,max}}, q_{\mathrm{heat,max}}, n_{\mathrm{ax,max}}\}$ and $X_{\mathrm{opt,VEGA}} = \{M_{\mathrm{prop,P80}}, M_{\mathrm{prop,Z23}}, M_{\mathrm{prop,Z9}}, M_{\mathrm{prop,AVUM}}, (L/D)_{\mathrm{P80}}, (L/D)_{\mathrm{Z9}}, q_{\mathrm{dyn,max}}, q_{\mathrm{heat,max}}, n_{\mathrm{ax,max}}\}$.

The convergence histories for multiple PSO runs, accounting for stochastic effects, are shown in Fig. 9, with satisfactory convergence reached after 1000 iterations (less than 3% difference among the three runs). In both cases, MDO is capable of sensibly reducing the GTW of an existing launcher with respect to the real-world figures. In particular, after local refinements with WORHP allowing for an additional improvement of 1–2%, GTWs of 506 and 132 tons were obtained for Ariane-5 ECA and VEGA (−34.0 and −4.6% with respect to the real-world launch masses), respectively. Nonetheless, it is important to point out that this reduction is not directly corresponding to the real design improvement brought by MDO. The GTW should, in fact, be compared to the one assessed by the MDA models for an equal payload to correct for the errors in the multidisciplinary models. Since the P estimated by the MDA is different from the one in the manual (see the previous section), the GTW from the MDA needs to be scaled through an approximate linear sensitivity of the launch mass to the payload mass. This was derived from multi-objective MDO runs for the minimum GTW and maximum P, from which a linear regression can be obtained as an estimate of $\partial P/\partial \mathrm{GTW}$, shown in Fig. 10. Using these approximate derivatives (~17 and ~19 kg/ton for Ariane and VEGA, respectively) to scale the GTW from MDA for a comparison to the GTW from MDOs, the actual improvement in takeoff mass brought

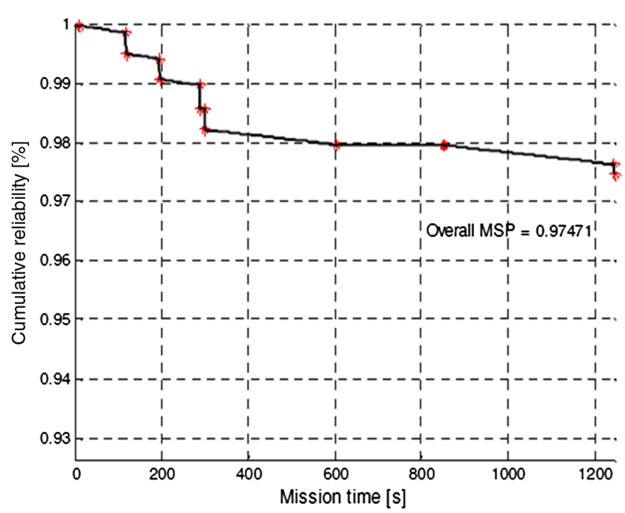

Fig. 8 Cumulative reliability over time for VEGA.

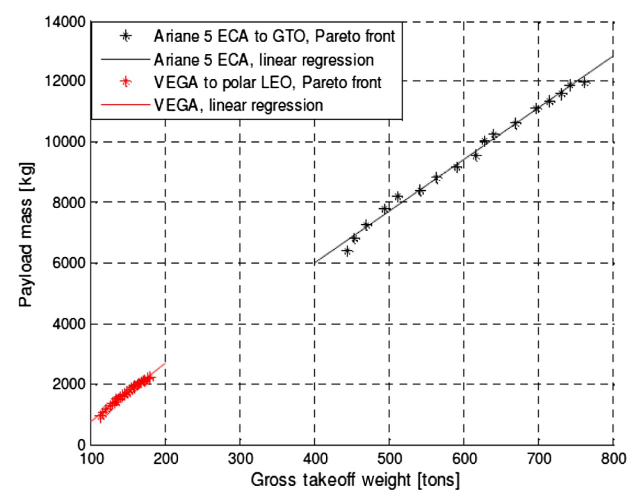

Fig. 10 GTW vs P multi-objective Pareto fronts for Ariane-5 ECA and VEGA test MDO problems.

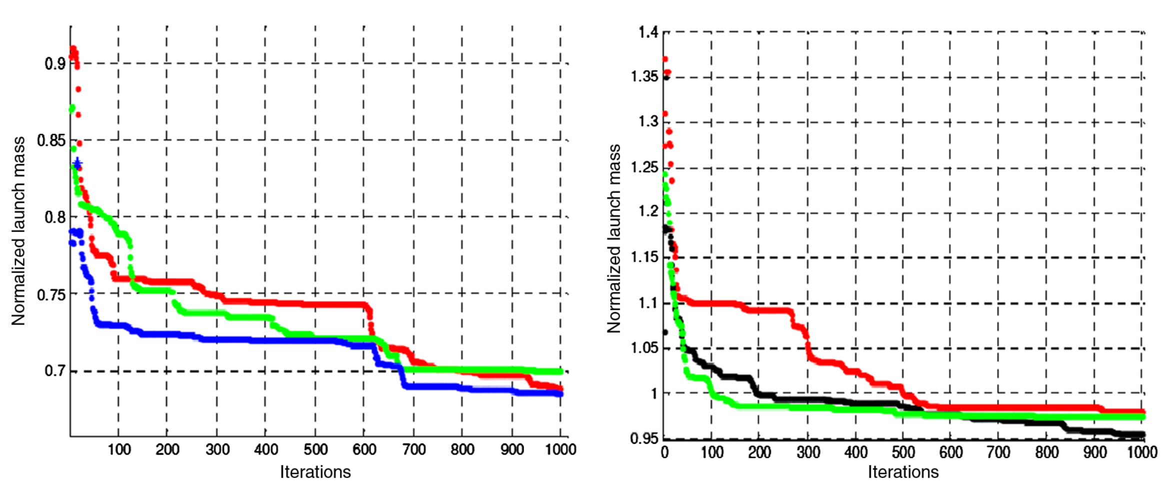

Fig. 9 Convergence of the SO-PSO over three runs for the MDO test problems of Ariane-5 (left) and VEGA (right).

by the MDO methodology results to be of ~20% for Ariane and ~9% for VEGA.

These enhancements are mainly obtained through the reallocation of the propellant among stages/boosters and the reduction of the trajectory loads. The propellant mass changes reflect well the involved physical tradeoffs: As expectable, Ariane's design appears to be less "staging optimal" than VEGA's, with the optimizer reducing its SP boosters and increasing the loading of the upper stage. However, the reduction of n_{ax} and q_{dyn} appears to be more a modeling artifact rather than a physical reality since no accurate structural analysis is performed to validate the resulting decrease in the dry mass.

V. Multidisciplinary Modeling for Early Preliminary Design of ELVs

The early preliminary MDO environment was obtained through upgrades in all disciplines, derived from a critical analysis of the validation results of Sec. IV, complemented by ESA reviewer's comments. The following paragraphs present an overview of the main modeling enhancements, aimed either at improving the accuracy of the models or adding specific functionalities that were deemed necessary for MDO's industrial applicability.

A. System Level

The DSM describing the MDA remains rather similar to that of the conceptual models. Most of the vectors $\boldsymbol{P}_j$, $\boldsymbol{X}_j$, $\boldsymbol{X}_{jk}$, and $\boldsymbol{Y}_j$ include larger numbers of parameters, reflecting the model enhancements described in this section, but the basic structure remains unchanged. The only major difference is constituted by the introduction in the design cycle of a structural analysis module, as represented in Fig. 11. This requires iterations with the trajectory block, for which loads are generated, to achieve the convergence on the inert masses of the stages and boosters.

Besides the multidisciplinary feasible (MDF) problem formulation, i.e., trajectory-structures iterations to achieve feasibility for each solution, the individual disciplines feasible (IDF) approach was also tested (see [35] for a classification of problem formulations). The IDF approach involves introducing one optimization variable and one constraint for each stage and boosters set. These slack variables represent the inert masses for trajectory integration, and the compatibility constraints define the consistencies of these values with those computed within the structural/weight models. The iterations of the design cycle are therefore no longer needed. However, the consistency of the design is not ensured for each solution, but only at optimization convergence. As a consequence, the IDF formulation has the advantage of cutting the CPU time for a single MDA at the cost of a slower convergence of the optimization process. Although

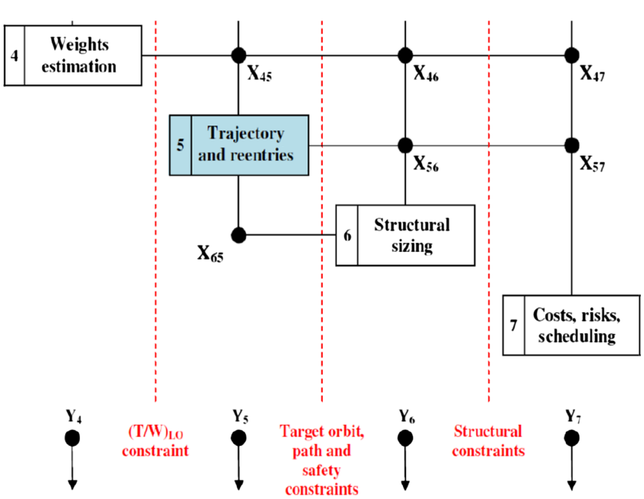

Fig. 11 Lower part of the DSM for ELVs early preliminary design, with the upper part identical to Fig. 3.

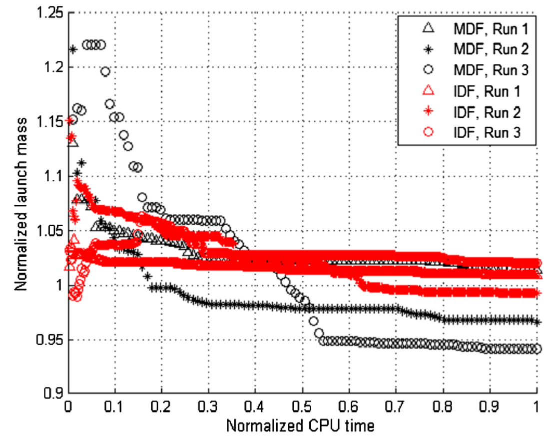

Fig. 12 Comparison of MDF and IDF formulations: SO-PSO algorithm convergence for the Ariane-5 problem.

the literature often reports drastic advantages of the IDF approach over the MDF approach in terms of overall optimization times (e.g., [36,37]), it was found in this study that the performances of the two approaches are very similar for this specific problem. In fact, the required number of trajectory-structures iterations is typically low (2 to 4 are sufficient for both Ariane and VEGA test cases); hence, the MDF formulation overall appears slightly superior. The convergence histories for the Ariane-5 ECA small MDO problems are shown as an example in Fig. 12.

B. Propulsion

From the analysis of the conceptual models, two key weaknesses were identified in the areas of the SP and LP: the lack of realistic thrust profiles for SRMs and of an analysis of the pressurization system connected to both tanks pressure and pumps cavitation conditions. These two aspects are described in more details in the following text, whereas numerous secondary enhancements to the propulsion models are reported in [34], covering both added functionalities (more fuels such as Russian kerosene vs rocket propellant one (RP-1), multiple thrust chambers, and extendable and submerged nozzles) and an improved accuracy (I_{sp} loss for SP engines as a function of the expansion ratio and throat erosion, the detailed assessment of unused LP mass including an end-of-life propellant budget, and better tuning of the WERs).

1. Solid Rocket Motors Design

Two upgrades were targeted at modeling realistic SRMs thrust profiles: the OTS database was extended to include SRMs with predefined vacuum thrust and I_{sp} profiles (e.g., European P-241, P-80, Z-23, Z-9, and several US motors from ATK), and models to describe the solid grain geometry were introduced with the purpose of determining the resulting thrust profile and analyzing the internal ballistics to derive the chamber pressure and I_{sp} variation during the burn. Hence, for new design SRMs, three options can be selected. First, custom motors with filling factor (FF), sliver fraction (SF), normalized thrust and I_{sp} profiles given by the user, and scalable maximum thrust and propellant mass can be designed. Custom motors are used to represent modifications of existing motors, for which no geometric design or internal ballistic analyses are performed since all characteristic are assumed unchanged. Second, end-burning motors can be used for upper stage or kick motors, with a constant burn area A_b, internal pressure p_{cc}, thrust T, and specific impulse I_{sp} (FF $= 1$ and SF $= 0$). Third, internal-burning motors can be chosen for sea-level applications, for which a detailed geometrical design of the SP grain for realistic $A_b/p_{\mathrm{cc}}/T/I_{\mathrm{sp}}$ profiles representation is allowed, even though ignition and burnout transitories are neglected.

For the latter option, up to five segments with different grain cross sections and relative lengths can be specified, each with one of three

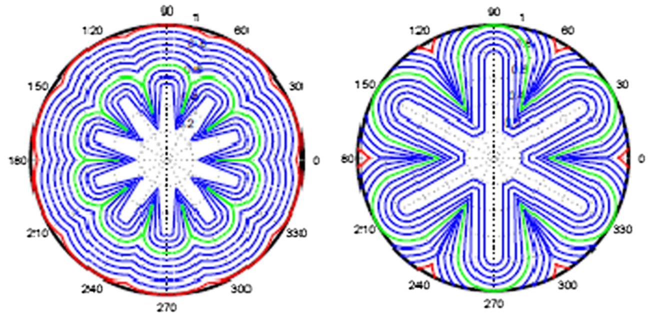

Fig. 13 Examples of star (left) and slot (right) grain geometries with a burnback diagram (values in degrees).

internal perforation types: 1) tube grain, 2) slot grain, or 3) star grain. Ignition buildup is assumed instantaneous, and thrust varies according to the grain burn area until the sliver is reached. The equations for the burn perimeter and port area are taken from literature [38,39], with the web fraction (WF) and additional slot/star measures as geometric parameters. Typical geometries are shown in Fig. 13, including the switch between phases I and II of the burnback equations. The geometric analysis is a part of the SRMs design procedure, which consists of several steps. First, if $N_{\mathrm{segments}} > 1$, the web length for all segments except the first is adjusted to have a constant WF throughout the grain. Then, the normalized geometric analysis is performed, with an adimensional burn area profile, SF, and FF as outputs. From these, all the dimensional parameters, including mass flow, can be obtained imposing a grain scaling as follows:

$$L_{\mathrm{grain}} = \frac{M_{\mathrm{prop,usable}}}{\pi \cdot R_{\mathrm{ext}}^2 \cdot \rho_{\mathrm{grain}} \cdot \mathrm{FF} \cdot (1 - \mathrm{SF})} \tag{1}$$

CEA analysis, I_{sp} losses, and minimum altitude evaluation follow as in conceptual models, whereas the nozzle scaling is performed for maximum mass flow conditions through the mass balance between the grain and throat. Inert masses estimation can thus be completed on the basis of the scaled engine design, as in conceptual models, and the time profiles of p_{cc}, I_{sp}, and T can be computed for constant chamber properties and assuming an isoentropic expansion in the nozzle. The last step consists of the evaluation of the burn time consistency: the burn rate r_b is an optimization variable, which is varied by the optimizer so that the burn time obtained by integrating $T(t)$ and $I_{\mathrm{sp}}(t)$ profiles matches the one from the web length. As an example of thrust profile that can obtained with this procedure, Fig. 14 shows the calculated $T(t)$ for a two-segment tube-slot grain configuration in which five free geometric parameters were optimized to minimize the error with respect to VEGA P80's thrust profile.

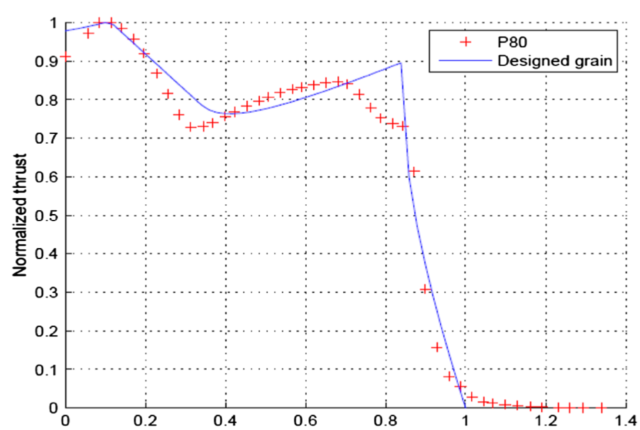

Fig. 14 Example of thrust profile obtained with one tube and one slot segment to approximate VEGA's P80 (values in degrees).

2. *Liquid Propulsion Pressurization System Sizing and Pumps Cavitation Analysis*

The objective of the pressurization and cavitation analyses is twofold: first, to improve the estimation of the propulsion system dry mass and, second, to allow for a fair evaluation of the tradeoff on the LP tanks pressure, which is particularly important since a higher pressure determines heavier tanks and a higher pressurization system at the advantage of easier suppression of the pumps cavitation. Three different pressurization types can be selected by the user (or optimized) for both the oxidizer and fuel: evaporated propellants, only feasible for LO_x or LH_2; heated helium, and stored helium. In the former case, high-pressure He is stored in cryogenic tanks and heated up through exchangers in the turbines discharge or nozzle. This allows reducing the mass of both the gas and its tank, due to the much higher density of cryogenic He and, hence, lowers the tank's volume and mass. In the latter, the high-pressure He is directly expanded from an ambient temperature tank, ensuring maximum simplicity at the cost of a larger mass and therefore being well suited to smaller pressure-fed upper stages (e.g., VEGA AVUM).

Two different models were implemented for the estimation of the pressurization gas mass: an ideal gas law applied to the final ullage conditions and an energy conservation approach considering the adiabatic expansion of the pressurization gas. Through calibration of the relevant parameters on Saturn V, the space shuttle, Ariane-5, and VEGA stages, the first method was chosen for stored He and evaporated propellants and the second for heated He. Pressure vessel sizing relations are instead used for the tanks mass estimation, with different He densities and materials characteristics for heated vs stored systems. In spite of the simplicity of these models, the available figures for the total pressurization system mass of the considered stages are reasonably well matched, with average and maximum errors lower than 10 and 20%, respectively.

Although a full engine cycle analysis and component-level sizing (gas generators, turbomachinery, etc.) were deemed too complex for this level of fidelity, a pumps cavitation analysis was implemented in order to measure the effects of the tanks pressure in pump-fed systems. If, depending on the values of p_{tanks}, p_{cc}, gravity/acceleration head rise, and pressure losses, cavitation is detected, the mass of the required boost turbopumps is estimated through a quadratic regression obtained from NASA literature [40] and is then added to the engine inert mass. The analysis procedure to detect cavitation, derived from the standard bibliography [27,28,40], consists of several steps. First, the friction losses from tank to pump and the head rise due to tank elevation are evaluated, allowing one to derive the pump suction pressure and, hence, the net pump suction head (NPSH) as

$$\mathrm{NPSH} = (p_{\mathrm{suction}} - p_{\mathrm{vapor}})/(\rho \cdot g_0) \tag{2}$$

The pump discharge pressure can instead be computed from p_{cc}, with different pressure losses depending on engine cycle and nozzle cooling, either regenerative or ablative. The pump head rise (PHR) and required pump suction head (RPSH) are therefore obtained as

$$\mathrm{PHR} = (p_{\mathrm{discharge}} - p_{\mathrm{suction}})/(\rho \cdot g_0) \tag{3}$$

$$\mathrm{RPSH} = \left(21.2 \cdot N \cdot \sqrt{\dot{V}}/S\right)^{4/3} \quad \text{with} \quad N = N_S \cdot \mathrm{PHR}^{0.75}/21.2 \cdot \sqrt{\dot{V}} \tag{4}$$

where $\dot{V}$ is the volumetric flow, the pump specific speed N_S and suction specific speed S are taken from the literature, and all data are in British units. Finally, the pumps are considered to be cavitating when

$$\mathrm{RPSH} \geq \mathrm{CM} \cdot \mathrm{NPSH} \tag{5}$$

for which a cavitation margin (CM) equal to 0.8 corresponds to requiring for proper pump functioning a net suction head 20% higher than the one causing cavitation. Again, in spite of its simplicity, the cavitation model is able to correctly predict the need for a boost system in 27 out of 29 tested liquid rocket engines. The application of the boost turbopump assembly mass regression for current technology is, however, questionable since the employed historical data from [40] are referred to engines designed before 1975. More recent data could not be retrieved.

C. Geometry and Aerodynamics

The aerodynamic models were not modified, except for the introduction of multiple aerodynamic configurations (e.g., with and without boosters), due to the small sensitivity of the global performance to errors in this discipline. Three important geometric functionalities were, however, introduced. First, the separation plane between stages was modeled through a minimum nozzle disengagement angle of 15 deg, which ensures the minimization of the mass staying on the upper stage and a reasonable margin to avoid dangerous nozzle/interstage contacts during separation. Second, the option of the underfairing configuration was defined for small upper stages in order to improve the weight estimation for components such as VEGA's AVUM. Third, tanks geometries such as enclosed tanks (e.g., Ariane-5's ESC-A) and multiple tanks (e.g., Soyuz's Fregat, Proton's Breeze, and VEGA's AVUM) were added, again with the aim of enhancing the accuracy in the inert mass estimation of the upper stages.

D. Trajectory

The most relevant weaknesses highlighted from the validation of the conceptual models are the lack of wind and of steering losses associated with aerodynamic moment compensation. A full wind trajectory analysis (3σ dispersion of insertion error and control system dynamic verification) was considered much too detailed; hence, a simple wind model based on lookup tables from NASA's Handbook on Terrestrial Environment [41] was implemented. Steady-state horizontal wind is then used at each altitude to compute the AOA under worst-case wind for static controllability verification. A synthetic wind profile constructed from steady-state wind, wind shears, and wind gusts is instead used for the evaluation of the wind-related steering losses, to be integrated along the trajectory. Both profiles are reported in Fig. 15, whereas Fig. 16 shows the thrust vector control's (TVC) deflection angle computed for a typical Ariane-5 flight from the lateral thrust necessary to both balance the aerodynamic moment and provide the angular acceleration for pitch/yaw maneuvers. Although maneuvers losses are minor, the overall ΔV introduced by the new models is not negligible, with an increase of around 30 and 45 m/s of steering ΔV for Ariane-5 and VEGA.

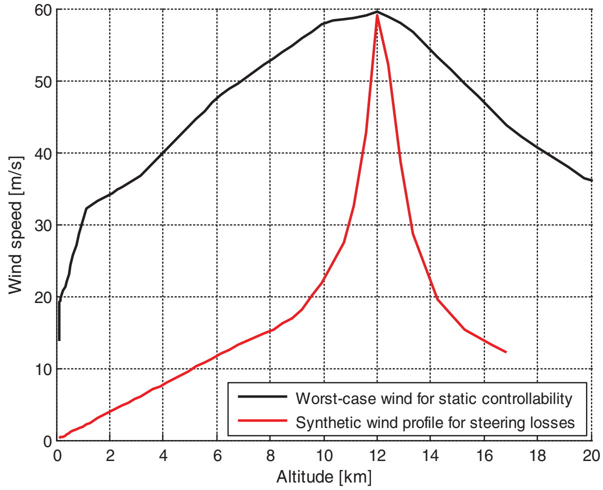

Fig. 15 Worst-case steady-state profile and synthetic wind profile for launches from Kourou.

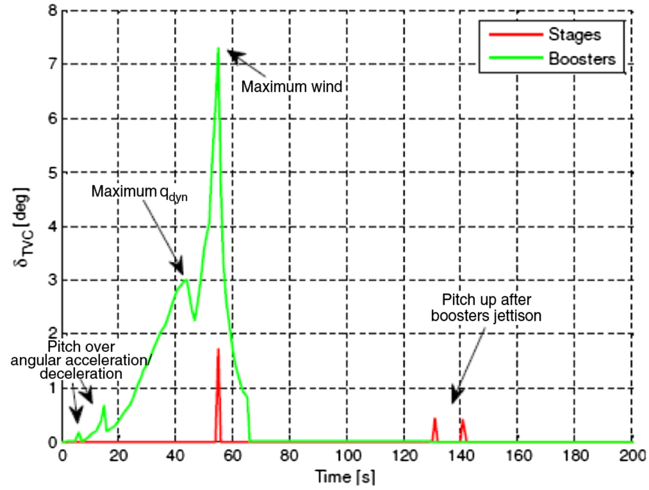

Fig. 16 Required TVC angle for a typical Ariane-5 flight, including both aerodynamic moments and maneuvers.

E. Safety

A critical industrial need lacking in the conceptual design environment is the analysis of reentry trajectories for all suborbital components of ELVs since safety requirements prevent inhabited regions to be within the expected ground impact ellipses. This may represent a severely constraining factor for several launch sites and target orbits, sensibly affecting the performance of the launch vehicle. Two different models were thus implemented for assessing the impact ellipse: 1) the integration of ballistic 3 DOF equations from the jettison instant and 2) the propagation of the Keplerian parameters complemented with empirical models to estimate the downrange reduction due to drag.

In both cases, to avoid the computationally expensive Monte Carlo approach, only two trajectories are simulated with minimum-drag and maximum-drag ballistic coefficients, and a constant minor-to-major axis ratio equal to 0.3 is assumed to determine an approximate impact ellipse. Finally, an Earth population density map*** is overlapped, computing the average density on the ellipse to be used to impose the safety constraint, which is set by default to 0 persons/km^2. This simple safety model was verified for an Ariane-5 flight to a 6 deg GTO, for which optimal trajectories of the conceptual models involve a core stage's impact point over populated central Africa. Figure 17 shows how the introduction of the safety constraint leads to an adjustment of the trajectory to allow for an impact ellipse either in the Gulf of Guinea or, if an additional uncertainty margin on the ellipse is assumed, west of Liberia. The first option results in a very limited ~20 kg payload loss, whereas a ~680 kg penalty needs to be paid for the safer second option.

F. Structural Analysis

In light of the criticality of the weights assessment in the conceptual models, a structural analysis and sizing module was introduced with the purpose of better estimating the mass of all structural components: SRM cases, liquid tanks, skirts, intertanks and interstages, thrust frames, payload adapters and fairings, and booster nose ogives. The model was derived from a recent work [42] in which a beam approximation was applied to ELV structures, which was complemented by classical structural analysis practices from Ref. [43]. Although much simplifying the physics of the problem, the implemented procedure results are rather complex. First, the load cases from the trajectory are identified (e.g., on-pad, takeoff,

***Data available online at Gridded Population of the World, version 3 (GPWv3), http://sedac.ciesin.columbia.edu/gpw/ [retrieved 29 July 2013].

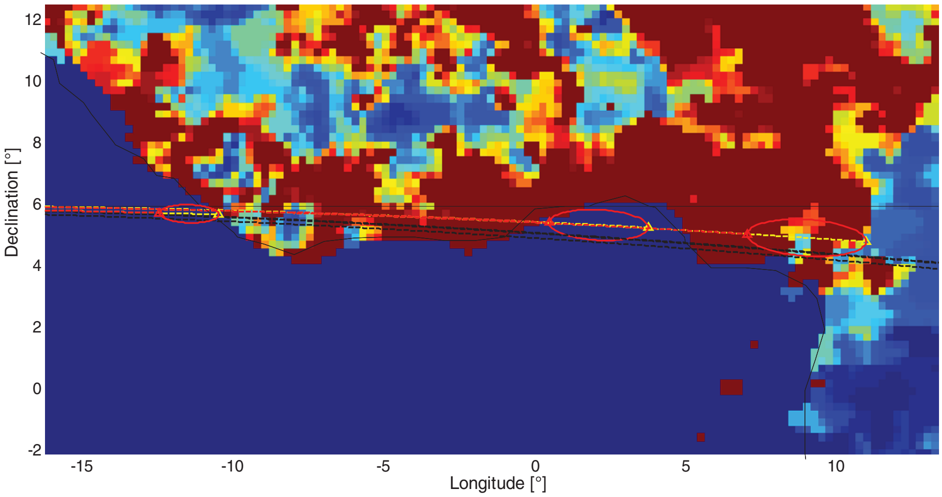

Fig. 17 Safety constraint activation for the reentry of Ariane-5's core stage: optimal trajectories without (right ellipse) and with (center and left ellipses) the safety constraint.

Mach = 1, max q_{dyn}, max $q_{\mathrm{dyn}} \cdot \alpha$, max n_{ax}, max n_{lat}). Then, for each of these, the mass distributions along the core/booster longitudinal axis are defined, assuming each component is constituted by a separate beam. Each mass item is described by a mass value, start and end positions (which coincide for concentrated masses), and a reaction station position in case of cantilevered items (e.g., engines, payload, and liquids). Again, for each load case, the center of gravity (COG), center of pressure, thrust application point, and longitudinal inertia are computed, allowing one to estimate the external loads along the core/booster beams from inertia, aerodynamic, and thrust loads. The considered external loads are axial force, shear force, and bending moment. For each station along the beams, the internal running loads can thus be determined, consisting only of hoop (circumferential), axial (longitudinal), and shear (transverse) stresses as obtained from the contributions of both external loads and internal tank/case pressure (ullage and head pressures). The combination of all loading conditions allows at this point to determine the worst-case running loads along the beams for each of the structural stations, which are grouped to form the different structural components. Hence, the required shell thickness to withstand such running loads can be assessed, imposing a material minimum gauge and considering three failure modes: the ultimate strength through the von Mises criterion, yield strength through the maximum principal stress criterion, and buckling through the minimum weight equation for wide column stiffened shells as derived by Crawford and Burns in [44]. To prevent general instability, different shell configurations are associated to longitudinal frames, for which the smeared thickness determined from Shanley's criterion [45] is summed to the shell thickness. Finally, the primary structural mass of all components can be assessed by a simple integration of the calculated thickness along the beam.

This procedure, for which more details are provided in [34], is repeated for the core and all booster sets included in the architecture, accounting for the load transfer at the predefined attachment points. This allows an estimate of the optimal structural weight of all structural components. However, the approach of an analysis based exclusively on fundamental structural principles prevents one from considering nonoptimum weights such as bulkheads, minor frames, coverings, fasteners, and joints. Hence, structural weights are generally underestimated, and correlations to existing vehicles were implemented to correct for this inaccuracy. With this model, two additional optimization variables are introduced for each structural component: the material and stiffening concepts. The former can be chosen among the Al 7075 alloy, Ti 6Al-4V alloy, 4340 steel, Al-Li 2195 alloy, and C epoxy. For the latter instead, a simple integrally stiffened shell, Z-stiffened shells and a truss-core sandwich design can be selected, each with different values of the buckling efficiency, Shanley equation's exponent, and minimum gauge parameter.

VI. Early Preliminary Modeling Validation

A. Disciplinary Models and Sensitivity Analyses

The validation of the early preliminary models followed the same procedure developed for the conceptual models (Sec. IV), with the goal of ensuring a fair comparison between the two modeling steps. The subsystem-level validation process showed a significant improvement in most of the disciplines, as can be evinced by the comparison of the results in Tables 1 and 4. Again, full results are detailed in [34], but it has to be remarked here how the accuracy in the assessments of propulsion and especially weight outputs was significantly increased. In terms of sensitivities, one-variable-at-a-time analyses highlighted a significantly lower criticality of the inert masses with respect to the conceptual models. In particular, the estimated errors on disciplinary outputs determine global P errors of at most ~5% for structures and weights, against ~3% for propulsion and ~1% for aerodynamics. Besides, the overall lower values of σ in

Table 4 Summary of early preliminary models disciplinary validation results, to be compared with Table 1

Discipline	Parameter	Description	E, %	M, %	μ, %	σ, %
Propulsion	$I_{\mathrm{sp,vac},LP}$, s	Vacuum specific impulse, LREs	0.98	1.60	−0.02	0.71
Propulsion	$I_{\mathrm{sp,vac},SP}$, s	Vacuum specific impulse, SRMs	0.31	−0.50	−0.03	0.27
Propulsion	A_e, m^2	Nozzle exhaust area [for I_{sp}(h)]	14.63	30.11	−0.80	15.37
Aerodynamics	C_D	Drag coefficient	9.35	81.80	4.28	9.27
Aerodynamics	C_L	Lift coefficient	10.40	98.47	9.10	14.27
Weights	$M_{\mathrm{inert},LP}$, kg	LP stages total inert mass	8.13	−27.38	−2.94	8.47
Weights	$M_{\mathrm{inert},SP}$, kg	SP stages/boosters total inert mass	7.81	+16.36	+4.12	9.61
Weights	M_{PF}, kg	Payload fairing mass	6.93	+18.60	+1.33	8.55

all disciplines allow obtaining a much lower dispersion in the P performance from the Monte Carlo analyses. The results are reported in Table 5, with expectable 1σ errors that are within 12%. This is a remarkable result for an early preliminary design environment, especially considering the necessary simplifications in the analysis models. Note, however, how most of the models were tuned with available data from European launchers; hence, the application to other non-European ELVs may result in larger discrepancies.

Table 5 also shows the payload estimates obtained for Ariane-5 ECA and VEGA with the new trajectory upgrades, when fixing all design variables to the actual values. The main purpose of these enhancements was in fact the improvement of the performance assessment, which was shown to be rather optimistic with the initial conceptual-level models (9 and 14% payload overestimation for Ariane and VEGA). Mainly as a consequence of the realistic representation of SRMs thrust profiles as well as of the introduction of wind steering losses and safety constraints, new estimates were obtained, matching very closely the reference performances: 10187 kg for Ariane-5 and 1573 kg for VEGA. The residual overestimation of less than 5% could not be eliminated and may be due to possible errors in the input data (e.g., inert masses, specific impulses, etc.) and to remaining inaccuracies such as the 3 DOF nature of the dynamical model or the lack of wind impact on the nominal optimized trajectory.

B. MDA and MDO

With the wide range of improvements developed for most disciplinary areas, the testing of MDA processes proved a significantly better capability to assess the performance of existing European launchers. In particular, the lengths estimations for Ariane and VEGA stages/boosters was drastically improved due to the better representation of the interstage sections (skirts modeling), the introduction of the underfairing configuration, and a more accurate definition of the divergent nozzle angles for SRMs. This increased accuracy, allowing a better division of the masses among different stages, combined with the newly developed structural analysis module provides much more realistic WBSs for both Ariane and VEGA. Even though the assessed GTWs are farther from the actual values with respect to the conceptual models, the errors on the inert masses of the single components are largely reduced. As a consequence of these enhancements, as well as those in the propulsion and trajectory disciplines, the resulting P performances are only overestimated by 3.6 and 5.6% for the two European ELVs. Most importantly, the offset with respect to the real-world figures is on the same side for both launchers, which have a completely different architecture and design, suggesting how the developed models may be used for fair performance tradeoffs among different ELV configurations. With respect to cost and reliability, no relevant modifications were performed; hence, the MDA results are similar to those described in Sec. IV.C.

In regard to the full MDO process, the small problems for European launchers described for the conceptual models were reused. With the structural analysis directly using the full trajectory data to derive the relevant load cases, the trajectory load parameters $q_{\mathrm{dyn,max}}$, $n_{\mathrm{ax,max}}$, and $q_{\mathrm{heat,max}}$ lose significance; hence, Ariane and VEGA problems are reduced to only four and seven design variables in addition to the 10 and 14 trajectory control variables, respectively. From multi-objective runs, the values of $\partial P/\partial\mathrm{GTW}$ were assessed to be ~22 and ~25 kg/ton so that the corrected net design improvement brought by MDO results to be around 6% for Ariane and 2% for VEGA. This confirms that VEGA's design is more staging optimal than Ariane's, as already found with the conceptual models. However, the introduction of the analytical structural sizing prevents one from decreasing the inert masses through the synthetic reduction of the trajectory load parameters, which was described in Sec. IV.D. This determines the much more limited impact of MDO on the design, with GTW improvements mostly obtained through the reallocation of the propellant masses, which is a more realistic modification of the design.

C. Critical Comparison of the Two Modeling Steps

This paragraph compares conceptual and early preliminary-level models, mainly on the basis of performance estimates, as well as of the required CPU times. The MDO CPU time is intended as the single-processor time required for three global optimization runs with DG-MOPSO, each involving 50,000 MDA evaluations, and the related three local refinement processes with WORHP. All the main results from the fixed design trajectory optimizations, Monte Carlo sensitivity analyses, and MDA and MDO processes are condensed in Table 6, from which a few key considerations originate. First, the two-steps modeling process allowed the definition of a wide range of enhancements, directly stemming from the validation of the conceptual-level multidisciplinary design cycle. Several missing functionalities were therefore enabled, and sensible improvements in accuracy were achieved in most of the disciplines. For instance, the errors on $I_{\mathrm{sp,vac}}$ and M_{nozzle} for SP systems, on the length of all launcher components and on the inert mass of all stages/booster types, were significantly diminished. The incidental one-variable-at-a-time sensitivity analyses showed how the criticality of the weight models was definitely reduced with the introduction of the new structural sizing module. The inert masses remain the largest cause of inaccuracies, with the remaining global performance error due to average errors on a single parameter reduced to ~5%.

Monte Carlo runs resulted instead in narrower $1\sigma P$ variability ranges, with the maximum expectable error on the global performance lowered from ~19 to ~12% with the upgraded models. At the same time, the error on the P mass from MDA was decreased to ~4 and ~6% for Ariane-5 ECA and VEGA, respectively, with a consistently optimistic performance bias for two ELVs with different architectures. Finally, the smaller effect of the MDO on the GTW of the existing European launchers is to be seen as a good sign since the large drop in the GTW obtained with the conceptual models was at least partially synthetic in nature due to the exaggerated effect on the structural masses of the trajectory load parameters. In summary, the modifications and additions introduced in the multidisciplinary design models for ELVs significantly improved their behavior in terms of performance accuracy and confidence to be placed in the design solutions and tradeoffs obtained with the MDO. This was obtained at a limited price in terms of computational effort; no high-fidelity analyses were included, nor were other relatively computationally intensive engineering methods such as the linear aerodynamics, 6 DOF trajectory, integrated Monte Carlo simulations, or bidimensional propulsive system analyses. As a consequence, the computational times for both MDAs and MDOs were only roughly doubled with the model upgrades, for a total design cycle time well below 2 s. This figure is in line with the soft constraint defined in Sec. I and seems acceptable in light of the significant accuracy advantage. In fact, it was verified that a <2 s design time leads to

Table 5 Ariane-5 ECA's and VEGA's performances from manual, fixed design trajectory optimization, and Monte Carlo sensitivity analyses distribution, to be compared with Table 2

	Ariane-5 ECA, kg		VEGA, kg	
Payload mass from manual	10,050	— —	1500	— —
Payload mass for fixed actual design	10,187	+1.4%	1573	+4.8%
Payload mass distribution mean value μ_P	10,272	+2.2%	1552	+3.5%
Payload mass distribution standard deviation σ_P	724	5.3%	132	8.8%
Payload mass expectable ($1\sigma_P$) range	[9548;10,996]	[−5.0; +9.4]%	[1420;1684]	[−5.3; +12.3]

Table 6 Summary of system-level conceptual and early preliminary results for trajectory optimizations with a fixed launcher design, Monte Carlo sensitivity analyses, MDA processes, and MDO processes

	Actual		V1C1 (Conceptual)		V1C1 (Early preliminary)	
	Ariane-5 ECA	VEGA	Ariane-5 ECA	VEGA	Ariane-5 ECA	VEGA
Fixed design P	10,050 kg	1500 kg	10,944 kg (+8.9%)	1715 kg (+14.3%)	10,187 kg (+1.4%)	1573 kg (+4.8%)
Expectable 1σP	— —	— —	[+4.0; +19.2]%	[−15.7; +16.1]%	[−5.0; +9.4]%	[−5.3; +12.3]%
MDA P	10,050 kg	1500 kg	12,472 kg (+24.1%)	1403 kg (−6.5%)	10,409 kg (+3.6%)	1585 kg (+5.6%)
MDA GTW	766.4 tons	138.1 tons	764.8 tons (−0.2%)	139.4 tons (+0.9%)	771.2 tons (+0.6%)	139.8 tons (+1.2%)
MDA CPL	173.0 M€	32.0 M€	171.6 M€ (−0.8%)	34.8 M€ (+8.7%)	180.5 M€ (+4.3%)	35.1 M€ (+9.7%)
MDA MSP	0.966	0.980	0.927 (−4.0%)	0.975 (−0.5%)	0.926 (−4.1%)	0.977 (−0.3%)
MDA CPU time	— —	— —	~0.6 s	~0.5 s	~1.4 s (2 trajectory–structure iteration, no reentries)	~1.2 s
MDO GTW	— —	— —	505.8 tons (−34.0% with respect to actual −20.5% MDO effect)	131.7 tons (−4.6% with respect to actual −8.9% MDO effect)	711.6 tons (−7.2% with respect to actual −5.8% MDO effect)	134.2 tons (−2.8% with respect to actual −1.6% MDO effect)
MDO CPL	— —	— —	137.4 M€ (−20.6% with respect to actual)	33.2 M€ (+10.7% with respect to actual)	173.6 M€ (+0.4% with respect to actual)	35.7 M€ (11.6% with respect to actual)
MDO MSP	— —	— —	0.926 (−4.1% with respect to actual)	0.975 (−50% with respect to actual)	0.931 (−3.6% with respect tot actual)	0.977 (−0.3% with respect to actual)
MDO CPU time	— —	— —	~15 h(~4 h + ~1 h global + local, x3)	~13 h(~3 h + ~1 h global + local, x3)	~32 h(~9 h + ~2 h global + local, x3)	~27 h(~7 h + ~2 h global + local, x3)

nearly converged optimal solutions for medium-size MDO problems (i.e., 50 variables) in about 1 day. Because of the potential for an almost linear speedup through the parallelization of stochastic population-based global algorithms, a very large search space exploration could be enabled by small-scale HPC.

VII. Conclusions

This paper presented a detailed analysis of the application of the multidisciplinary design optimization approach to the concept and early design exploration of expendable launch vehicles. The rationale behind the use of MDO lies in the reduction of the design effort, and hence financial cost, as well as in the wider investigation of the design space, which allows the identification of better design solutions with respect to traditional methods, for instance, in terms of the LCC.

The focus of the research was on the engineering modeling aspects, with the idea of attempting a detailed analysis of the accuracy of the developed methods, both in terms of disciplinary errors and of system-level sensitivities. Although MDO research has often been applied to launch systems, the evaluation of accuracy and reliability of the design models to the extent described in the present research is a rather new endeavor, which may prove useful for the advancement of the field. In fact, the difficulty of finding a suitable compromise between the analysis fidelity and computational effort represents one of MDO's most challenging aspects, having contributed to the prevention of its successful widespread industrial application. The main objective of the present research was therefore to quantitatively assess the accuracy of engineering-level MDO models to determine their applicability for industrial design activities. In particular, the feasibility of developing relatively simple analyses, permitting fast MDA cycles while still ensuring sufficient confidence in the resulting solutions, was investigated. A straightforward answer is not easy to find; although the required computational effort matches the original target (i.e., <2 s for a complete MDA with the final models), it is not nearly as easy to measure the accuracy requirement. It is, however, clear that the careful, incremental modeling process described in Secs. III through VI proved to be of key importance for achieving good subsystem-level accuracy in most areas, at a manageable price in the computational effort. Moreover, an accurate and robust assessment of the launcher payload mass through trajectory optimization is a key capability for ensuring a fair comparison of different design concepts. This was obtained through significant efforts in this area, leading to payload estimates always within 5% of the actual figures for existing launchers.

As a consequence, expectable 1σ errors on the global performance are estimated to be lower than 12% with the final models, which should be sufficient in most of the cases to fairly compare two significantly different design solutions through the MDO approach. Extensive testing of the MDA functionality on the Ariane-5 ECA and VEGA test cases confirmed that performance errors are within this range for European launchers. Besides, the MDO was shown to be capable of physically sound design modifications, leading to sensible reductions in the GTW of existing vehicles. As far as purely performance objectives are concerned (i.e., GTW and P mass), the multi-objective MDO also provides very useful design information. For example, the minimum GTW vs maximum P optimization can be exploited to understand the sensitivity to variations in the payload mass requirement, an extremely important capability for real-world design exercises.

Nevertheless, important limitations to the MDO methodology were also highlighted, which should be further addressed as a future development. First, models tuning is a critical aspect largely affecting the outcome of the MDO process. In fact, the good behavior of the models described so far is subject to the consideration that only European launchers were tested, and the same vehicles had often been used for models calibration due to a lack of information on other systems. In particular, inert masses and components lengths are more prone to experimental tuning than other modeling aspects. Further testing on vehicles of widely different technologies and design concepts therefore appears necessary to understand the level of accuracy that results from such extrapolations. Moreover, although

CPL and MSP figures achieved for Ariane and VEGA are quite accurate, the introduction of the cost and reliability objectives in the MDO was not particularly successful. Specifically, mathematical features intrinsic to the CPL and MSP formulations constitute a serious obstacle for global multi-objective optimization. In fact, top-down cost estimation is mostly a mass-based approach, often resulting in cost reductions when minimizing the mass and vice versa. The MSP model is instead quite insensitive to the values of all continuous design variables, being mostly affected by the architectural and technological aspects. Finally, numerous parameters are defined within the cost and reliability models, again strongly affecting the results of the optimizations. Parameters tuning is thus a particularly critical open point for the MDO, which needs to be further addressed with additional testing on a wider set of test cases.

Besides these immediate areas of research, several other possible developments can be foreseen, both to further improve the modeling fidelity and to expand the applicability of the multidisciplinary environment to other types of space transportation systems. The evaluation of the impact of introducing high-fidelity information in the MDA cycle seems of particular interest, for example, in the form of the automated finite elements analysis coupled to analytical failure modes analysis to refine the structural models. Although high-fidelity methods should, in general, improve the accuracy, the increased computational requirement would impose the exploitation of high-performance computing and possibly of metamodeling techniques, which would become additional investigation areas.

In conclusion, the research presented in this paper highlighted how the development and tuning of reliable MDO environments is a particularly complex and challenging task, requiring large efforts in most engineering areas as well as computer science and mathematics. However, detailed investigations showed that reasonable accuracies and physically sound design modifications can be obtained through the MDO approach, even when exploiting only fast engineering-level models. With today's computational resources, possibly enabling the introduction of high-fidelity information in the design cycle, the MDO guided by human expertise is a powerful approach for the initial design phases of launchers and other space transportation systems. Although the substantial initial investment in terms of development and personnel training is a major obstacle to its widespread industrial application, the resulting benefit in terms of design quality may very well be worth the effort, with the potential of contributing to the long-term goal of achieving a low-cost access to space.

Acknowledgments

The research presented in this paper is part of a fruitful collaboration of the Aerospace Engineering Department of Politecnico di Milano and the Zentrum für Technomathematik of University of Bremen within the framework of ESA's Program in Education for Space, Technology, Innovation and Knowledge Ph.D. program. The authors would therefore like to thank Guillermo Ortega, Antonio Rinalducci, Alvaro Martinez Barrio and Sven Erb of ESA's TEC-ECN section for their continuing support and technical advice as well as Javier Ventura-Traveset and Carlos Lopez from the Education Office at European Space Astronomy Centre in Madrid, who started the Program in Education for Space, Technology, Innovation and Knowledge program and sponsored the Ph.D. research on MDO. Besides, many people from European Space TEchnology Centre and Von Karman Institute contributed with expert advice in different areas for the modeling of launch and reentry vehicles: Jens Kauffmann, Carlos Corral Van Damme, Gianluigi Baldesi, Marco De Rosa, Francesco Di Matteo, Marco Chiappone, Michel Van Pelt, Cem Ozam, Patrick Rambaud, Massimiliano Bottacini, and Fabio Caramelli. The authors would also like to acknowledge the researchers that provided the code and advice for the optimization algorithms: Kalyanmoy Deb for NSGA-II, Joshua Knowles for PAES, Abel Garcia Najera for MOACOr and the NOMAD and WORHP development teams. Finally, special thanks to Paolo Martino, who was responsible for the development of the cost and reliability models and to Chris Laurel for his great help in all software-related issues.

References

[1] Hammond, W. E., *Design Methodologies for Space Transportation Systems*, 1st ed., AIAA Education Series, AIAA, Reston, VA, 2001, pp. XX–XX.

[2] Sobieski, J., "Multidisciplinary Design Optimization: Attempt at Definition," *Proceedings of Industry-University Workshop on Multidisciplinary Aircraft Design*, Virginia Polytechnic Institute and State University, Blacksburg, VA, 1993, pp. 23–48.

[3] AIAA Technical Committee on Multidisciplinary Design Optimization, "White Paper on Current State of the Art," AIAA, Washington, D.C., Dec. 1991.

[4] Sobieszczanski-Sobieski, J., "Multidisciplinary Optimization for Engineering Systems: Achievements and Potential," NASA TM-101566, March 1989.

[5] Malone, B., "On the Financial Impact of MDO on the Corporation," AIAA Paper 2002-5495, 2002.

[6] Diedrich, A., Hileman, J., Tan, D., Willcox, K., and Spakovszky, Z., "Multidisciplinary Design and Optimization of the Silent Aircraft," AIAA Paper 2006-1323, 2006.

[7] Olds, J. R., "Multidisciplinary Design Techniques Applied to Conceptual Aerospace Vehicle Design," Ph.D. Thesis, North Carolina State Univ., Raleigh, NC, 1993.

[8] Braun, R. D., "Collaborative Optimization: An Architecture for Large-Scale Distributed Design," Ph.D. Thesis, Stanford Univ., Stanford, CA, 1996.

[9] Durant, N., Dufour, A., Pain, V., Baudrillard, G., and Schoenauer, M., "Multidisciplinary Analysis and Optimisation Approach for the Design of Expendable Launchers," AIAA Paper 2004-4441, 2004.

[10] Marius, T. J., "Multidisciplinary Optimization for Early System Design for Expandable Launchers," AIAA Paper 2006-7125, 2006.

[11] Bayley, D. J., Hartfield, R. J. Jr., Burkhalter, J. E., and Jenkins, R. M., "Design Optimization of a Space Launch Vehicle Using a Genetic Algorithm," AIAA Paper 2007-1863, 2007.

[12] Young, D. A., Olds, J. R., Hutchinson, V., Krevor, Z., Pimentel, J., Reeves, J. D., Sakai, T., and Young, J., "Centurion: A Heavy-Lift Launch Vehicle Family for CisLunar Exploration," AIAA Paper 2004-3735, 2004.

[13] Castellini, F., Riccardi, A., Lavagna, M., and Büskens, C., "SVAGO MDO Environment for the Design of Expendable Launch Vehicles," *5th International Conference on Astrodynamics Tools and Techniques*, ESA/European Space Research and Technology Centre, June 2012.

[14] Kennedy, J., and Eberhart, R., "Particle Swarm Optimization," *Proceedings of IEEE International Conference on Neural Networks*, Vol. IV, Inst. of Electrical and Electronics Engineers, New York, pp. 1942–1948.

[15] Castellini, F., "Global Optimization Techniques in Space Missions Design," Masters Thesis, Aerospace Engineering Dept., Politecnico di Milano, Milan, 2008.

[16] Deb, K., Agrawal, S., Pratap, A., and Meyarivan, T., "A Fast Elitist Non-Dominated Sorting Genetic Algorithm for Multi-Objective Optimization: NSGA-II," *Proceedings of Parallel Problem Solving from Nature VI Conference*, Springer, New York, Sept. 2000, pp. 849–858.

[17] Najera, A. G., "Extending ACOR to Solve Multi-Objective Problems," *Proceedings of the 2007 UK Workshop on Computation Intelligence*, July 2007.

[18] Castellini, F., Riccardi, A., Lavagna, M., and Bueskens, C., "Multidisciplinary Design Optimization Models and Algorithms for Space Launch Vehicles," *13th AIAA Multidisciplinary Analysis and Optimization Conference*, AIAA Paper 2010-9086, Sept. 2010.

[19] Castellini, F., and Lavagna, M., "Comparative Analysis of Global Techniques for Performance and Design Optimization of Launchers," *Journal of Spacecraft and Rockets*, Vol. 49, No. 2, March–April 2012, pp. 274–285.
doi:10.2514/1.51749

[20] Nikolayzik, T., Bueskens, C., and Gerdts, M., "Nonlinear Large-Scale Optimization with WORHP," *13th AIAA Multidisciplinary Analysis and Optimization Conference*, AIAA Paper 2010-9136, Sept. 2010.

[21] Balesdent, M., Bérend, N., and Dépincé, P., "Optimal Design of Expendable Launch Vehicles Using Stage-Wise MDO Formulation," AIAA Paper 2010-9324, 2010.

[22] Mollmann, C., Wiegand, A., Dalheimer, M., Martinez Barrio, A., and Kauffmann, J., "Multidisciplinary Design Optimisation of Expendable Launchers in ASTOS," *4th International Conference on Astrodynamics Tools and Techniques*, May 2010.

[23] Young, D. A., Olds, J. R., Hutchinson, V., Krevor, Z., Pimentel, J., Reeves, J. D., Sakai, T., and Young, J., "Centurion: A Heavy-Lift Launch Vehicle Family for Cislunar Exploration," AIAA Paper 2004-3735, 2004.

[24] Isakowitz, S. J., *International Reference Guide to Space Launch Systems*, 3rd ed., AIAA, Reston, VA, 1998, pp. XX–XX.

[25] Gordon, S., and McBride, B. J., "Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Applications, Volume I: Analysis," NASA RP-1311-P1, Oct. 1994.

[26] Gordon, S., and McBride, B. J., "Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Applications, Volume II: Users Manual and Program Description," NASA RP-1311-P2, June 1996.

[27] Sutton, G. P., and Biblarz, O., *Rocket Propulsion Elements*, 7th ed., NASA Lewis Research Center, Cleveland, Ohio, 2001.

[28] Huzel, D. K., and Huang, D. H., "Design of Liquid Propellant Rocket Engines," NASA SP-125, 1967.

[29] Craidon, C. B., "A Description of the Langley Wireframe Geometry Standard (LaWGS) Format," NASA TM-85767, Feb. 1985.

[30] Vukelich, S. R., Stoy, S. L., and Burns, K. A., "Missile DATCOM, Volume I: Final Report," McDonnel Douglas Technical Report AFWAL-TR-86-3091, Saint Louis, Missouri, Dec. 1988.

[31] Rohrschneider, R. R., "Development of a Mass Estimating Relationship Database for Launch Vehicle Conceptual Design," Georgia Inst. of Technology, Rept. AE-8900, Atlanta, April 2002.

[32] Riccardi, A., Castellini, F., Bueskens, C., and Lavagna, M., "Global and Local Optimization Approaches for Launch Vehicles Ascent Trajectory Design," *62nd International Astronautical Congress*, XXXXX, XXXX, XX, Oct. 2011.

[33] Koelle, D. E., *Handbook of Cost Engineering for Space Transportation Systems: TRANSCOST-Model 7.21*, rev. 2007, TransCostSystems, Germany, 2007.

[34] Castellini, F., "Multidisciplinary Design Optimization for Expendable Launch Vehicles," Ph.D. Dissertation, Aerospace Engineering Dept., Politecnico di Milano, Milan, 2012.

[35] Cramer, E. J., Frank, P. D., Shubin, G. R., Dennis, J. E. Jr., and Lewis, R. M., "On Alternative Problem Formulations for Multidisciplinary Design Optimization," AIAA Paper 1992-4752, Sept. 1992.

[36] Kroo, I., Altus, S., Braun, R., Gage, P., and Sobieszczanski–Sobieski, J., "Multidisciplinary Optimization Methods for Aircraft Preliminary Design," AIAA Paper 1994-4325, 1994.

[37] Ledsinger, L. A., and Olds, J. R., "Optimized Solutions for Kistler K-1 Branching Trajectories Using Multidisciplinary Design Optimization Techniques," *Journal of Spacecraft and Rockets*, Vol. 39, No. 3, May–June 2002, pp. 420–429.
doi:10.2514/2.3825

[38] Ricciardi, A., "Complete Geometric Analysis of Cylindrical Star Grains," AIAA Paper 1989-2783, 1989.

[39] Hartfield, R., Jenkins, R., Burkhalter, J., and Foster, W., "A Review of Analytical Methods for Solid Rocket Motor Grain Analysis," AIAA Paper 2003-4506, 2003.

[40] "Turbopump Systems for Liquid Rocket Engines," NASA SP-8107, Aug. 1974.

[41] "Terrestrial Environment (Climatic) Criteria Handbook for Use in Aerospace Vehicle Development," NASA HDBK-1001, Aug. 2000.

[42] Hutchinson, V. L., and Olds, J. R., "Estimation of Launch Vehicle Propellant Tank Structural Weight Using Simplified Beam Approximation," AIAA Paper 2004-3661, 2004.

[43] Bruhn, E. F., Orlando, J. I., and Meyers, J. F., *Analysis and Design of Missile Structures*, Tri-State Offset Co., Cincinnati, Ohio, 1967.

[44] Crawford, R. F., and Burns, A. B., "Minimum Weight Potentials for Stiffened Plates and Shells," *AIAA Journal*, Vol. 1, No. 4, April 1963, pp. 879–886.
doi:10.2514/3.1658

[45] Shanley, F. R., *Weight Strength Analysis of Aircraft Structures*, 2nd ed., Dover, New York, 1960, pp. XX–XX.

D. Thunnissen
Associate Editor

Queries

1. AU: Please review the revised proof carefully to ensure your corrections have been inserted properly and to your satisfaction.