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LAUNCH VEHICLES MULTIDISCIPLINARY OPTIMIZATION, A STEP FROM CONCEPTUAL TO EARLY PRELIMINARY DESIGN

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A recent collaboration between Politecnico di Milano and Universität Bremen within ESA's PRESTIGE PhD program has stemmed a significant research effort in the field of Multidisciplinary Design Optimization (MDO) for launch vehicles. This work is aimed at the development and integration of optimization algorithms and engineering methods in a software environment capable of assisting in the conceptual and early preliminary design of space launchers, potentially leading to relevant reductions in development effort and life cycle cost. The implemented MDO approach allows in fact efficient exploration of the design space throughout successive global and local, single and multi-objective optimization processes, guided by the engineering experience of the designer.

The main obstacle to the successful application of MDO lays in the difficult task of finding a good compromise between models simplicity and accuracy. To tackle this issue, the engineering models were developed in two successive levels of detail, from conceptual to early-preliminary design. The paper is focused on this modelling effort, showing how a critical analysis of the first level's results was exploited to improve fidelity and functionality.

An overview of the conceptual design models is first presented, together with a quantitative assessment of their accuracy and of the impact of the disciplinary errors on global performance indexes. The models selection converged towards well-known disciplinary tools (NASA's CEA and USAF's Missile DATCOM), complemented by a set of ad hoc models in the following disciplines: propulsion, geometry, aerodynamics, weights, trajectory, guidance and control, costs and reliability assessment. The validation campaign showed how system-level errors in performance below 20% can be expected, and allowed identifying the most critical modelling aspects to be improved.

In a second part, the paper focuses on the model enhancements stemming from the analysis of the conceptual design results, in particular: solid grain geometry and internal ballistics analysis, pressurization systems and engine cycles modelling, simplified structural sizing for all load bearing components, effect of wind and steering losses on the trajectories, and safety-related analyses (boosters/stages impact ellipse determination, upper stage end-of-life strategy). Validation results are presented with a comparison of the conceptual and early preliminary frameworks, highlighting the advantages in terms of accuracy (down to 12% of worst case system error on performance) with a limited increase in computational effort.

The foreseen future research lines are finally discussed, especially those aimed at further increasing the design fidelity and at targeting less traditional launch systems, such as manned and reusable vehicles.

I. INTRODUCTION

The European Space Agency (ESA) proposed in 2009 to co-fund together with the Aerospace Engineering Department of Politecnico di Milano and the Centre for Industrial Mathematics of Universität Bremen a joint research in the field of Multidisciplinary Design Optimization (MDO). This work is aimed at developing and comparing different optimization algorithms, MDO architectures and engineering methods to identify the most suitable for Expendable Launch Vehicles (ELV) design. The target is for the conceptual and early preliminary level of detail, for use in early design studies of ELVs, considering extensions to more complex applications such as manned and reusable systems.

In the past decades, the development of ELVs and Space Transportation Systems (STS) in general has been affected by continuously increasing financial concerns. For the foreseeable future, this limitation is not anticipated to change. In fact, no remarkable advancement in the exploration and exploitation of space is to be expected worldwide unless a drastic reduction of the cost for the access to space can be achieved. In Europe, the Future Launchers Preparatory Program (FLPP)[1] is aimed at paving the way, through both technology developments and system studies, for a Next Generation Launcher (NGL), with the goal of reducing specific launch costs and increasing the flexibility of the current launchers family.

In the sector of STS, it has recently been recognized that around 80% of the Life Cycle Cost (LCC) is determined in the conceptual design phase[2], whereas design decisions performed later only have marginal effect. For this reason, the approach of MDO, if successfully applied to the conceptual design of new vehicles, has the potential not only to reduce the development times and costs, but also to minimize the LCC of the system. MDO was described by the AIAA's MDO Technical Committee as "a methodology for the design of complex engineering systems and subsystems that coherently exploits the synergy of mutually interacting phenomena"[3]. In practice, MDO models incorporate all relevant disciplines simultaneously, allowing to achieve optimal solutions superior to those found by optimizing each discipline sequentially. The availability of a reliable MDO environment supporting the designers lowers the manpower necessary for the early design phases. Besides, the design space can be more rapidly explored, analyzing a higher number of possible solutions and obtaining Pareto optimal fronts under different aspects, such as mass, cost, reliability, or mission flexibility. Designers can then select the most promising concepts to be used as starting points for refinements with more traditional design methodologies. This should in theory ensure the selection of the design option granting for example the lowest possible LCC, if this is the overall project's goal.

However, the set-up and calibration of MDO methodology in conceptual and early preliminary design studies is particularly challenging, so that successful industrial applications are extremely rare. The major hurdle to be overcome lays in the difficult task of finding a compromise between the simplicity and the accuracy of the engineering models: unmanageable computational times need to be avoided, but the physics of the problem has to be sufficiently represented in order to validate the obtained design solutions and place confidence in the performed trade-offs.

Examples of recent aerospace industrial applications are NASA/Boeing's Blended-Wing-Body (BWB) research aircraft[4] and the European Union's HISAC project[5] for an environmentally friendly supersonic transport, which made extensive use of MDO. The final configurations selected in both projects were strongly affected by the MDO results. The BWB reached the level of flying prototype, validating the MDO process employed in the conceptual and preliminary stages.

More specifically for launch vehicles and space transportation systems, MDO applications have been limited to pilot researches and design exercises, none resulting in MDO being used to the extent described above for advanced aircraft projects. The first steps in the development of multi-disciplinary models were undertaken in the 1990s by Olds, Braun and others[6,7,8,9], but the lack of computational power restrained the application to the study of specific launcher configurations and prevented from the introduction in the optimization cycle of complex disciplinary models. Besides, the Global Optimization (GO) approach that appears necessary when dealing with large multi-modal and mixed continuous-discrete search spaces and with multiple contrasting objectives was never used due to its limited maturity. More recently, some industrial[10],[11] and academic[12] efforts in this area considered automatic trade-offs among different configurations with Genetic Algorithms (GA), leading to interesting results. However, these solutions are limited to an early conceptual level, employing simple disciplinary models and lacking of efficient distributed architectures as well as of multi-objective and Local Optimization (LO) refinement capability. On the other hand, a purely local approach was followed in an industrial environment[13,14] achieving optimized design at a conceptual level starting from an initial guess in the desired region of the global search space.

Elaborating on the background presented above, the present research combines the advantages of GO and LO, with the aim of tackling engineering models for ELVs suitable up to the early preliminary design level. This approach, synergically developed by the two involved research centres, is being implemented by means of a modular object oriented (C++) software tool named SVAGO (Space Vehicles Analysis and Global Optimization). The end customer is the European Space Agency, who will use it for concurrent design and industrial design evaluation in the early phases of launch systems development, such as for the FLPP. In this context, the role of one or few experienced system engineers is still critical, since the MDO environment is designed to be extremely interactive. The user is in fact allowed (and encouraged) to guide the optimization throughout successive global and local runs, with full control over the design variables, constraints and objectives. As an outcome of this MDO-based design process, a few preferred concepts can be chosen for more detailed studies, to be executed with Concurrent Engineering or more traditional sequential design approaches.

The optimization architecture developed for SVAGO is constituted of several global stochastic population based algorithms and a local gradient-based method. In this first class, a single-objective version of the Particle Swarm Optimization (PSO) method is implemented, as well as four multi-objective codes: the Non-Dominated Sorting Genetic Algorithm II (NSGA-II)[15], the Double-Grid Multi-Objective Particle Swarm Optimization (DG-MOPSO)[16], the Multi-Objective Ant Colony Optimization for real values (MOACOr)[17] and their collaborative hybridization Hybrid-GO. In the second class instead, the large and sparse NLP solver WORHP (We Optimize Really Huge Problems)[18] developed in Europe is integrated.

This paper focuses on the engineering models for the problem of ELVs design, rather than on the developed MDO framework, optimization architecture and algorithms or MDO results, for which details are given in other publications[19,20]. With regards to the major modelling hindrance mentioned above, the engineering models were developed in two steps. First, a conceptual level modelling 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, which allows for higher fidelity and larger functionality at a reasonable price in terms of computational effort. The enhancements were implemented in a second modelling step, targeted to the early preliminary design, with a further validation campaign assessing the improvements in accuracy.

For the validation processes, two European launch vehicles were chosen as reference cases, Ariane 5 ECA and VEGA. Besides the good knowledge of their specifications which is available within ESA, these launchers have the advantage of being significantly different in architecture and technologies, constituting a sufficiently large base of comparison, at least in a European framework.

The paper is divided in the following sections:

- Section II: overview of engineering modelling for step 1: conceptual level disciplinary models and Multidisciplinary Design Analysis (MDA) cycle, main validation results and critical weaknesses.
- Section III: overview of engineering modelling for step 2: model enhancements for early preliminary level design and validation results in comparison with those of Section II.
- Section IV: concluding remarks with focus on the main modelling aspects still to be targeted both for further fidelity upgrade and for the extension to other classes of STS.

II. CONCEPTUAL LEVEL MDO

II.I Modelling

The engineering modelling of launch systems is a particularly complex task, even 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 execution of a full MDA on a single processor personal computer in less than a second. When this constraint is combined with the need to exploit only freely available software, the choice of the engineering models is rather limited. For this reason, many researchers in the past have independently converged toward common tools, such as Chemical Equilibrium with Applications (CEA) for propulsion performance analysis and Missile DATCOM for aerodynamics, or analogous in-house developed algorithms.

The disciplinary models for SVAGO were developed following this common approach, with the additional decision of allowing different modelling fidelities according to the anticipated impact of each discipline on the vehicle's performance. Assumptions were taken regarding such impacts, which were a posteriori verified with sensitivity studies. These assumptions include largely favouring the propulsion system analysis, which inarguably constitutes the most relevant aspect of any launch vehicle, with respect to other disciplines such as aerodynamics and weights.

Stemming from these considerations, the implemented disciplinary models are briefly described below together with their average accuracy measured in the validation procedure.

Propulsion

The propulsion analysis for each stage/booster is performed by either picking up an Off-The-Shelf (OTS) system from a database of 38 currently flying liquid rocket engines, collected from the International Guide to Space Launch Systems[21] and web sources, or by designing a new Liquid Propellant (LP) or Solid Propellant (SP) system. For new designs, chamber pressure, mixture ratio (only for LP engines) and expansion ratio are optimized in different ranges depending on the propellants and feed system type, and NASA's CEA[22] is run to compute the theoretical performance. Empirical corrections mostly derived from standard propulsion sources[23,24] and further calibrated on existing engines are applied for I_{sp} losses, whereas the inert masses are obtained through Weight Estimation Relationships (WER) developed from the above database. Additional models are implemented for geometric dimensions (following the scaling based on the optimized thrust level) and minimum operational altitude due to shock waves in the nozzle.

The validation has been performed comparing the model outputs with a database of existing liquid and solid engines, showing extremely good accuracy on vacuum I_{sp} (less than 1% average error) and larger discrepancies on other parameters (e.g. around 20 % for the average error on the engine's inert mass).

Geometry

Cylindrical stages and boosters with cylindrical or conical interstages and power law fairing ogive are assumed, considering only the diameters continuity or the length-over-diameter ratio as optimizable variables. The external geometry parameters are obtained mainly from the propulsion system dimensions, allowing additional volume for interstages, intertanks and equipment compartments. These are translated into geometry files in the Langley Wireframe Geometry Standard (LaWGS)[25], which was selected as geometry format due to its simplicity and common interfaces with many simplified aerodynamic analysis codes. Simple 2D and 3D Gnuplot-based visualization tools from the Public Domain Aeronautical Software (PDAS[26]) (3-View and Silhouette) were also linked for easier visual inspection.

Aerodynamics

The largest influence of aerodynamics on the global performance of ELV is in the subsonic and low supersonic regimes (i.e. M=[0.6-3]), for which even linear aerodynamics panel codes involve rather high computational loads. However, this impact is still fairly limited, hence Missile DATCOM[27] is integrated in SVAGO to determine lift, drag and pitching moment coefficients as a function of Mach and total Angle of Attack (AoA). Being a collection of semi-empirical formulas with components build-up approach, DATCOM allows a database of 20-by-5 Mach-AoA points to be generated in about 0.2 seconds, and can therefore be efficiently executed within the MDA loop to provide C_L and C_D for trajectory integration as well as C_m for static controllability verification.

Validation against VEGA and Ariane-5 data internally available at ESA have shown average errors in the 10-15% range for CD and 15-20% for CL and Cm, even though deviations as high as 100% occur for several flight conditions.

Weights

No structural analysis model is implemented in the conceptual level environment, due to the excessive computational load associated with Finite Elements (FE) and the complexity increase related to simplified methods such as beam approximations. Hence, simple WERs are implemented for both structural and non structural weights, mainly taken or adapted from a comprehensive Georgia Tech published collection[28]. The only exception is related to Solid Propulsion (SP) stages and boosters, for which the case's mass is computed with pressure vessel sizing relations, which have shown very good correlation with a set of solid rocket motors despite their simplicity.

Validation against existing launchers mass properties has given satisfactory results for conceptual level design, with most errors ranging from 10% to 25%.

Trajectory and control

A 3-DoF dynamics and limited environmental models (zero-order gravity, US 76 atmosphere, no wind) are considered appropriate for this step, and a Runge-Kutta-Fehlberg 45 integrator allows to simulate the launcher trajectory from launch to orbit insertion. Parameterized pitch and yaw constitute the control parameters and a set of standard guidance laws (vertical launch, linear pitch-over, target inclination, gravity turn, bi-linear tangent law, plus a final circularization burn) defines a first guess for the optimization. For LP, 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 thrust, linearly decreasing thrust (i.e. to match acceleration constraints) or two-level thrust (i.e. to match dynamic pressure constraints). Additional models are included to account for propulsion performance variation with altitude and boosters/core in-flight ignitions, as well as for a series of path constraints evaluation (heat flux, axial and lateral accelerations, dynamic pressure, static controllability, geographic heading). Validation of the trajectory integration and guidance strategies was performed against the AeroSpace Trajectory Optimization Software (ASTOS)[29] and the results are presented in a previous work[30], showing negligible errors on simulated trajectories.

An accurate and robust evaluation of the Payload (PL) performance through trajectory optimization is particularly critical for the multidisciplinary design of launch vehicles, since a fair comparison of the different concepts can only be ensured if errors in the payload assessment are small. A separate paper[31] describes in details the implemented performance optimization process, with focus on the specific aspects that allow for efficient and robust utilization of local (gradient-based) algorithms for this task.

It has however to be noted that the developed 3-DoF ascent trajectory description tends to overestimate the payload performance. In particular, Ariane 5 ECA's maximum payload to standard GTO is of ***10944 kg*** (***+8.9%*** with respect to the reference 10050 kg from the manual) and VEGA's payload to 700 km polar LEO is of ***1715 kg*** (***+14.3%*** with respect to the reference 1500 kg). Although these discrepancies may be partially due to uncertainties in the design parameters (inert masses, aerodynamic coefficients, specific impulses and exhaust areas), several modelling weaknesses were identified: a) roughness of SP thrust model, only approximately following actual motors profiles; b) lack of steering losses due to aerodynamic moment compensation, manoeuvres or wind; c) simplicity of the controllability verification models, neglecting dynamic effects or wind;

d) overestimation of the I_{sp}, which is assumed constant in spite of the degradations occurring as the flight progresses due to throat erosion or pressure variations.

Cost and reliability

Particular attention was paid to the non performance related modelling aspects, which are often neglected in MDO studies but represent driving criteria in today's design-to-cost and design-to-reliability approaches. This is reflected in SVAGO in the definition of the total Life Cycle Cost (LCC) and Mission Success Probability (MSP) of the ELV being designed, which can be used by the user as MDO objectives together with classical performance-based criteria. The LCC of the launch vehicle is estimated through Cost Estimation Relationships (CER) from the transparent TRANSCOST model[32], adapted to fully reflect all technological trade-offs defined by the optimization variables of the MDO process and complemented with additional CERs internally available at ESA. The MSP is instead evaluated through a time-dependent analysis of the failure chains in the different mission phases, with components failure rates data again provided by ESA. The validation of both cost and reliability models was performed against ESA data for both European and non-European launchers. The cost models were shown to generically overestimate the launch cost of Ariane-5, VEGA, Sojuz, Delta and Falcon families, but their ranking is well predicted. Similarly, MSPs obtained with SVAGO are in line with the historical failure rates for Ariane-5, Sojuz and Delta, whereas VEGA's foreseen reliability (ESA estimate) is accurately matched. Finally, reliability advantages coming from features such as Falcon's engine-out-capability or Common Core Boosters (CCB) configurations are clearly reflected in the model results.

Multidisciplinary Design Analysis cycle

The most common tool used to represent an engineering multidisciplinary design cycle is the Design Structure Matrix (DSM), which shows the flow of data among the different disciplines constituting the MDA. From the disciplinary analyses described in the previous paragraphs, the rather self-explicative DSM of Fig. 1 is built up, where Fig. 2 qualitatively details the involved vectors of user parameters, optimization variables, cross-disciplinary variables and disciplinary outputs.

The common practice in launchers design is to set-up a feed-back of the structural, thermal and possibly control loads from the trajectory back to the structural analysis and weights estimation models, therefore requiring an iterative loop to close the design cycle. Due to the simple WERs implemented in the MDA cycle, the only trajectory parameters required for the execution of the weights module are the maximum encountered axial acceleration, heat flux and dynamic pressure, represented by the below-diagonal term X_{54}. In order to eliminate this feed-back and thus avoid iterations, these parameters are introduced as system level optimization variables. In this way, they can simultaneously be used in the weights module to define the components' dry masses and in the trajectory module to define the thresholds of the related path constraints.

For this conceptual-level design environment, the launcher design and trajectory optimization variables are divided in discrete and continuous variables as follows:

- *Discrete variables*: architectural parameters (number of stages and boosters, boosters arrangement configuration, CCB architecture option, reuse of the same engine on all stages, number of engines per each stage and booster), technological trade-offs for propulsion (OTS vs. new design, propellants, feed system, nozzle and thrust vector control types, restart and throttle capabilities), geometry (continuous vs. discontinuous diameters), weights (tanks arrangement and type, materials concept, smarts redundancy level) and finally cost/reliability oriented variables (horizontal/vertical processing, number of qualification tests for the engines, low-cost engine option, engine-out-capability).
- *Continuous variables*: stages/boosters geometry (length-over-diameter ratios) and propulsion systems parameters (propellant mass, nominal thrust, chamber pressure, area ratios, mixture ratio, ...), trajectory load parameters (heat flux, axial acceleration, dynamic pressure), and finally trajectory control variables (discretized pitch, yaw and thrust along different flight phases, ignition times).

The resulting number of launcher design variables equals $13+26\cdot(N_{stages,max}+N_{boosterSets,max})$, where $N_{stages,max}$ and $N_{boosterSets,max}$ represent the maximum number of stages and of booster sets to be included in the design. The number of trajectory optimization variables largely varies depending on the controls discretization settings. However, feasible trajectories can be obtained with very small problems (i.e. one parameter per each control per each flight phase, usually resulting in 10-15 optimizable parameters), allowing for the best robustness and efficiency of the process at the price of very small performance losses. 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.

The most relevant constraints are imposed in the trajectory module in the form of final errors on the orbital parameters and path constraints on heat and structural loads, static controllability and ground-track. Design-related constraints are instead imposed on the lift-off thrust-to-weight, geometric interferences, thrust

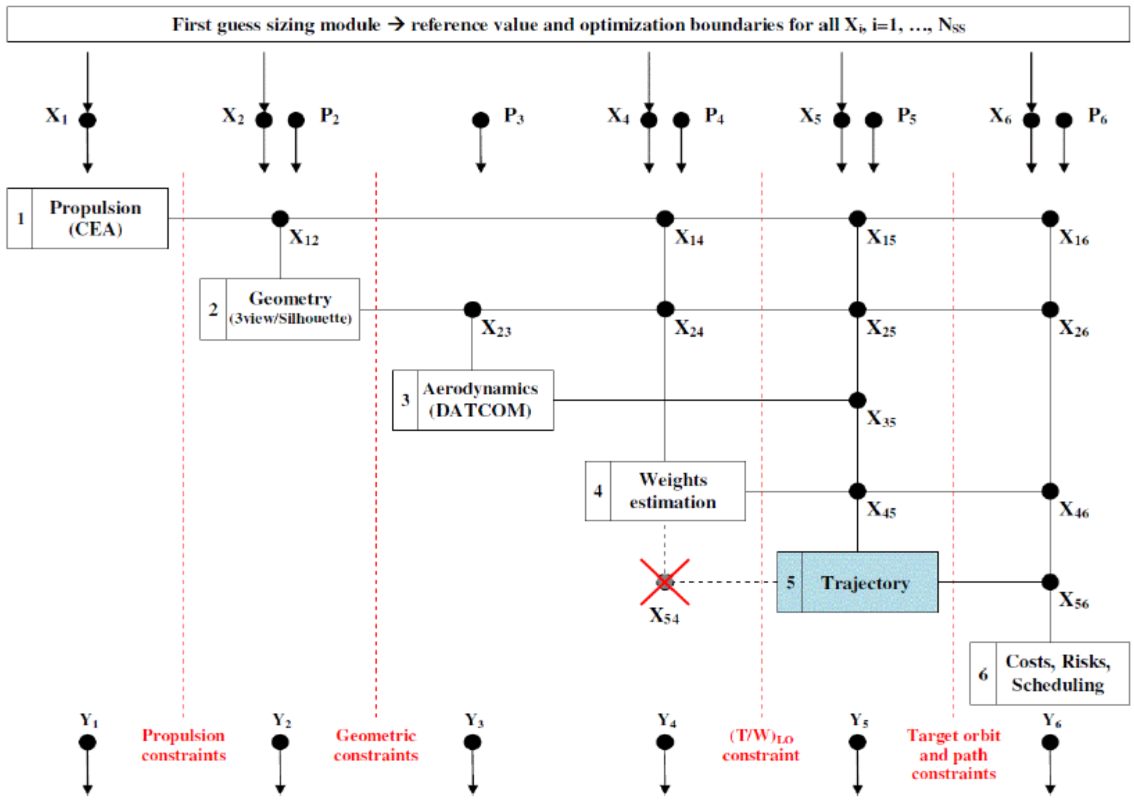

Fig. 1: ELV conceptual-level DSM: design proceeds sequentially along the diagonal from left to right, with terms above/below the diagonal representing feed-forward and feed-back information flows. All input/output/coupling variables included in the DSM are qualitatively described in Fig. 2.

P_2	Payload (PL) length and diameter	X_{12}	Prop. system lengths and diameters	X_{46}	All non propulsion masses
P_3	Aerodyn. database discretization settings	X_{14}	Prop. system masses and CoG positions	X_{56}	Flight phases durations
P_4	PL max. heat and structural loads	X_{15}	I_{sp}, min. operative altitude, exhaust area	X_{54}	Encountered loads: this feedback is eliminated by using max. heat and struct. loads as optimiz. variables, input to weights and constraints to traj.
P_5	PL mass and target orbit (a, e, i)	X_{16}	Prop. system masses		
P_6	Program and cost factors	X_{23}	Complete launch vehicle geometry	Y_1	Propulsion constraints, all prop. sys. specifications
X_1	Architecture, prop. 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, non prop. technol. trade-offs, max. heat and structural loads	X_{26}	Fairing length and volume	Y_4	Detailed weights breakdown structure, total mass at launch, take-off 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 technol. trade-offs, cost & reliability oriented variables	X_{45}	All non propulsion masses and CoG longitudinal positions	Y_6	Detailed cost breakdown structure, mission success probability vs. mission time, LCC, global reliability

Fig. 2: Qualitative definition of all the vectors of variables represented in the DSM of Fig. 1 for each discipline j: P_j: parameters fixed by the user; X_j: optimization variables (groups of variables may be in common with other disciplines); X_{jk}: coupling variables with discipline k; Y_j: disciplinary outputs (objectives or constraints).

range for each technology, and failure of one of the external design tools (CEA or DATCOM).

Finally, four criteria or any of their combinations through weight factors are considered for single- or multi-objective optimization: LCC, MSP, Gross Take-Off Weight (GTOW), and Payload (PL) mass excess with respect to the required performance on the reference target orbit.

II.II Validation results

The validation of the engineering models was started at the subsystem-level, evaluating the accuracy of each disciplinary analysis through comparison with real world data or other software. Results for this phase are briefly mentioned in the previous paragraphs, and a summary is given in Table 1.

Discipl.	Parameter	M [%]	μ [%]	σ [%]
Propulsion	$I_{sp,vac}$ [s]	3.1	-0.6	1.3
Propulsion	A_e [m^2]	31.2	-0.9	15.0
Propulsion	M_{engine} [kg]	28.7	-0.1	11.2
Aerodyn.	C_D (M,α)	81.8	+4.3	9.3
Aerodyn.	C_L (M,α)	98.5	+9.1	14.3
Weights	$M_{fairing}$ [kg]	33.6	-8.7	16.4
Weights	$M_{inert,SP}$ [kg]	36.1	+8.3	16.1
Weights	$M_{inert,LP}$ [kg]	37.6	-3.3	14.2

Table 1: Summary of subsystem level validation results: maximum M, mean μ and standard deviation σ of the errors with respect to database on disciplinary outputs.

However, in order to evaluate the suitability of the developed models, the accuracy assessment for each discipline is not sufficient. The different errors combine in fact in the MDA process to determine the overall error on the global performance of the vehicle. A detailed analysis of the sensitivity of the global performance to the disciplinary errors was therefore executed. Hence, trajectory optimization loops were run for the test launch vehicles (Ariane 5 ECA and VEGA), with the main disciplinary output parameters perturbed with respect to the actual launchers design by percentages reflecting μ and σ of the disciplinary errors.

Two analyses types were executed:

- *One-variable-at-a-time analyses*, i.e. increasing or decreasing one of the relevant parameters in Table 1 by a percentage equal to μ±σ, the goal being identifying the critical disciplines most likely to determine large errors in the global performance, on the basis of both their relevance and modelling accuracy.
- *Monte-Carlo simulations*, i.e. randomly varying all parameters at the same time according to a (μ,σ) Gaussian distribution. This generates a distribution for the launcher's performance due to the realistic Gaussians of the disciplinary outputs. Any bias toward higher or lower performance can therefore be identified and, even more important in a MDO context, 1σ or 3σ confidence ranges for the performance can be estimated.

The one-variable-at-a-time analyses clearly highlighted how the most critical discipline is the weights analysis, with large errors on the payload performance when the perturbed parameter is the total inert mass of one of the stages or boosters. In particular, the large payload sensitivity to the upper stage mass $\left(\partial M_{PL} / \partial M_{dry,us} = -1\right)$ leads to errors up to 17.5%. This suggests investing relevant modelling efforts for the early preliminary environment on the structural analysis and weight estimation. On the contrary, propulsive parameters such as the vacuum specific impulse are less critical in spite of the larger sensitivity, due to the much higher accuracy available. Finally, aerodynamic errors do not seem to largely affect the performance, due to a very low sensitivity (ΔM_{PL} is lower than 1.5% when a constant ΔC_D of 15% is assumed throughout the flight).

Results of the Montecarlo analyses are instead summarized in Table 2, with the mean values of the performance distributions showing a bias towards respectively over and underestimation of the payload mass for Ariane 5 and VEGA. This different behaviour can be again traced back mainly to the weight models, and in particular to the different mean errors for the dry masses of LP and SP systems. As regards to the standard deviation, σ=16% for Ariane 5 and σ=8% for VEGA were instead obtained. Although these are reasonable figures for 1σ performance error in conceptual MDO, an improvement in accuracy appears essential to allow for industrial applicability of the design environment.

	Ariane 5 ECA to standard GTO	VEGA to circular polar 700 km LEO
$\mu_{payload}$	11217 kg (+11.6%)	1488 kg (-0.2%)
$\sigma_{payload}$	762 kg (+7.6%)	239 kg (+15.9%)

Table 2: Montecarlo sensitivity analyses results: mean μ and standard deviation σ of the payload performance for Ariane 5 ECA and VEGA.

The final step in the validation of the conceptual level design models is represented by the evaluation of the payload figures obtained with a MDA for Ariane 5 ECA and VEGA. The purpose is in this case of assessing the capability of the multidisciplinary design environment of providing reasonable performance figures, when starting from actual launch vehicle's design. Hence, all input design variables (i.e. X_j in Fig. 1) were frozen to the actual values, and the complete design cycle was executed including a nested trajectory optimization loop for Ariane 5's GTO and VEGA's polar LEO. The system level outputs of the MDA process are reported in Table 3, confirming the overestimation of Ariane 5's performance and the underestimation of VEGA's. The quantitative figures (+24.1% and -6.5%) fall within the expected 1σ payload variability from the Montecarlo analyses, confirming the consistency of the results. The largest cause for the errors is again mainly to be attributed to the lack of accuracy in the inert masses estimation. Although the GTOW of both Ariane and VEGA very closely matches the correct values, the mass breakdown among components shows relevant errors. In particular, Ariane's upper stage dry mass is underestimated by 2.7 tons with respect to the ESC-A stage plus Vehicle

Equipment Bay, due to insufficient modeling of the different structural and non structural components located in the upper stage. For VEGA instead, P80 and Z23 motors' dry masses are sensibly overestimated (25% and 19%), probably because VEGA nozzles employ new technologies and materials which are not captured with historical SP motors weight models.

Although it was possible to quantitatively assess the accuracy of the models with respect to performance indexes, only a qualitative understanding of the fidelity of the cost and reliability assessment is possible, since detailed cost breakdown structures or failure data are not available for comparison. Launch costs appear to be in general overestimated with respect to the declared prices for Ariane 5 ECA and VEGA. The MSP=0.975 obtained for VEGA matches very well the target 98% reliability, whereas a pessimistic MSP=0.927 for Ariane 5 ECA (97% success rate with 1 failure out of 29 launches) suggests that LP systems' reliability may be underestimated.

	Ariane 5 ECA		VEGA	
	Actual	MDA	Actual	MDA
M_{PL}	10050 kg	12476 kg	1500 kg	1402 kg
GTOW	763.4 t	764.8 t	138.1 t	139.4 t
CpL	150 M€	171 M€	30 M€	37 M€
MSP	0.966	0.927	0.980	0.975

Table 3: Summary of system-level MDA results for Ariane 5 ECA and VEGA, including payload mass, total mass, cost per launch and reliability.

III. EARLY PRELIMINARY MDO

III.I Modelling

The early preliminary MDO environment was obtained through the upgrade of all disciplines in the DSM of Fig. 1. These improvements were derived in part from the critical analysis of the validation results, as presented in the previous section, and in part from an independent review by ESA. The following paragraphs present an overview of the main modelling enhancements which were deemed necessary, either to improve the overall accuracy of the models (at a manageable price in terms of computational time) or to add specific functionalities necessary for an industrial application of the MDO tool.

Solid rocket motors design

From the analysis of the conceptual trajectory models, one of the key weaknesses was identified in the lack of realistic thrust profiles for Solid Rocket Motors (SRM) for either boosters or stages. This was solved with two upgrades: 1) the OTS engines database was extended to include SRMs with predefined thrust and vacuum specific impulse profiles; 2) models were added for the geometric description of the solid grain in new design SRMs, including the generation of the thrust profiles as a function of the geometry and the analysis of the internal ballistics to assess the chamber pressure and specific impulse variation during the burn.

Among the OTS SP engines, current European boosters and stages (P-241, P-80, Zefiro-23, Zefiro-9) as well as a selection of 11 motors from ATK (including the Shuttle's RSRM) are included in the database.

For new design grains instead, three options are available for either user selection or optimization:

- *Custom motors*: Filling Factor (FF), Sliver Fraction (SF), normalized thrust and I_{sp} profiles are given by the user, and the motor's maximum thrust and propellant mass (hence dimensions) are allowed to be scaled in given ranges. Custom motors can be used to represent modifications of existing motors, for which grain characteristics are assumed unchanged in front of scale variations. No geometric design or internal ballistic analyses are therefore to be performed.
- *End-burning motors*: used for upper stage or kick motors, it is the simplest grain geometry available. Burn area A_b, internal pressure p_{cc}, thrust T, and specific impulse I_{sp} are assumed to be constant, with FF=1 and SF=0. No grain geometric or internal ballistic analysis is hence required, except for the matching of burn time t_b and web length L_{web}.
- *Internal-burning motors*: used for booster or lower stages applications, this option allows a rather detailed geometrical design of the SP grain. The resulting thrust profiles are realistic representations of actual booster/stages, even though the ignition and burn-out transitories are neglected.

For the latter option, up to 5 segments with different grain cross-sections and relative lengths can be specified, each with one of three internal perforation types: 1) *tube grain*, 2) *slot grain*, or 3) *star grain*. Thrust build-up is assumed to be instantaneous and thrust varies according to the grain geometrical design until the sliver is reached. Tube grains have the Web Fraction (WF) as the only geometric parameter $w = L_{web} / R_{ext}$ (where L_{web} is the web length and R_{ext} is the external radius of the stage/booster), whereas slot and star grains respectively need 3 and 4 additional parameters for a complete geometry description. Typical geometries for slot and star grains are shown in Fig. 3, and the mathematical relations for the burn perimeter and port area, which determines the FF for t=0 and the SF for $t=t_b$, were derived from bibliography[33,34]. The burn perimeter and port area are multiplied by the length of each segment and summed over the total number of segments to give the overall burn area and internal cavity volume.

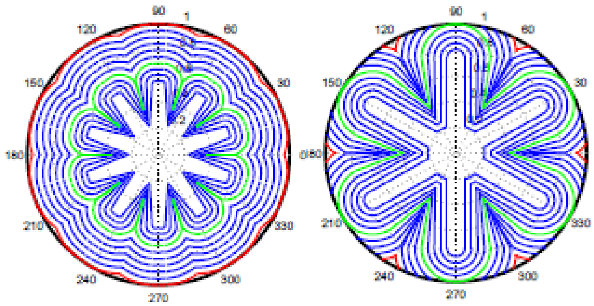

Fig. 3: Examples of star (left) and slot (right) grain geometries with burn back diagram. Green line: switch between phase I and II burning. Red line: sliver fraction of the grain (defining burn-out)

The geometric analysis is only a part of the SRMs design, which is made up of several steps as follows:

1. If $N_{segments}>1$: w for all segments except the first is adjusted to have constant WF throughout the grain.
2. Geometric analysis, with all parameters normalized so that the outputs are the adimensional burn area $A_{burn}/(L_{grain}\cdot R_{ext})\,(t)$, SF and FF.
3. Grain scaling, defining the length of the grain as:
$$L_{grain}=\frac{M_{prop,usable}}{\pi\cdot R_{ext}^{2}\cdot\rho_{grain}\cdot FF\cdot(1-SF)}$$
from which the dimensional parameters are obtained.
4. CEA nominal performance analysis, I_{sp} losses and minimum altitude evaluation, as in conceptual models.
5. Engine nozzle scaling, performed at maximum mass flow conditions (maximum burn area and pressure) through mass balance between grain and throat.
6. Inert masses estimation, as in conceptual models.
7. Evaluation of p_{cc}, I_{sp} and T profiles over time, either through a CEA iterative loop for each time point or through an approximated but much faster procedure assuming constant chamber properties and using the isoentropic relations for the expansion in the nozzle.
8. Evaluation of the matching constraint between burn time and web length: t_b can be derived integrating the calculated T(t) and I_{sp}(t) profiles, the web length is obtained from WF and R_{ext}, and the burn rate r_b is an optimization variable (used also at steps 5 and 7 above) that can be varied by the optimizer to match the constraint $L_{web}=r_b\cdot t_b$. It is implicitly assumed that the user provided range of burn rates can be obtained by modifying the propellant formulation with suitable quantities of burn-rate modifiers.

As an example of thrust profile that can obtained with this procedure, Fig. 4 shows the calculated T(t) for a tube-slot grain configuration where the 5 free geometric parameters were optimized to minimize the error with respect to VEGA P80's thrust profile.

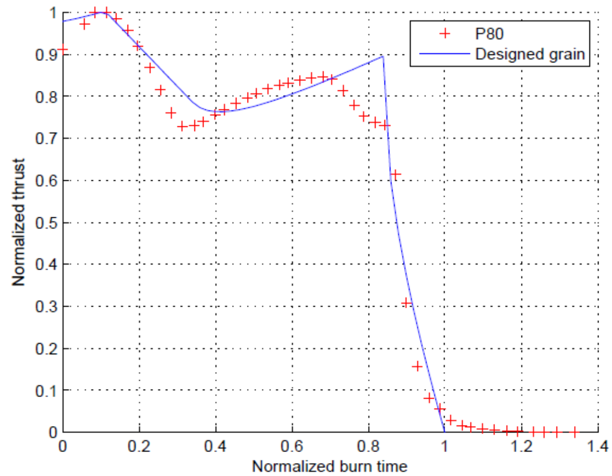

Fig. 4: Examples of thrust profile obtained with 1 tube and 1 slot segment to approximate VEGA's P80.

<u>Pressurization system and cavitation analysis</u>

The second largest upgrade in the propulsion models is related to the analysis of pressurization systems and of the pumps cavitation in LP turbopump systems. The goal is twofold: to improve the estimation of the inert masses of the propulsion system and to introduce the pressure in the LP tanks p_{tanks} among the design variables to be optimized. The trade-off on p_{tanks} is in fact particularly important in LP stages/boosters, since a higher pressure determines heavier tanks and pressurization system at the advantage of an easier suppression of the pumps cavitation, which cannot instead be achieved without a boost turbopump system if p_{tanks} is too low.

Three different types of pressurization can be selected by the uer (or optimized) for both the oxidizer and fuel: 1) *Evaporated propellant*, only for cryogenic propellants and best suited for hydrogen due to its low molecular weight (e.g. Shuttle's External Tank, Ariane 5's EPC and ESC-A stages); 2) *Heated Helium*, with a source of high pressure He stored in separate tanks and heated up through heat exchangers in the turbines discharge. This allows both to reduce the mass of He required for the pressurization and to store He at very low cryogenic temperatures, therefore resulting in much higher density and lower tanks volume and mass (e.g. Ariane 5's EPC, ESC-A oxygen pressurization); 3) *Stored Helium*: with He expansion directly from its tank to the propellants tanks, thus avoiding the complication of heat exchangers. This ensures maximum simplicity, at the cost of a larger mass, and is therefore suited to smaller pressure-fed upper stages (e.g. VEGA AVUM).

Two different models were implemented for the estimation of the pressurization gas mass (evaporated propellant or separate He): ideal gas law applied to the final ullage conditions and an energy conservation approach considering adiabatic expansion of the pressurization gas. Through calibration of few parameters on Saturn V, 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 mass figures for the overall pressurization system mass of the considered stages are reasonably well matched, with average and maximum errors lower than respectively 10% and 20%.

Although a full engine cycle analysis and turbomachinery design is out of scope (there would be insufficient reliable data to develop complete weight estimation relationships at the component level), a pumps cavitation analysis was implemented, starting from the tanks pressure, chamber pressure, head rise due to gravity/accelerations and pressure losses in the lines/turbopumps. If cavitation is detected, the required mass for boost pumps is estimated through a quadratic regression obtained from NASA literature[35] (see Fig. 5) and is added to the engine's inert mass.

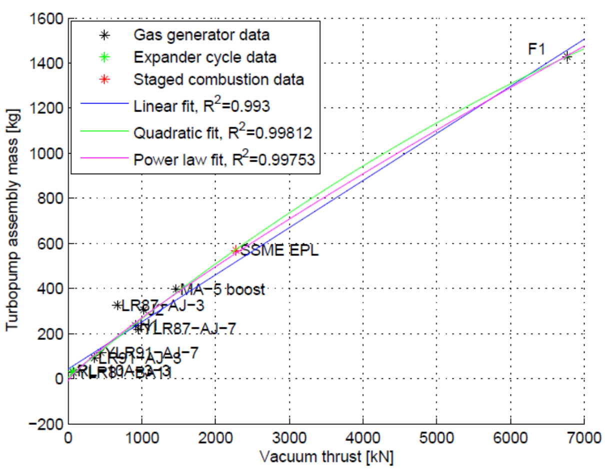

Fig. 5: Boost turbo-pump assembly weight estimation relationship developed from NASA historical data.

The procedure to detect cavitation can be summarized in the following steps:

1. Compute friction losses from tank to pump
2. Compute the additional head rise due to the tank's elevation and hence the pump's suction pressure.
3. Compute the Net Pump Suction Head (NPSH), defined as: $NPSH = (p_{suction} - p_{vapour})/(\rho \cdot g_0)$
4. Compute the pump's discharge pressure from p_{cc}, using different loss coefficients depending on engine cycle and nozzle cooling (ablative or regenerative), hence oxidizer/fuel's path from pump to chamber.
5. Compute the Pump Head Rise (PHR), defined as: $PHR = (p_{discharge} - p_{suction})/(\rho \cdot g_0)$
6. Fixing the pump's specific speed N_S and the pump's suction specific speed S from literature, the Required Pump Suction Head (RPSH) can be estimated as:
$$RPSH = \left(\frac{21.2 \cdot N \cdot \sqrt{\dot{V}}}{S}\right)^{4/3} \quad with \;\; N = \frac{N_S \cdot PHR^{0.75}}{21.2 \cdot \sqrt{\dot{V}}}$$
in British units and $\dot{V}$ being the volumetric flow.
7. No cavitation condition is then assumed when $RPSH \leq CM \cdot NPSH$, with the cavitation margin CM=0.8 corresponding to a net suction head 20% higher than that causing cavitation. Note that in spite of its simplicity, this model is able to correctly predict the need for a boost system in 27 out of 29 tested liquid rocket engines. However, the accuracy of the boost turbopump assembly mass estimation for current technology is questionable, since all engines in Fig. 5 used for the WER generation were designed before 1975.

Other propulsion enhancements

Several other minor model additions were necessary to extend functionalities and improve accuracy. In particular, the following are aimed at extending the functionalities of the conceptual models: 1) introduction of additional LP fuels with detailed properties (e.g. Russian Kerosene vs US RP-1); 2) multiple thrust chambers and extendable nozzle options for LP engines, with trade-off between engine's length and inert mass; 3) submerged nozzle option for SP engines, with trade-off between engine's length and I_{sp} loss; 4) linear I_{sp} degradation versus burn time for ablatively cooled nozzles due to throat erosion.

On the other side, several modifications are aimed at improving the I_{sp} or inert mass estimation accuracy: 1) introduction of a corrective coefficient for the I_{sp} loss of SP engines as a linear function of the nozzle's expansion ratio; 2) enhanced modelling of the unused LP mass, with breakdown in trapped propellants, mixture ratio unbalance, cryogenics boil-off (upper stages with coast phases only), contingencies (all stages/boosters) and reserve (upper stages only), and End-of-Life (EoL) propellant for upper stages de-orbiting or graveyard orbit transfer; 3) modification of all the existing WERs from the conceptual level models for the inert mass of SP and LP systems, necessary to account for the new modelling features described above.

The subsystem-level validation process described in Section II.I for the conceptual models was repeated for the early preliminary models, showing a significant improvement in the I_{sp} accuracy for SP engines, which reaches 0.3% as average error. As regards to the engine's dry mass instead, the average error is reduced to 13% for SP and 12% for LP.

Geometry and aerodynamics upgrades

Aerodynamic models were not largely modified, except for the introduction of multiple aerodynamic configurations (e.g. with and without boosters). This is justified by the small sensitivity of the global performance to errors in this discipline, as highlighted in the validation of the conceptual models.

Three new important geometric functionalities were however introduced:

- Definition of the separation plane between stages, determined with a minimum nozzle disengagement angle of 15 deg, as shown in Fig. 6. This ensures the minimization of the mass staying on the upper stage.
- Introduction of under-fairing configurations for the upper stages, which allows improving the weight estimation for small stages such as VEGA's AVUM.
- Introduction of unconventional tanks geometries such as enclosed tanks (e.g. Ariane 5's ESC-A) and multiple tanks (e.g. Fregat, Breeze, AVUM), again with the aim of enhancing the inert mass accuracy.

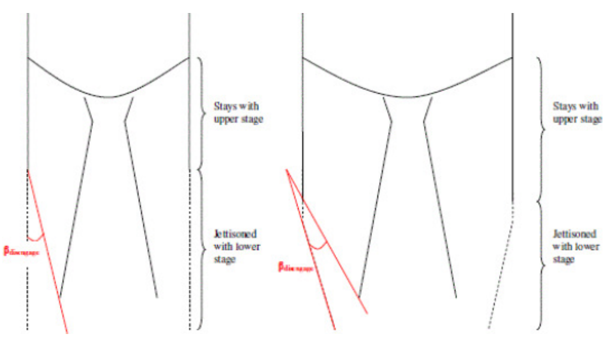

Fig. 6: Separation plane definition through nozzle disengagement angle for stages with constant and varying diameters.

Structural analysis and sizing

The structural analysis and sizing is a disciplinary block introduced specifically for the early preliminary models and aimed at estimating the mass of all structural components of stages and boosters. In particular, the following mass items can be sized: SRM case, fuel and oxidizer tanks, intertanks and interstages, thrust frames, pad interface structures, payload adapter, payload fairing and boosters nose ogive.

The procedure was derived from a recent work[36] defining a "beam approximation" for launch vehicle structures, complemented by more classical structural analyses practices[37]. Few simplifying assumptions were taken, in particular considering all longitudinal cross-sections as circular and aerodynamic forces as concentrated in the Centre of Pressure (CoP). The resulting structural analysis process follows:

1. Define the load cases from the trajectory simulation, in particular: take-off, Mach=1, max Q_{dyn}, max $Q_{dyn}\cdot\alpha$, max n_{ax} for each stage/booster.
2. For each load case, define the distribution of masses along the longitudinal axis of the launcher's core and boosters separately. Fig. 7 shows an example of mass distribution for Ariane 5's core. Each mass item per each load case is described by a mass value, a start and end position (which coincide for concentrated masses) and possibly a reaction station position in case of cantilevered items.
3. For each load case, determine CoG, CoP, thrust application point and longitudinal inertia.

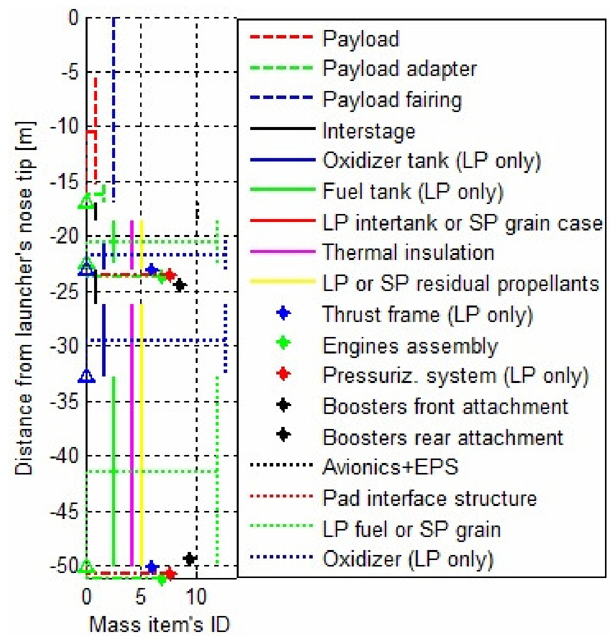

Fig. 7: Example of mass distribution along the beam approximation of Ariane 5 ECA's core. Concentrated masses are identified by a star. Cantilevered masses have an additional triangle at the reaction station.

4. For each load case, compute the external loads at each discrete longitudinal station, starting from the inertia, aerodynamic and thrust loads. The three types of external loads considered are axial force P, shear force T, and bending moment M.
5. For each load case and station, compute four considered internal running loads: hoop (circumferential) and axial (longitudinal) loads, shear (transverse) load and longitudinal bending moment. Contributions from the external loads and the internal tank/case pressure are both taken into account. The internal pressure (both ullage pressure and head pressure are considered) is the only cause of hoop load and contributes to the axial load.
6. Determine the worst case running loads for each station along the longitudinal axis, which are grouped to form the different structural components.
7. For each station, estimate the required shell thickness necessary to withstand the running loads. A material minimum gage is imposed, and three failure

modes are considered: ultimate strength, yield strength, and buckling. To prevent general instability, the available shell configurations are associated to longitudinal frames, whose smeared thickness is determined from the Shanley's criterion[38].

8. From the calculated shell and frames thicknesses, determine the mass of each structural component by station-by-station integration.

This procedure (more details are given in Ref. 36) 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 principals prevents from considering non-optimum weights such as bulkheads, minor frames, coverings, fasteners, and joints. Hence, structural weights are generally underestimated, and correlations to existing vehicles introduced in Ref. 36 were implemented to correct for this inaccuracy.

As a final note, two optimization variables were introduced for each structural component: the structural material and the stiffening concept. The selected materials are aluminium alloy 7075, titanium alloy 6Al-4V, steel, an Al-Li alloy and carbon-epoxy. Three stiffening concepts are instead available[36], with different values of buckling efficiency, Shanley equation's exponent and minimum gage parameter: simple integrally stiffened shell, Z-stiffened shells, and a truss-core sandwich shell design.

Wind model and steering losses

The most relevant weaknesses highlighted from the validation of the conceptual models are the lack of steering losses associated to aerodynamic moment compensation and the lack of wind modelling, affecting both the steering losses and the static controllability verification.

Wind influences the nominal trajectory as well as the structural loads and the control requirements, with the most critical effects from the horizontal winds in the [6-15] km altitude range. The analysis of the effects of wind on the nominal trajectory (control system dynamic verification, 3σ dispersion of insertion error) was considered too much detailed in the frame of an early preliminary semi-automatic design environment. However, a wind model based on look up tables was implemented from NASA's Handbook on Terrestrial Environment[39]. Tables for the steady-state horizontal wind are used as worst-case wind at each altitude for the static controllability verification. A synthetic wind profile constructed from steady-state wind, wind shears and wind gusts is instead used as continuous profile for the evaluation of the wind-related steering losses to be integrated along the trajectory. Both profiles are shown in Fig. 8.

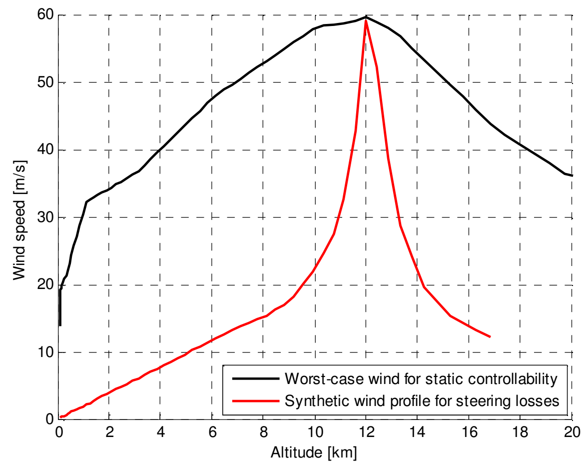

Fig. 8: Worst-case and synthetic wind profiles assumed for launches from Kourou.

The static controllability analysis is maintained unchanged for the early preliminary models, with the thrust torque constrained to be higher than the aerodynamic moment multiplied by a controllability margin of 1.5. The only difference is that the AoA provided by the guidance laws is modified due to the presence of a worst-case wind speed. This is conservatively taken as the steady-state envelope value for the given altitude plus a design wind gust as described in NASA handbook.

In order to mitigate the performance overestimation evinced from the conceptual models validation, steering losses were also introduced. For this purpose, the lateral thrust necessary to balance the aerodynamic moment caused by AoA and synthetic wind, and to allow for the required angular acceleration in case of pitch/yaw manoeuvres, is computed. The deflection angle for the Thrust Vector Control (TVC) that ensures the required torque is then obtained, starting from the boosters and following with the core in case of parallel architectures.

Although testing showed how the losses due to discrete manoeuvres are negligible, the overall loss is relevant, with an increase of respectively 28 m/s and 42 m/s of steering ΔV for Ariane 5 and VEGA optimal trajectories.

Safety issues: re-entry analyses

A critical industrial need lacking in the conceptual design environment is the availability of a set of safety related analyses for all components of a multi-stage launch vehicle. For orbital upper stages, this involves adding to the propellant budget the quantity necessary for a ΔV ensuring a safe de-orbit burn from LEO or a graveyard orbit transfer for GEO. As regards to suborbital boosters and stages instead, safety issues impose that the expected ground impact ellipse is not located within inhabited regions. This may represent a very important constraint for many launch sites and target orbits, affecting the final performance of the launch vehicle.

To model the impact safety constraint, some form of simulation of the re-entry of all suborbital components of the launcher is necessary. Two different models were implemented for this purpose: 1) integration of the 3-DoF equations, simplified by neglecting the lift force and therefore representing fully ballistic trajectories, and 2) propagation of the Keplerian parameters from jettison to ground impact, complemented with empirical models developed from the 3-DoF simulations to estimate the downrange reduction due to drag.

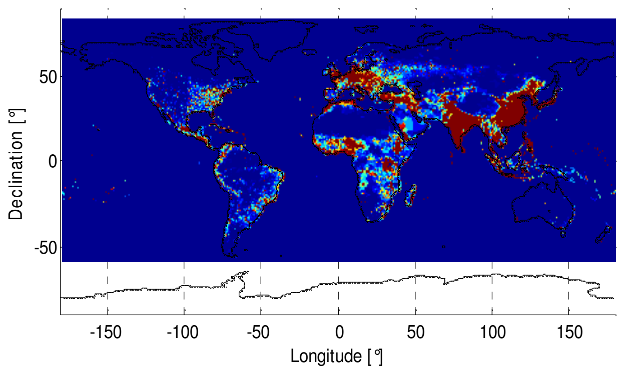

Fig. 9: Earth's population density map from GPWv3.

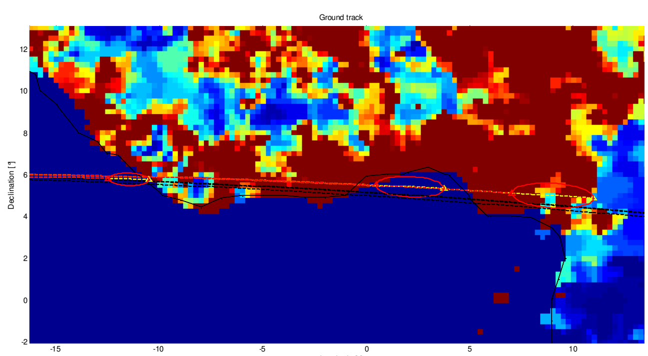

Fig. 10: Safety constraint activation for the re-entry of Ariane 5's core stage: left/centre ellipses correspond to two optimal trajectories with safety constraint, right ellipse to trajectory without safety constraint.

In both cases, lift and other effects on the trajectory such as wind and off-nominal jettison conditions aer neglected, due to the large parameters' uncertainties. To avoid the computationally expensive Montecarlo approach, only two trajectories are simulated with min-drag and max-drag ballistic coefficients. A constant ratio of minor-to-major ellipse axis equal to 0.3 is then assumed to determine an approximate impact ellipse. Finally, a population density map of the Earth is overlapped, from which the maximum population density on the ellipse is obtained. The allowed maximum value is set by default to 0 persons/km^2, defining the safety constraint.

The population projections for 2015 from the Gridded Population of the World, version 3 (GPWv3)[40] are employed, with a resolution of 0.25 deg (Fig. 9). The correct imposition of the safety constraint was verified for the Ariane 5 ECA's 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. Fig. 10 shows how the introduction of the safety constraint leads to an adjustment of the trajectory to allow for an impact ellipse either in Gulf of Guinea or west of Liberia, if an additional uncertainty margin on the ellipse is assumed. 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. Note that an alternative is to launch on a 7 deg GTO (which is actually the case for real launches), resulting in a more Southern impact location for a ~110 kg loss.

The combination of the trajectory model upgrades described in this paragraph results in a better assessment of the payload performance for Ariane 5 ECA and VEGA test launchers. Specifically, when freezing all specific impulses, inert masses and aerodynamic properties to the available data, the performance estimated through the conceptual and early preliminary models is reported in Table 4, showing errors lower than 5% for both launchers.

	Reference	Conceptual models	Early preliminary models
Ariane 5 ECA to GTO	10050 kg	10944 kg (+8.9 %)	10172 kg (+1.2 %)
VEGA to polar 700 km LEO	1500 kg	1715 kg (+14.3 %)	1573 kg (+4.8 %)

Table 4: Performance assessment for Ariane 5 ECA and VEGA (frozen design) with conceptual and early preliminary level trajectory models.

Multidisciplinary Design Analysis cycle

The DSM describing the MDA cycle remains rather similar to that of the conceptual level models. Most of the vectors P_j, X_j, X_{jk} and Y_j include a larger number of parameters, reflecting the model enhancements described in this section, but the basic structure remains unchanged. Besides, the objective functions are the same, with only few additional constraints (related to safety and structures). The only major difference is represented by the introduction in the design cycle of the structural analysis module, which requires iterations with the trajectory block to achieve convergence on the inert mass of stages and boosters. In fact, at each iteration, loads from the simulated trajectory are used to size the structural components, therefore changing the inert masses, as shown in Fig. 11.

As a last remark, the Multi-Disciplinary Feasible (MDF) problem formulation (i.e. Trajectory-Structures iterations to achieve feasibility at each solution) can be substituted by an Individual Disciplines Feasible (IDF) formulation. This consists in adding $N_{stages}+N_{boosterSets}$ optimization variables corresponding to the inert masses of all stages and booster sets, which are used for the trajectory integration and are afterwards compared with the inert masses from the weight estimation and structural analysis blocks. The optimizer hence needs to adjust these variables to match the $N_{stages}+N_{boosterSets}$ additional constraints, so that the design feasibility is reached only at the algorithm's convergence. The IDF formulation has the advantage of cutting the CPU time for a single MDA, avoiding the Trajectory-Structures iterations, at the cost of slower convergence due to the additional variables and constraints. The overall convenience of one strategy over the other still has to be assessed.

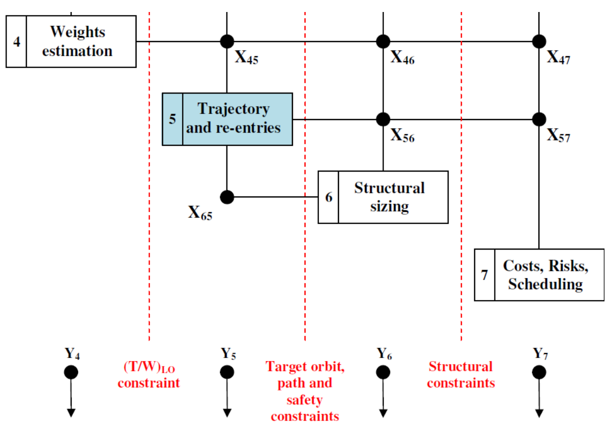

Fig. 11: ELV early preliminary level DSM, lower part only. The upper part is identical to that of Fig. 1.

III.II Validation results

The validation of the early preliminary models followed an identical procedure as for the conceptual models, with disciplinary stand-alone validation followed by sensitivity analyses and MDAs with nested trajectory optimizations for Ariane 5 ECA and VEGA to assess the capability of the multidisciplinary modelling to estimate the performances, cost and reliability of real world launch vehicles.

Subsystem-level results are summarized in Table 5, where the errors related to output parameters which were not affected by the model upgrades are not mentioned. The table shows lower disciplinary errors dispersion, in particular for the inert masses estimation. This determines a much lower dispersion of the launchers' performance in the Montecarlo analyses, whose results are reported in Table 6. Expectable 1σ performance errors are within 12%, which is a rather remarkable result given that high fidelity methodologies such as FEM or CFD were not used.

The expected performance error figures of Table 6 were confirmed by the MDA results, summarized in Table 7. The estimated payload values for Ariane 5 ECA to GTO and VEGA to polar 700 km LEO are respectively of ***10754 kg (+7.0%)*** and ***1433 kg (-4.5%)***, very close to the actual performances.

Discipline	Parameter	M [%]	μ [%]	σ [%]
Propulsion	$I_{sp,vac,LP}$ [s]	+1.6	-0.0	0.7
Propulsion	$I_{sp,vac,SP}$ [s]	-0.4	-0.0	0.3
Propulsion	$M_{engine,LP}$ [kg]	+31.8	-0.9	12.6
Propulsion	$M_{engine,SP}$ [kg]	-24.3	-2.5	12.1
Weights	$M_{fairing}$ [kg]	18.6	-1.3	9.5
Weights	$M_{inert,SP}$ [kg]	26.5	+2.2	13.5
Weights	$M_{inert,LP}$ [kg]	-27.4	+1.3	11.6

Table 5: Summary of subsystem level validation results for the early preliminary models, to be compared with conceptual results in Table 1.

	Ariane 5 ECA to standard GTO	VEGA to circular polar 700 km LEO
$\mu_{payload}$	10467 kg (+4.1%)	1467 kg (-2.2%)
$\sigma_{payload}$	532 kg (+5.3%)	142 kg (+9.5%)

Table 6: Montecarlo sensitivity analyses results for early preliminary model, to be compared with conceptual results in Table 2.

	Ariane 5 ECA		VEGA	
	Actual	MDA	Actual	MDA
M_{PL}	10050 kg	10754 kg	1500 kg	1433 kg
GTOW	763.4 t	759.8 t	138.1 t	138.5 t
CpL	150 M€	168 M€	30 M€	37 M€
MSP	0.966	0.928	0.980	0.977

Table 7: Summary of system-level MDA results for Ariane 5 and VEGA for early preliminary model, to be compared with conceptual results in Table 3.

Computational times for a complete MDA cycle are of extreme relevance in a MDO environment, because the MDA may need to be repeated thousands of times to achieve good design solutions. As a term of comparison, 3 global runs of 500 solutions per iteration and 200 iterations appear reasonable for large size MDO problems with the developed models, resulting in 300000 solutions to be evaluated. Table 7 summarizes the quantitative results achieved in terms of accuracy on the assessed performance and required CPU times for Ariane 5 ECA and VEGA test launch vehicles. The error on the estimated payload testifies a significant improvement from conceptual to early preliminary models, obtained at the cost of a limited and therefore justifiable increase in computational effort (from ~1 s to ~2 s).

Note that all CPU times are referred to the execution on a single processor of type Intel Core Duo T6500, 2.10 GHz, with 4 GB DDR2 RAM. Besides, ***1.7 s*** and ***2.0 s*** for the CPU time of the early preliminary models execution are obtained with the IDF formulation, therefore removing the Trajectory-Structures iterations. For the MDF formulation, about ***10-15 s*** (6-9 iterations) are instead in general necessary with a 1 kg tolerance to reach convergence.

	Conceptual models		Early preliminary models	
	Ariane-like	VEGA-like	Ariane-like	VEGA-like
Expectable 1σ PL error	[+4.0; +19.2] %	[-15.7; +16.1]%	[-1.2; +9.4] %	[-11.7; +7.3]%
MDA PL error	+24.1%	-6.5%	+7.0%	-4.5%
MDA CPU time	~1.0 s	~1.2 s	~1.7 s	~2.0 s

Table 8: Summary of expectable 1σ performance errors and CPU times for Ariane and VEGA MDAs (conceptual and early preliminary models).

IV. CONCLUSIONS

The paper presented an engineering modelling effort as part of a recent research on MDO carried out by Politecnico di Milano and Universität Bremen within ESA's PRESTIGE PhD program. The final goal of the research is to develop a reliable MDO environment, capable of assisting designers in the early design phases of ELVs and other STSs, potentially leading to large reductions in development effort and LCC.

Two multidisciplinary models were described, the first targeting the conceptual level of detail for ELVs design, and the second extending its functionalities and improving its accuracy for an early preliminary applicability. The objective was to obtain a reasonable compromise between models simplicity and accuracy, which represents the largest hindrance to the success of MDO approach. This was achieved through the critical analysis of the validation results obtained with the conceptual models, which allowed identifying the critical weaknesses and hence define the upgrades required to improve fidelity with limited increase in computational times.

The validation effort, using the European Ariane 5 ECA and VEGA as test cases, showed that errors on global performance indexes below 20% can be expected with the conceptual level models, with computational times for a full MDA cycle of ~1 s. The upgraded early preliminary models, besides introducing several new functionalities, ensure the reduction of this figure down to 12%, with full MDA cycles executed in ~2 s. This is an encouraging result, since such a level of accuracy allows placing reasonable confidence in the design solutions obtained with MDO processes, which only require computational times in the 5-20 hours range when exploiting modern workstations with multiple cores.

Several other upgrades of the disciplinary models can clearly still be carried out, such as: component level sizing for LP and SP systems, use of aerodynamic panel methods (linear aerodynamics and impact methods) substituting DATCOM, introduction of dynamic controllability verification with 6-DoF or multibody simulations, launcher's structural flexibility assessment, and enhancement of the accuracy of cost and reliability models through extensive benchmarking.

Finally, the design environment can be extended to include other classes of vehicles. For example, man-rating functionalities (e.g. launch escape system, crew interfaces, ...) can be added. Alternatively, the aerodynamic, weight, structural, guidance and control models for re-entry vehicles can be introduced, up to tackling the design of reusable systems. With further work on these modelling aspects, as well as on parallel computing capabilities, MDO represents a powerful tool in the initial design phases of ELVs and other STSs, with the potential to contribute to the long term goal of achieving low cost access to space.

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