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Collaborative Approach to Launch Vehicle Design

R. D. Braun* and A. A. Moore*
NASA Langley Research Center, Hampton, Virginia 23681
and
I. M. Kroo†
Stanford University, Stanford, California 94305

Collaborative optimization is a new design architecture specifically created for large-scale distributed-analysis applications. In this approach, a problem is decomposed into a user-defined number of subspace optimization problems that are driven toward interdisciplinary compatibility and the appropriate solution by a system-level coordination process. This decentralized design strategy allows domain-specific issues to be accommodated by disciplinary analysts while requiring interdisciplinary decisions to be reached by consensus. This investigation focuses on application of the collaborative optimization architecture to the multidisciplinary design of a single-stage-to-orbit launch vehicle. Vehicle design, trajectory, and cost issues are directly modeled in this problem, which is characterized by 95 design variables and 16 constraints. Numerous collaborative solutions are obtained. Comparison of these solutions demonstrates the influence that an a priori ascent-abort criterion has on the vehicle design and the distinction between minimum weight and minimum cost concepts. The operational advantages of the collaborative optimization architecture include minimal framework integration requirements, the ability to use domain-specific analyses, which already provide optimization without modification, inherent system flexibility and modularity, a distributed analysis and optimization capability, and a significant reduction in interdisciplinary communication requirements.

Nomenclature

A	= nozzle area, ft^2
c	= subspace constraints
F	= system-level objective
F_z	= wing normal force, lb
g	= interdisciplinary compatibility functions
I_{sp}	= specific impulse, s
S_{ref}	= wing area, ft^2
T	= thrust, lb
T/W	= thrust-to-weight ratio
W	= weight, lb
x	= interdisciplinary design variables
$\bar{x}$	= domain-specific design variables
y	= interdisciplinary outputs
z	= system-level design variables
α	= angle of attack, deg
ΔV	= ideal velocity
%	= propellant mass fraction, e.g., $\%LOX = W_{LOX}/W_{prop}$

Subscripts

e	= exit
prop	= total propellant
sl	= sea-level
vac	= vacuum

Superscript

$*$	= subspace solution

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*Aerospace Engineer, Space Systems and Concepts Division. Senior Member AIAA.

†Associate Professor, Department of Aeronautics and Astronautics. Senior Member AIAA.

Introduction

FOR several years, NASA has been examining various Earth-to-orbit transportation options with the goal of reducing operating costs relative to the current U.S. launch fleet.[1–4] Many of these solutions have focused on fully reusable systems employing various levels of advanced technology.[5,6] Although a wide range of options have been examined, including single- and two-stage systems using rocket and/or air-breathing propulsion, recent emphasis has been placed on single-stage-to-orbit, rocket-powered vehicles.[7–11]

The design of a single-stage-to-orbit launch vehicle is a complex, multidisciplinary process that may be characterized by thousands of design variables and nonlinear constraints. A complete design requires analysis of aerodynamics, propulsion, weights and sizing, performance, heating, controls, operations, cost, and other disciplines.[9] Although it is vital that each of these aspects be addressed at the conceptual level, the use of a design architecture that provides rapid performance of this multidisciplinary analysis while allowing the analyses to evolve as design maturity increases is equally imperative.

Enabled by the computational advances of the past few decades, a suite of multidisciplinary analysis and optimization architectures has emerged.[12,13] Collaborative optimization is a new design architecture whose characteristics are well suited to large-scale, distributed design.[14] The fundamental concept behind the development of this architecture is the belief that disciplinary experts should be able to contribute to the design process while not having to fully address local changes imposed by other groups of the system. To facilitate this decentralized design approach, a problem is decomposed into subproblems along domain-specific boundaries. Through subspace optimization, each group is given control over its own set of local design variables and is charged with satisfying its own domain-specific constraints. Communication requirements are minimal because knowledge of the other groups' constraints or local design variables is not required. The objective of each subproblem is to reach agreement with the other groups on values of the interdisciplinary variables. A system-level optimizer is employed to orchestrate this interdisciplinary compatibility process while minimizing the overall objective. This decomposition strategy allows for the use of existing disciplinary analyses without major modification[14] and is also well suited to parallel execution across a network of heterogeneous computers.[15]

Development of the collaborative optimization architecture is described elsewhere.[14,16,17] Comparison to other optimization architectures also has been performed.[14,17,18] The architecture has been successfully used to solve several analytical test problems,[14,19] trajectory optimization problems,[14,16] and aircraft design problems.[15,18] The present investigation focuses on the application of this architecture to the multidisciplinary design of a single-stage-to-orbit launch vehicle.

Disciplinary Analyses

In this analysis, design of a single-stage-to-orbit launch vehicle includes specification of the ascent trajectory, as well as determination of the subsystem weights and sizes and assessment of the technology maturation and development costs. From the results of a previous study,[20] an aerodynamically viable shape is modeled. Propulsion system characteristics to be identified include the liftoff thrust-to-weight ratio, nozzle area ratio, and propellant mass fractions. Multiple mass fractions require specification because the propulsion system is operated in different manners (modes 1 and 2) during flight. From liftoff, liquid hydrogen (LH_2) and liquid hydrocarbon (LHC) are both burned as fuel with liquid oxygen (LOX), whereas during a later portion of the ascent, only the hydrogen–oxygen mixture is used. The increased bulk density of such a dual-fuel concept has been shown to provide significant dry-weight reductions.[21] The time to transition from mode-1 propulsion to mode 2 is also optimally determined. Development cost is minimized. In the present analysis, this is defined to include the vehicle hardware design and development costs incurred under a full-scale development program. Other cost elements such as program management, fees, reserves, and software are not included in the objective function.

Historically, preliminary launch vehicle design has not included cost issues within the design loop. Instead, the pertinent engineering analyses are integrated and optimized to find either a minimum dry weight or minimum gross-liftoff-weight (GLOW) vehicle. A costing analysis subsequently is provided and, time permitting, some iteration may be performed. The goal of NASA's Reusable Launch Vehicle program is to provide technology development and demonstration of a low-cost reliable space transportation system.[11] As a result, this sequential design and costing approach is no longer appropriate; instead, it has become imperative to consider cost issues at the preliminary design stage.[22,23] Hence, in this investigation, the impact of including cost considerations within the preliminary design process is discussed for a representative, single-stage-to-orbit concept.

The vehicle is sized to deliver and return a 25,000-lb payload to the Space Station following launch from the Eastern Test Range at the Kennedy Space Center. For this analysis, the Space Station is assumed to be in a 220-n mile-altitude orbit with a 51.6-deg inclination. The single-stage-to-orbit vehicle is flown into a 50×100 n mile altitude orbit with the correct inclination; onboard propellant is used to transfer to and circularize at 220 n miles. Burnout constraints on altitude, flight-path angle, and inclination are enforced, as are maximum inflight normal force, angle of attack, pitch rate, and dynamic pressure limits. An extension limit is placed on the dual-position rocket nozzle. In addition, technology maturation costs are constrained along a representative funding profile.[24]

Propulsion Analysis

Propulsion system parametrics were supplied by Pratt and Whitney on the basis of characteristics of the Russian RD-701 dual-fuel engine.[21] This system is designed to burn either a hydrogen–kerosene (RP) mixture (mode 1) or pure hydrogen (mode 2) as fuel. During mode 1, hydrogen is included in the fuel mixture to provide nozzle cooling and increased I_{sp}. To further enhance performance, the dual-fuel engine is fitted with a dual-position nozzle. After a regression analysis, these parametric data can be used as shown in Fig. 1. Given the two nozzle area ratios, the LH_2 propellant mass fraction in mode 1, and the mode 2 fuel-to-oxidizer mixture ratio, numerous engine parameters are computed. These parameters include sea-level engine thrust to weight, specific impulse, and maximum allowable thrust (required inputs to the weights and sizing analysis), as well as vacuum thrust and nozzle exit area (required trajectory inputs). A 2:1 exit area limit is placed on the allowable extension of the dual-position nozzle to accommodate packaging of the seven engines on the vehicle base.

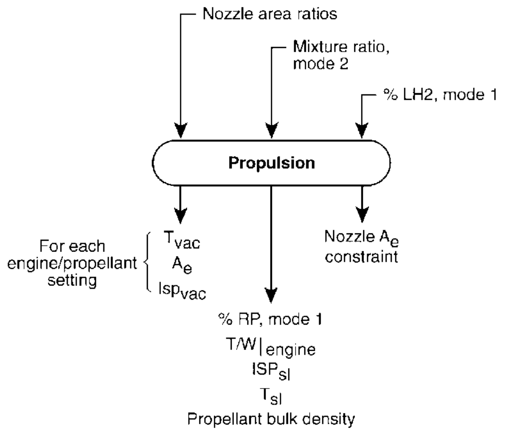

Fig. 1 Propulsion-analysis inputs and outputs.

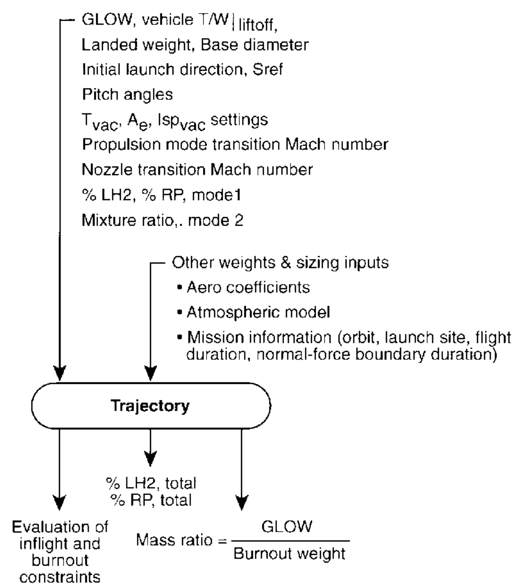

Fig. 2 Trajectory-analysis inputs and outputs.

Trajectory Analysis

To analyze the ascent flight path, a three-degree-of-freedom trajectory analysis is performed with the Program to Optimize Simulated Trajectories (POST).[25] Within POST, the equations of motion are numerically integrated from an initial to a terminal set of state conditions. Within the present investigation, the vehicle is modeled as a point-mass, Earth rotation and oblateness are treated, and the 1976 standard atmosphere (no winds) is used.

As shown in Fig. 2, the required set of trajectory inputs includes vehicle, e.g., GLOW, liftoff thrust-to-weight ratio, and aerodynamic coefficients and reference area, as well as trajectory parameters (pitch-angle history, launch azimuth, and the propulsion-mode and nozzle-transition Mach numbers). This domain-specific analysis is responsible for evaluation of the inflight and terminal constraints, computation of the vehicle mass ratio, and determination of the required total propellant mass fractions (weights and sizing inputs). Terminal constraints on altitude, velocity, and flight-path angle as well as maximum inflight dynamic pressure, angle of attack, pitch rate, and normal force (wing sizing constraint based on landed weight) limits are enforced.

Weights and Sizing Analysis

The Configuration Sizing (CONSIZ) program developed at NASA Langley Research Center is used to size the vehicle and determine subsystem weights. This sizing process is performed to meet vehicle mass ratio and landed wing-loading constraints. As shown in Fig. 3, the liftoff thrust to weight and mode-1 and total propellant mass fractions, as well as several propulsion system parameters, are required inputs to CONSIZ. Within CONSIZ, the mode-1 propellant mass fractions are needed in the propellant startup calculations, whereas the total propellant mass fractions are used in the mass ratio and tank sizing relations. Many of the CONSIZ inputs, e.g., the total propellant mass fractions and I_{sp} and T settings, are computed by one of the other two disciplinary analyses. In addition to the subsystem weights, which are required inputs to the cost analysis, CONSIZ computes the gross liftoff weight, reference aerodynamic surface area, landed weight, and base diameter (each of which is a required trajectory input).

Cost Analysis

As depicted in Fig. 4, the cost model used in the present investigation consists of two sets of cost-estimating relationships. The first set, technology maturation costs, provides an estimate of the investment required to advance a given subsystem from its present technology readiness level to a higher technology readiness level within five years (the presumed beginning of the development cycle). As presented in Fig. 5, the NASA technology readiness scale was used as a measure of subsystem maturity. Data from Ref. 24 were used to derive these technology maturation relationships. For this estimation, the vehicle subsystems are partitioned into nine segments: avionics, structures, electromechanical actuation, electrical conversion and distribution, auxiliary propulsion, prime power, main propulsion, propellant tanks, and thermal protection. The total technology maturation cost is defined as the sum of the cost of maturing these nine elements. In contrast to the study in Ref. 24, the present investigation does not require all of the technologies to reach a readiness level of 6 at the initiation of the development cycle. Instead, the total technology maturation funding is constrained to the level reported in Ref. 24, but an optimum subsystem maturity profile is determined as part of the design process.

The second set of vehicle-specific cost-estimating relationships was developed consistent with the subsystem breakdown structure of CONSIZ. These nonlinear parametric relations predict subsystem development cost as functions of weight, complexity, and technology readiness level. Additional relationships for subsystems that do not rely on advanced technology, e.g., landing gear, were included to obtain an estimate of the total vehicle development cost assuming a full-scale development program. The development cost relations were formulated through application of the Parametric Review of Information for Costing and Evaluation—Hardware (PRICE-H) multivariate model.[26] This model has been used to generate integrated cost and schedule estimates for numerous industry and government aerospace programs. The model provides a computational method for deriving cost estimates of electronic and mechanical hardware assemblies and systems.

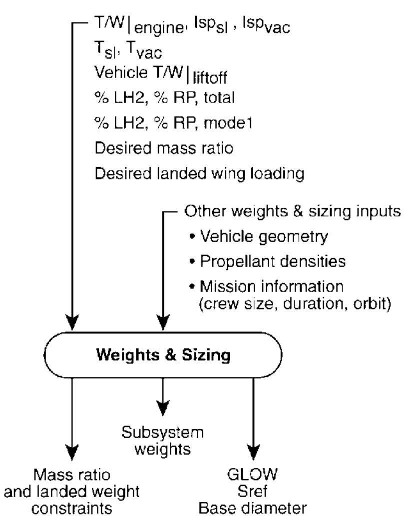

Fig. 3 Weights and sizing-analysis inputs and outputs.

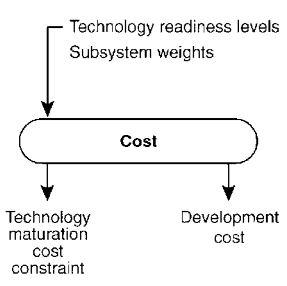

Fig. 4 Cost-analysis inputs and outputs.

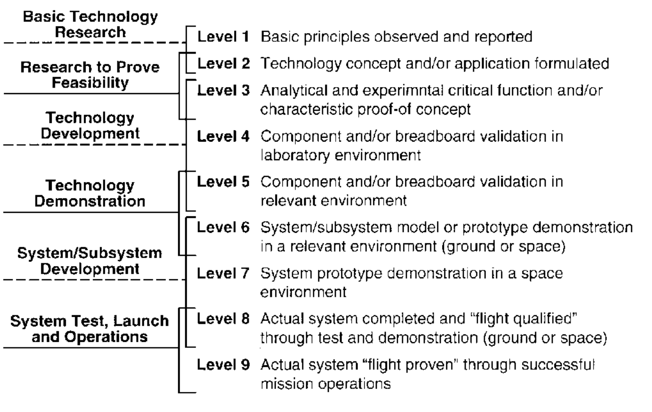

Fig. 5 NASA technology readiness levels.

Interdisciplinary Coupling

From Figs. 1–4, it is clear that solution of this problem requires an iterative approach because each analysis requires inputs that are computed by another discipline. For example, one must ensure that the reference aerodynamic surface area resulting from the vehicle sizing process (Fig. 3) is the same as the value used to compute the aerodynamic forces (Fig. 2). Interdisciplinary compatibility also must be achieved in regard to the GLOW, landed weight, base diameter, mass ratio, propellant mass fractions, and each of the propulsion discipline outputs. The liftoff thrust to weight, mode-1 propellant mass fractions, and mode-2 fuel-to-oxidizer-mixture ratio must be treated in a similar fashion because these parameters are input to more than one analysis. The interdisciplinary coupling structure of this launch vehicle design problem is shown in Table 1. Here, the I/O structure of the propulsion, trajectory, and weights, sizing, and cost subspaces is depicted. The weights, sizing, and cost analyses are grouped together because they are integrated within the same subspace in the collaborative architecture solutions.

Table 1 Interdisciplinary coupling present in launch vehicle design problem

Interdisciplinary variable	Propulsion	Trajectory	Weights cost
GLOW		input	output
Aero reference area		input	output
Landed weight		input	output
Base diameter		input	output
Vehicle liftoff T/W		input	output
$\%LH_2$, mode 1	input	input	input
%RP, mode 1	output	input	input
Mixture ratio, mode 2	input	input	
Vac. thrust, mode 1a	output	input	
Vac. I_{sp}, mode 1a	output	input	
Vac. thrust, mode 1b	output	input	
Vac. I_{sp}, mode 1b	output	input	
Vac. thrust, mode 2	output	input	
Vac. I_{sp}, mode 2	output	input	
Nozzle area, retracted (a)	output	input	
Nozzle area, extended (b)	output	input	
Mass ratio		output	input
$\%LH_2$, total		output	input
%RP, total		output	input
Engine sea-level T/W	output		input
Sea-level I_{sp}, mode 1	output		input
Allowable thrust ratio	output		input

Potential Optimization Architectures

Numerous optimization architectures have been proposed for launch vehicle design. In this section, some of the potential solution strategies are discussed. The design problem, as posed within the collaborative optimization architecture, is then presented.

The disciplinary tools used in this analysis were originally developed as stand-alone programs, each operated by a disciplinary expert. To obtain a credible vehicle, a design team was required to manually iterate among these disciplinary analyses. In many cases, "optimization" was performed through trade studies in which the parameters were varied one at a time.[27,28]

Improvement over this one-variable-at-a-time approach has been achieved using response surface methods (RSM).[9,20,21,29,30] In an RSM strategy, feasible designs are computed at numerous points in the design space and a surface is fitted to these points. Optimization then is performed on this approximate representation of the design space. Although use of RSM is a significant improvement over one-variable-at-a-time trade studies, the method suffers from several drawbacks including the requirement to produce numerous feasible design candidates and the limitation to problems characterized by a relatively small number of variables. For larger applications, as of interest in this work, the use of RSM is not a viable option.

Numerous gradient-based optimization architectures are possible for solution of this launch vehicle design problem. For example, system sensitivity analyses have been applied to a similar problem (cost issues not addressed).[31] Within the launch vehicle design community, the traditional multidisciplinary design optimization approach is to integrate the appropriate disciplinary analyses in a nonhierarchic iterative loop with a single, system-level optimizer.[32–36] In this standard optimization approach, interdisciplinary compatibility is enforced through some form of loop convergence criterion. As a result, numerous analysis evaluations are required to produce a single design candidate. All-at-once architectures also have been proposed for launch vehicle design.[23,36] Here, the iterative loop of the standard approach is removed through the use of auxiliary variables and compatibility constraints.[15,36,37] Use of a sequential-analysis, all-at-once optimization architecture may significantly reduce the required number of function evaluations.[36] In a subsequent section, the standard and all-at-once solution strategies[36] are compared to the use of collaborative optimization.

Collaborative Optimization

Figure 6 depicts the launch vehicle design problem in a form suitable for collaborative optimization. Here, the design is decomposed into three subspaces and coordinated by a system-level optimization procedure. Note that the weights, sizing, and cost issues are all accommodated within a single subspace. This decomposition strategy was selected because the weight and cost models are tightly coupled through numerous subsystem weights. For example, in the present model, development cost is a function of 17 subsystem weights.

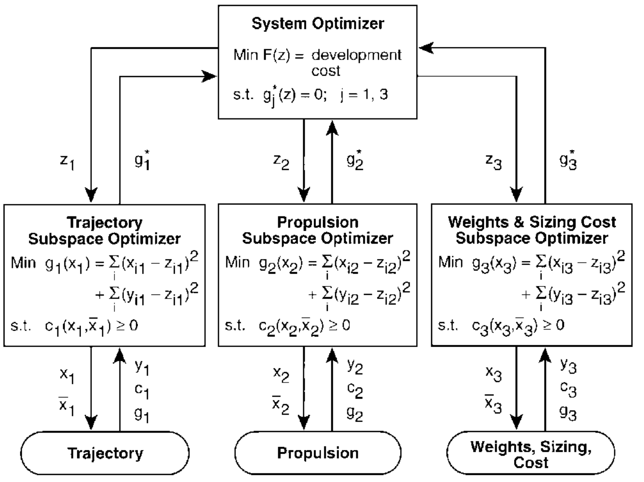

Fig. 6 Collaborative optimization architecture for launch vehicle design.

Table 2 Collaborative optimization launch vehicle design characteristics

Optimization level	Design variables	Constraints
System	23	3
Trajectory subspace	47	9
Propulsion subspace	4	1
Weights, sizing, cost subspace	21	3

Hence, in this case, analysis integration is useful in reducing the degree of interdisciplinary coupling.

Posed to suit the collaborative architecture, the problem is characterized by 95 design variables and 16 constraints. These design variables and constraints are partitioned among the system-level and subspaces as listed in Table 2. Whereas the nozzle exit area and technology maturation cost constraints are linear, each of the remaining 14 constraints is nonlinear.

When decomposed for solution with the collaborative optimization architecture, a single physical quantity may appear in the form of several optimization variables. For example, aerodynamic reference area is required as an input to the trajectory subspace, is computed in the weight, sizing, cost subspace, and is coordinated at the system level. In the notation of Fig. 6, this quantity is represented as x (interdisciplinary design variable), y (interdisciplinary output), and z (system-level design variable) in these respective optimization problems. Single-discipline variables, e.g., pitch angles in the trajectory subspace, are denoted as $\bar{x}$ in Fig. 6. The interdisciplinary compatibility functions g are used to ensure that a consistent design solution is achieved upon convergence of the system-level optimization problem. Further insight into the subspace and system-level formulations is provided elsewhere.[14,16]

Numerous collaborative solutions were obtained for this launch vehicle design problem. In each case, the subspace bounds for the interdisciplinary variables were not selected consistently. For example, in the trajectory subspace, the vehicle liftoff T/W was required to be between 1.0 and 1.3, whereas a range of 1.0–1.5 was placed on this interdisciplinary input within the weights, sizing, and cost subspace. Similarly, the percentage of LH_2 in mode 1 had bounds of 0–15, 3–20, and 3–15 within the various subspaces. These bound inconsistencies were included to allow for conflict and to simulate the architecture's application in a team environment where the various disciplinary specialists are each responsible for a particular domain-specific subproblem.

Prior to optimization, the subspace and system-level problems were scaled such that the design variables, constraints, and objective function were of order 1. However, in this investigation, no attempt was made to provide a twice-continuously differentiable model. A majority of the sources that contribute nonsmoothness are within the trajectory subspace—the ascent pitch profile, which is modeled by discrete control points connected by linear segments, as well as tabular linear interpolation of the aerodynamic and atmospheric properties (1976 standard model).

As an example of the collaborative convergence process, Fig. 7 depicts the interdisciplinary negotiation that occurs between the trajectory subspace, weights-sizing-cost subspace, and system level on the appropriate value of the wing aerodynamic reference area (S_{ref}). Figure 7 shows the domain-specific convergence histories for three system-level iterations. In the first iteration, the system-level proposes an S_{ref} value of 4000 ft^2. After meeting all of its domain-specific constraints, the trajectory subspace returns with a request to change this value to 3950 ft^2. Similarly, the weights-sizing-cost subspace returns with hopes of increasing this value to 4100 ft^2. Based on gradient information provided at the solution of each subspace, a system-level step is taken that alters the system-level value S_{ref} to 3955 ft^2 (and the value of the other 22 system-level variables). In this case, the subspaces are able to remain disciplinary constraint feasible while providing better agreement on this value of S_{ref}. In the third iteration, the value of S_{ref} is reduced even further with consent from the two domain-specific analyses.

Through repeated collaboration, the system-level optimizer orchestrates the interdisciplinary compatibility process. Note that,

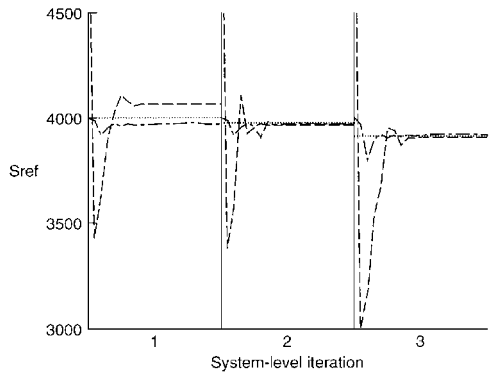

Fig. 7 Interdisciplinary negotiation within the collaborative optimization architecture: — — —, weights-sizing-cost subspace; —·—, trajectory subspace; and ······, system-level target.

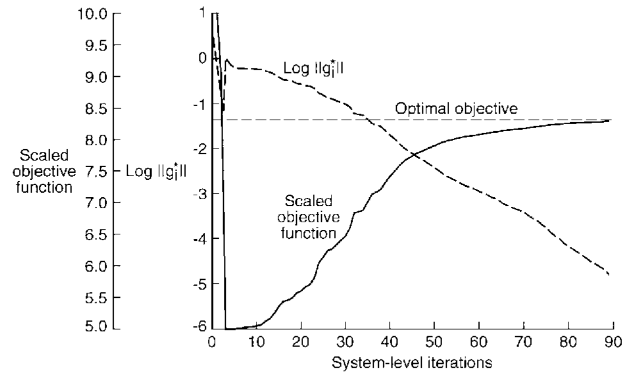

Fig. 8 Collaborative optimization system-level convergence profile: minimum-development-cost concept.

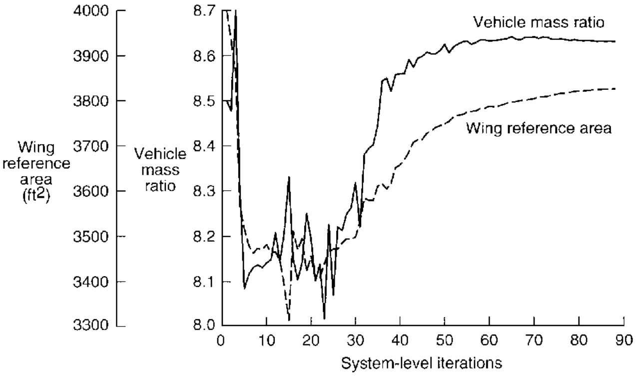

Fig. 9 Collaborative optimization convergence history: system-level values of vehicle mass ratio and wing aerodynamic reference area.

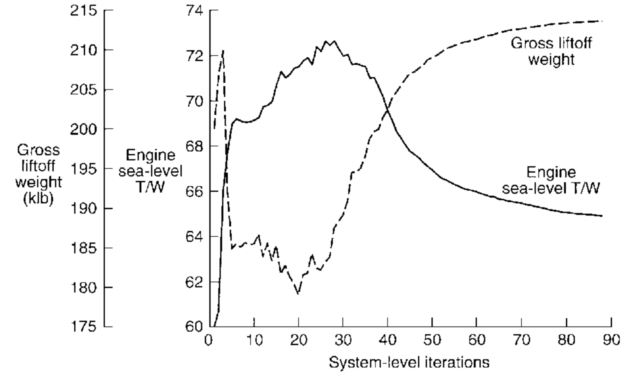

Fig. 10 Collaborative optimization convergence history: system-level values of main engine sea-level *T*/*W* and vehicle GLOW.

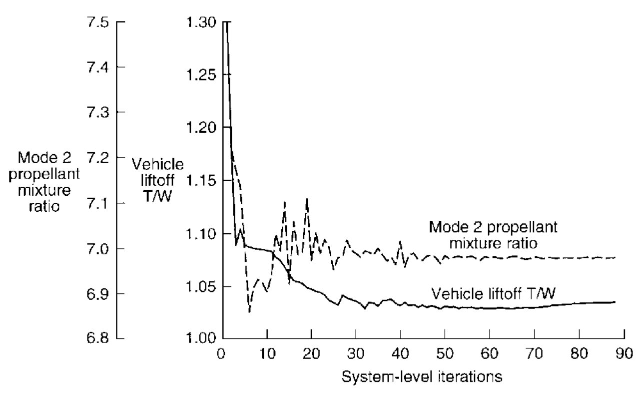

Fig. 11 Collaborative optimization convergence history: system-level values of vehicle liftoff *T*/*W* and mode-2 mixture ratio.

in Fig. 7, cold-start subspaces were used for illustrative purposes. Hence, within each system-level iteration, subspace optimization was initialized from a fixed, domain-specific point (an S_{ref} of 4000 ft^2 in the trajectory subspace and an S_{ref} of 5000 ft^2 in the weights-sizing-cost subspace). In the results that follow, subspace optimization is initialized from the prior solution (warm start) to reduce the computational requirements of repeated subspace optimization.[14,16]

Application of Collaborative Optimization Architecture

In this section, optimal solutions of several versions of the basic launch vehicle design problem are presented. Each case was obtained through application of the collaborative optimization architecture. To obtain these solutions, the refined architecture of Braun[14] was used. This implementation includes use of the linear system-level objective refinement as well as the warm-start and slack-variable subspace refinements.[14] The system-level Jacobian is obtained from the use of postoptimality information estimated at subspace solutions.[14,16,38] The sequential quadratic programming algorithm NPSOL[39] was used to provide system-level and subspace optimization. Central processing unit times quoted are based on use of a Silicon Graphics Challenge L machine outfitted with R8000, 90-MHz processors.

Minimum-Development-Cost Concept

The collaborative optimization convergence history for this solution is presented in Fig. 8. The 89 system-level iterations shown required 181 subspace optimizations of each of the 3 domain-specific analyses. This solution compares favorably (within 0.1% in the system-level objective function) with a solution previously obtained with an all-at-once optimization strategy.[23] The convergence profile illustrated in Fig. 8, in which the system-level objective initially drops and then approaches the solution from below as system-level feasibility is achieved, is similar to collaborative solutions obtained for other problems.[14]

To obtain this solution, 181 sets of subspace optimization were performed. Initialized from a fixed initial guess (cold start), a call to the subspaces requires approximately 1–3 h. At this rate, a cold-start solution would require approximately 1–3 weeks. Using warm-start subspaces (restarting from the previous domain-specific solution with knowledge of the prior optimum active set, Lagrange multipliers, and Hessian of the Lagrangian), the solution presented in Fig. 8 required approximately 4.5 days of computer time. Hence, warm-starting the subspaces provides a dramatic advantage. This level of efficiency gain is possible because the subspaces are tasked with solving a related sequence of subproblems.

Figures 9–11 demonstrate the convergence behavior of six of the interdisciplinary variables. These figures depict the system-level values of the vehicle mass ratio, wing aerodynamic reference area, main engine sea-level T/W, vehicle GLOW, vehicle liftoff T/W, and mode-2 mixture ratio. As shown, the system-level variables tend to exhibit either the same behavior as the system-level objective or a more damped oscillation about the optimal value.

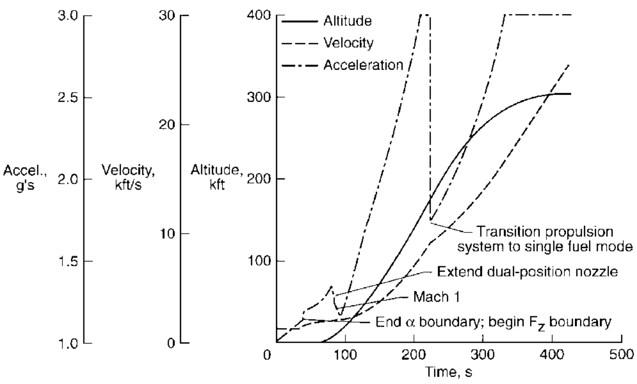

Fig. 12 Optimal flight profile (altitude, velocity, and acceleration).

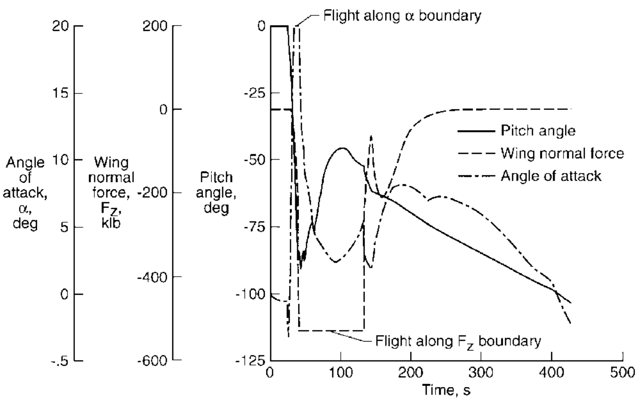

Fig. 13 Optimal pitch-angle and inflight constraint profiles.

The flight-path portion of this optimal solution is illustrated in Figs. 12 and 13. The ascent begins with a 400-ft vertical rise to clear the launch facility. Because the vehicle is characterized by a low liftoff T/W (1.036), this rise takes approximately 20 s. This vertical-flight segment is followed by a maximum pitch-rate segment in which the vehicle tries to attain its maximum-lift orientation. The pitch rate is limited to 5 deg/s to reflect control issues that are not modeled in this analysis. As shown in Fig. 13, during this segment of flight, the angle of attack α increases until reaching its maximum allowable value of 20 deg. Flying at this α, the normal force builds until the limit load (2.5 times the landed weight) is reached. The vehicle rides this normal-force boundary through peak dynamic pressure, which, for the optimum flight path, is about 990 psf. Hence, at this solution, the dynamic pressure limit of 1000 psf is not active. During this phase of flight at approximately 10,000 ft, the backpressure losses are low enough that the dual-position nozzle is extended to gain propulsive efficiency. As the nozzle extension is performed, the vehicle acceleration initially decreases as a result of flight through the transonic regime. At about 50,000 ft, as the dynamic pressure decreases, the vehicle comes off the normal-force boundary but continues to accelerate toward 3 g. The vehicle reaches 3 g while in the dual-fuel propulsive mode and uses engine throttling to maintain this level of acceleration. In this analysis, transition of all seven engines from a dual-fuel to a single-fuel mode is performed instantaneously. This instantaneous change in thrust results in the large decrease in acceleration shown in Fig. 12 at roughly 225 s and an altitude of about 175,000 ft. Operating in a single-fuel mode (LH_2), the vehicle accelerates back to 3 g and holds this acceleration level until reaching orbit. Note that, if the nozzle transition had been performed sequentially, a slightly better result could have been achieved.

This minimum development cost configuration is sketched in Fig. 14. The vehicle is characterized by a dry weight of 1.831×10^5 lb, a GLOW of 2.139×10^6 lb, and a length of 182.1 ft. Note that the optimum vehicle T/W is 1.036 at liftoff. This T/W level, which is significantly lower than the 1.2 value typically observed, is the subject of the following section.

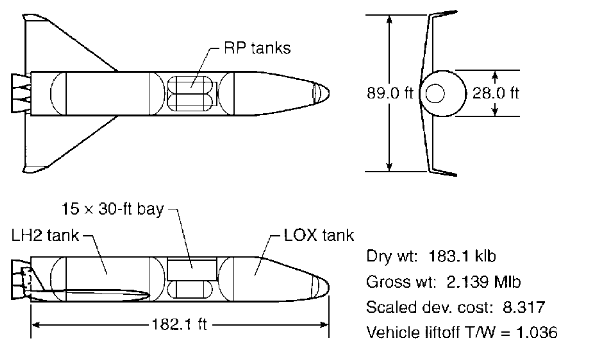

Fig. 14 Minimum development-cost vehicle description: no low-altitude ascent abort constraint.

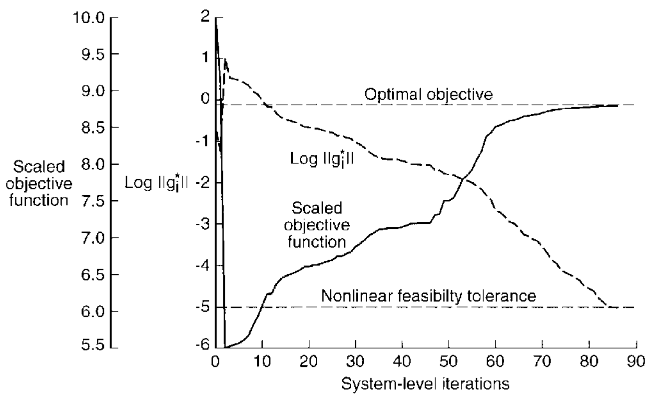

Fig. 15 Collaborative optimization system-level convergence profile: minimum development cost concept (vehicle liftoff $T/W \geq 1.2$).

How Many Engines?

With the system-level vehicle liftoff T/W lower bound increased to 1.2, the collaborative optimization architecture was rerun, giving the system-level performance profiles shown in Fig. 15. Here, 86 system-level iterations are required to obtain system-level convergence. Comparison of the solutions presented in Figs. 8 and 15 demonstrates that requiring a vehicle liftoff $T/W \geq 1.2$ induces a 4.5% increase in development cost.

Generally, a lower bound on the vehicle liftoff T/W is included at the preliminary design stage to provide sufficient engine-out capability during the first moments of ascent (vehicle T/W increases with time). In this case, 1.2 was chosen because the vehicle was designed with seven engines. Hence, with loss of a single engine at liftoff, the vehicle would still have propulsive control authority—the T/W would become 1.03 (Ref. 40). Inclusion of this engine-out capability is the primary reason that many launch vehicle design concepts have a liftoff T/W of 1.2 (Refs. 7, 9, 20, 21, 31, and 36). Unfortunately, this increased flexibility does not come without a price, as the minimum-development-cost concept would prefer a lower liftoff T/W.

To further complicate this issue, the number of engines is a significant driver on vehicle cost and reliability. As the number of engines increases, flight reliability concerns also increase. To maintain a certain level of system reliability, e.g., 99%, each engine in a seven-engine cluster must have a reliability of 99.86%. With just three engines, this individual engine reliability is reduced to 99.66%. Because reliability is a significant cost driver,[1,6] low-cost concepts such as those envisioned within the Reusable Launch Vehicle program[11] may not have the luxury of including this engine-out abort constraint. For example, with three engines, incorporation of a liftoff engine-out capability induces a severe penalty because the lower bound on the allowable liftoff T/W would be increased to 1.5. In this case, this liftoff engine-out requirement may have to be discarded. Although this is the philosophy selected in the single-stage-to-orbit designs of the present investigation, it is clear that this issue requires further study.

Objective Selection in Launch Vehicle Design

The concept of a "best" or optimal configuration has different meanings in different design groups. In launch vehicle design, minimum weight concepts traditionally have been sought. In some cases, liftoff gross weight is selected as the minimization variable.[27,28] With the realization that propellant is relatively inexpensive, a majority of the recent concepts have been designed to minimize dry weight.[7,9,20,21,31,36] Often, these minimum-dry-weight studies include a claim such as, "Since vehicle development costs tend to vary as a function of dry weight, this minimum dry weight vehicle may be considered a minimum development cost concept." However, as demonstrated in this section, such an assertion is not rigorously true, even when a weight-based cost model is used. Another objective function that has been suggested within the launch vehicle design community is minimum ΔV. This objective is sometimes chosen on the basis of application of the "rocket equation." However, as shown in this section, minimum ΔV concepts are vastly different from either minimum-development-cost or minimum-weight designs.

Distinctions among these four different objectives (gross weight, dry weight, development cost, and ΔV) are examined through application of the collaborative architecture. In each of these designs, the vehicle liftoff T/W is allowed to vary in the range 1.0–1.5. The minimum development cost solution, described in the preceding section, is used to normalize the other optimal results.

Design characteristics of these four optimal concepts are listed in Table 3. As shown by column 4, at the vehicle level, the minimum ΔV concept stands apart from the other three designs. This system is over 50% more expensive than the minimum-development-cost concept and more than 25% heavier than the minimum-dry-weight case (while yielding only a 4–6% improvement in ΔV). Furthermore, the vehicle's wing area is more than 20% larger than each of the other wings. With the exception of this concept, the other three concepts share a top-level similarity, the strongest correlation being between the minimum-development-cost and minimum-dry-weight configurations. However, even among these first three concepts, the maximum variance in dry weight is 3.5%, the maximum variance in development cost is 4.0%, and the maximum variance in GLOW is 6.7%.

The optimal liftoff T/W and mass ratios for each of the four concepts also are listed in Table 3. From the liftoff T/W comparison, it is clear why the minimum ΔV concept is so different. In this case, the optimization process attempts to minimize the ascent losses, e.g., gravitational, aerodynamic, nozzle backpressure, and thrust vectoring, by achieving orbit as quickly as possible. As shown in Fig. 16, to accomplish this, the propulsion system size is increased greatly and a larger RP propellant mass fraction (%LHC) is required. In this case, the vehicle liftoff T/W is at its upper bound of 1.5. Had this variable been allowed to increase even further, the vehicle would have continued to grow, reaching enormous proportions.

Figure 16 shows that subsystem differences are not limited to the minimum ΔV concept. For example, although a dual-fuel system is allowed, the minimum-gross-weight concept prefers LH_2 as its sole fuel. In this case, the dense RP propellant is eliminated such that I_{sp} is increased and the initial launch weight is reduced. This disregard for bulk density is in stark contrast to the minimum-dry-weight case, which relies on bulk density to provide dry-weight savings.[21] The goal of minimum GLOW also induces an 8.2% increase in vehicle liftoff T/W and a 10.6% decrease in vehicle mass ratio (see Table 3) relative to the minimum-development-cost system.

Table 3 Optimal launch vehicle comparison

Normalized design characteristics	Objective function: Development cost	Dry weight	Liftoff weight	ΔV
Development cost	1.0	1.001	1.040	1.510
Dry weight	1.0	0.987	1.035	1.262
Liftoff weight	1.0	1.007	0.925	1.196
ΔV	1.0	0.999	0.987	0.944
Body length	1.0	0.993	1.030	1.052
Wing area	1.0	0.993	1.043	1.239
Liftoff T/W	1.0	1.004	1.082	1.448
Mass ratio	1.0	1.008	0.893	0.959

Table 4 Comparison of three multidisciplinary optimization strategies for launch vehicle design

Optimization architecture	Function evaluations	Modification time, mo	Communication requirements
Standard approach[36]	10,482	4	66
Collaborative optimization	3,125–24,840	1	23
All-at-once[36]	3,182	3	65

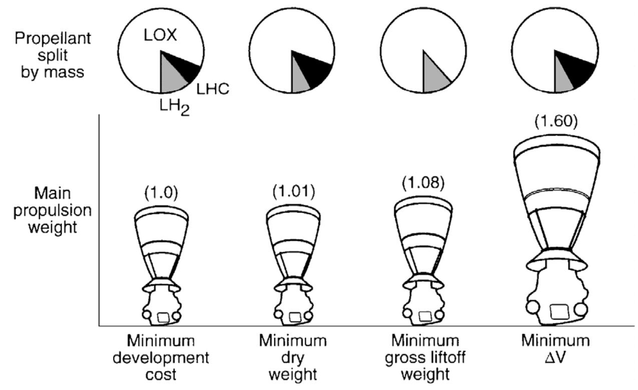

Fig. 16 Optimal launch vehicle comparison: main propulsion system.

Figure 16 also shows that the minimum-development-cost and minimum-dry-weight concepts are not equivalent. This finding is of even greater significance when one considers that the present analysis relies on a weight-based cost model. For example, relative to the minimum-dry-weight design, the minimum-development-cost concept uses 16% less RP (LHC). In this case, the minimum-development-cost concept seeks to reduce propulsion system mass because this subsystem is characterized by both a high development cost per pound and technology maturation cost. In essence, the minimum-development-cost case may be thought of as a "weighted" minimum-weight case, where all pounds are not created equal.

The results shown in this section demonstrate the importance of appropriate objective function selection. In particular, although minimum-weight and minimum-cost concepts may be similar, they certainly are not equivalent (even when a weight-based cost model is used). Differences at the subsystem level were shown to reflect the goal of the optimization process. Hence, if one is truly interested in developing minimum-cost concepts, these considerations must be factored into the preliminary design process prior to optimization.[22,23]

Operational Aspects of Architecture Selection

The computational performance of the collaborative optimization architecture has been shown to improve as the size of the domain-specific problems is increased without a significant increase in interdisciplinary coupling.[14] Hence, using computational performance as a yardstick, application to large-scale, loosely coupled problems has been suggested.[14] In a multidisciplinary design environment, computational performance is only one aspect of architecture selection. Often, in such a setting, analysis integration and communication requirements are much harder to resolve efficiently. In addition, system flexibility, analysis modularity, and resource management are also concerns. For example, if it requires a year to set up the appropriate multidisciplinary analysis system, run time is certainly inconsequential. Furthermore, if a system is integrated in a manner such that it is difficult to modify or extend, it is not likely to have a long life within the design organization. As the number and fidelity of the domain-specific analyses increase, these operational concerns become more significant.

Based on the results of this paper and other results[36] (where cost issues were not addressed), Table 4 presents the authors' experience solving a similar launch vehicle design problem with three optimization architectures. This table lists the function evaluations required to reach the solution from a common initial guess, the estimated analysis modification time, and the interdisciplinary communication requirements for each of the three optimization architectures. As a result of the variation in subspace problem sizes (see Table 2), the propulsion subspace required far fewer function evaluations than the trajectory subspace. This is the cause for the range of numbers listed in the function evaluation column of Table 4 for the collaborative architecture. Table 4 shows that, although in its present form the collaborative architecture is not computationally competitive with the all-at-once optimization architecture, it is generally competitive with the standard optimization approach. A similar conclusion also has been demonstrated in the solution of large-scale trajectory optimization problems.[14,16] Note that, because Table 4 lists function evaluations, the potential parallelization speedup inherent to the collaborative and the all-at-once approaches is not reflected.

Use of the collaborative architecture offers numerous other advantages. The largest of these is the estimated analysis modification time. Recall that this analysis is based on the modification of a set of previously stand-alone disciplinary programs. In such a case, the standard and the all-at-once approaches require a much higher degree of analysis modification. For the standard optimization approach, this time is spent providing integration of the disciplinary analyses. Because this integration was performed in an "engineering" sense, the resulting integrated system is not extremely flexible.

In the case of the all-at-once system, significant time was spent preparing the analyses for optimization. In a practical design environment, many disciplinary analyses are simply not set up for efficient optimization (design oriented). For example, in its original form, the trajectory analysis was not well suited for function evaluation. Instead, trajectory evaluation was explicitly coupled with optimization. Whereas the original analysis program was simply inserted into the collaborative architecture, significant time was spent providing a trajectory function evaluator for use with the other approaches.

Communications requirements of the collaborative architecture are also minimal (see Table 4), resulting solely from the interdisciplinary coupling inherent in the problem. In contrast, both the standard and the all-at-once approaches required significantly more communication. In the standard approach, 36 variables were passed between the optimizer and the set of multidisciplinary analyses. The remaining communication requirements involved coordination among the analyses. Similarly, in the all-at-once strategy, 40 variables were shared between the optimizer and the analyses, whereas 25 variables were passed among the analyses. In contrast, the collaborative architecture only requires communication of 23 variables between the system level and the subspaces. This decreased level of communication is a direct result of empowering the subspaces with domain-specific design responsibility.

In a multidisciplinary design environment, use of the collaborative architecture provides additional operational advantages as a result of its flexibility and modularity. Within this architecture, both analysis and optimization may be performed in a distributed manner. In fact, the distributed aspects of the architecture have been demonstrated on a set of heterogeneous computing platforms.[15] Furthermore, in collaborative optimization, the required decomposition into a set of coordinated subspaces necessitates a coarse-grained modularity. This modularity allows the system to be easily extended and modified. Over the lifetime of a design project, this flexibility may be used to adjust the fidelity of the disciplinary models. In addition, changes to one domain-specific analysis may be altered without impacting the rest of the system. In contrast, the standard optimization approach provides very little flexibility and was found to be difficult to modify, whereas the all-at-once approach only provides a distributed analysis capability.

Summary

In this investigation, the collaborative optimization architecture is used to perform multidisciplinary design of a dual-fuel, single-stage-to-orbit launch vehicle. Vehicle design, trajectory, and cost aspects are addressed directly. Posed to suit the collaborative architecture, this design problem is characterized by 95 design variables (23 interdisciplinary) and 16 constraints. A minimum-development-cost concept is obtained that compares favorably (within 0.1% in development cost) to a result produced by another optimization architecture. The influence of an a priori ascent-abort criterion on development cost and proper objective-function selection is discussed. Differences are highlighted between the minimum cost, weight, and ΔV concepts.

The operational aspects of the collaborative architecture in a multidisciplinary design environment are presented and compared with two other optimization architectures. Relative to these other optimization strategies, the advantages of the collaborative architecture include minimal framework integration requirements, the ability to use domain-specific analyses that already provide optimization without modification, inherent system flexibility and modularity, a distributed analysis and optimization capability, and a significant reduction in communication requirements. These practical advantages make the architecture well suited for use in a large-scale, multidisciplinary design environment.

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J. A. Martin
Associate Editor