JOURNAL OF SPACECRAFT AND ROCKETS
Vol. 45, No. 3, May–June 2008

Evaluating Technology Projections and Weight Prediction Method Uncertainty of Future Launch Vehicles

Alan W. Wilhite*
Georgia Institute of Technology, Atlanta, Georgia 30332
and
Sampson E. Gholtson,† Phillip A. Farrington,† and James J. Swain‡
University of Alabama in Huntsville,
Huntsville, Alabama 35899

DOI: 10.2514/1.30578

A process was developed for determining the impact of technology performance assumptions and weight prediction method uncertainty. Weight and performance uncertainties were defined for components from historical weight-estimating relationships that are typically used during the concept definition phase. A systems analysis model was developed that sizes vehicle geometry, propellant, and component weights to meet mission requirements. The uncertainties and system analysis model were integrated with a Monte Carlo simulation to determine the uncertainty probability on system weight. These uncertainties were integrated into the analyses of single-stage and two-stage reusable launch concepts to demonstrate the technology uncertainty influence on concepts having different gross weight sensitivities to component weight changes. Finally, this process was extended as a model for measuring the progress of technology development programs.

Nomenclature

f	=	vehicle dimension scaling factor
g_0	=	9.80665 m/s^2
I_{sp}	=	specific impulse, thrust/(g_0 · propellant mass burn rate)
T	=	thrust
W	=	weight
η	=	body volume packaging efficiency factor

I. Introduction

SINGLE-STAGE-TO-ORBIT (SSTO) and two-stage reusable launch vehicle concepts were analyzed for the effects of uncertainties in component weights, engine performance, aerodynamic drag, and available volume for propellant tanks. The mission is to deliver 25,000 lb to the International Space Station.

The SSTO concept lifts off vertically and lands horizontally like the space shuttle [1]. As shown in Fig. 1, this concept is a wing–body configuration with a circular body cross section, trapezoidal wing, and active fins on the wing tip for lateral-directional control. To achieve single-stage performance, advanced technologies are employed, such as graphite epoxy body, wing, fin, and fuel tank, aluminum lithium oxidizer tank, external metallic thermal protection system as employed on the X-33, advanced space shuttle main engines, hydrogen/oxygen fuel cells, and electrical-mechanical actuators. The payload bay and flight deck are located between the forward oxygen tank and aft hydrogen tank.

The two-stage concept, called a Bimese, was designed by the mating and sizing of two identical SSTO configurations for the two-stage mission mode. The Bimese dramatically reduces development cost of a two-stage system with the utilization of identical stages, and improves the part production quantity for a very small fleet size because both orbiters and boosters have to be produced and maintained. The concepts are mated belly to top to minimize wing interference aerodynamic thermal and pressure loads. The booster and orbiter engines are burned in parallel to minimize the number of engines employed, and propellant is cross fed from the booster to the orbiter resulting in a full orbiter at staging.

Received 26 February 2007; revision received 27 November 2007; accepted for publication 27 November 2007.

*Langley Professor, Daniel Guggenheim School of Aerospace Engineering, 270 Ferst Drive. AIAA Associate Fellow.

†Associate Professor, Industrial and Systems Engineering Department.

‡Department Chair/Professor, Industrial and Systems Engineering Department.

II. Analysis

To conduct the uncertainty study, a system weights and sizing model was developed called Launch Vehicle Sizer and Synthesis (LVSS). This systems analysis model was integrated with the @RISK Risk Analysis and Simulation vendor software§ for the uncertainty analysis.

A. System Weights and Sizing

The LVSS system was implemented on a spreadsheet and consists of four distinct modules. The first module is the input for frequently changed variables that include ascent requirements such as mass ratio (initial weight/injected weight) and thrust-to-weight ratio, orbital maneuvering delta-velocity requirements, propulsion parameters, geometry, etc. Ascent requirements are scaled from the baseline concepts [1] using the rocket equation.

The second module is a propellant and body volume model that computes required propellant volume and weight based on body tank packaging, ullage factors, oxidizer-to-fuel ratio, and propellant densities. The required ascent propellant weight is determined by the required mass ratio to meet the mission performance, the mass fraction computed from the component weight-estimating relationships, and the payload. Reserves, residuals, and in-flight losses are modeled as percentages of the ascent propellant (0.1, 0.4, and 0.3%, respectively) [2]. The volume of the liquid hydrogen (LH) and liquid oxygen (LOX) tanks are computed based on the oxidizer-to-fuel ratio (6:1), propellant densities (4.4 and 71.2 lb/ft^3), and a 4.5% ullage volume fraction of the total volume. For sizing the vehicle to meet mission requirements for a fixed payload weight, vehicle external and internal geometry (structure) is scaled photographically based on the change in required body volume to accommodate the new tank

§@Risk, http://www.palisades.com/.

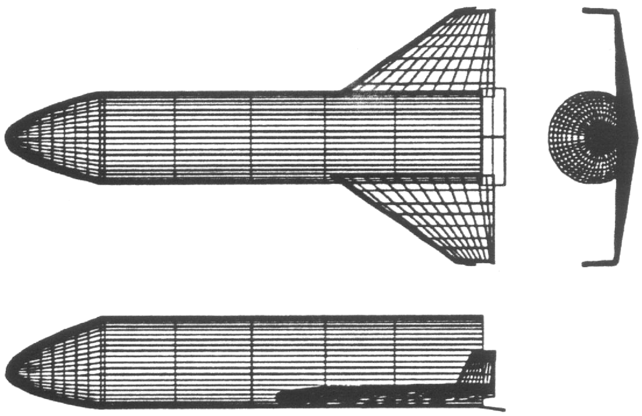

Fig. 1 SSTO concept [1].

volumes, the fixed payload volume, and a volumetric efficiency factor that accounts for the engines, subsystems, and unused volume. Thus, the dimensions of the vehicle are scaled by the factor f, defined by

$$f = \left[\frac{V_{\mathrm{body_{new}}}}{V_{\mathrm{body_{old}}}}\right]^{\frac{1}{3}} = \left[\frac{(V_{\mathrm{LHtank}} + V_{\mathrm{LOXtank}} + V_{\mathrm{payload}}) \cdot (1 + \eta_{\mathrm{packaging}})}{V_{\mathrm{body_{old}}}}\right]^{\frac{1}{3}}$$

The third module computes vehicle component weights. Dry weight components are defined using weight-estimating relationships such as the wing and rocket engine weights shown in Fig. 2. Propulsion engine thrust is sized to meet the initial thrust-to-gross weight requirement of 1.3; the engine weight is proportionally scaled to thrust using the method in [3] and shown in Fig. 2. Residuals, in-flight losses, and reserve propellants are modeled as fractions of the main propellant. Orbital and reaction control propellants are computed using the rocket equation, propulsion performance, and delta-velocity mission requirements. Because several weights depend on other weights (wing weight is a function of empty weight and empty weight is a function of wing weight), the weights module is iterated until convergence of gross weight.

The fourth module controls sizing to meet mission requirements. There are three sizing options. The first option only computes the component weights with no sizing and is used for verification of the component weights and volume models. The second option computes the required payload and propulsion for the defined vehicle, thrust-to-weight, and mission mass ratio performance. The third option sizes the propellant, geometry, and components to meet a fixed payload using the propellant and volume module. Because the component weights depend on geometry and/or other weights, the propellant weight and volume, engine thrust, vehicle external and tank geometry, and weight must be iterated until the mass ratio and thrust-to-weight requirements are met for the fixed payload weight.

B. Uncertainty Analysis

To determine the effect of component weight uncertainties, the @RISK software§ was integrated into the LVSS model. The @RISK software uses Monte Carlo simulation to compute the probabilistic outcome based on the technology uncertainties and LVSS model results.

Examples of the weight-estimating relationships are shown in Fig. 2 for the wing and engine. A least-squares regression analysis defines the trendline equations, and the uncertainty range is defined by the upper and lower bounds of the data relative to the trendline, as shown in Fig. 2. The weight-estimating relations for all the components are in [4]. The bounding errors of the historical data analysis show significant errors in several components, as shown in Table 1. These data are integrated into the system analysis to determine the sensitivity of the component errors and the overall system sensitivity for both single and two-stage launch vehicles.

Because of a lack of historical data to compute several key uncertainties, a ±30% uncertainty was assumed for the thermal protection system (TPS), propulsion system, and subsystems based on NASA's recommendation of a 25 to 35% growth factor for prephase A studies of spacecraft as shown in Table 2 [5].

Table 3 shows the actual Space Shuttle orbiter growth for phases C–D that resulted in an integrated dry weight growth of 12%. This growth does not include the 12.6% dry weight planned margin at the beginning of phase C. At the beginning of the program, the orbiter was baselined to incorporate current technology for all vehicle components, except for the thermal protection system and the space shuttle main engine, to control weight growth. As weight grew, advanced technologies such as composites were strategically used to control weight and maintain payload. However, even with the growth margin in the program and applying advanced technology over the initial baseline, the space shuttle payload weight had to be reduced by one-third to accommodate the total system launch weight requirements.

Three other uncertainties were also included in the present study. A range of ±5% for engine specific impulse (space shuttle main engine missed by 2.5 s), ±15% for aerodynamic drag unknowns in viscous drag, base drag, and real gas effects, and ±5% on available body volume to account for error in packaging the subsystems at this conceptual level where the subsystem volumes are based on unsized space shuttle subsystems.

With the uncertainties defined, @Risk was used to fit the distribution functions. Thirty-seven distributions were available (three are shown in Fig. 3). The uniform distribution was chosen for this study because it represents a random distribution of the uncertainty data. The data could be fit with normal distributions, but they are difficult to integrate with the weight system sizing because the distribution limits are ± infinity, which may cause the analysis to have numerical instabilities during the Monte Carlo simulation. Fitting the uncertainty data to other finite distributions, e.g., triangular, should be conducted in the future to compare changes in the results.

A Monte Carlo simulation was run in the vehicle-sizing model using the defined distribution functions. The simulation

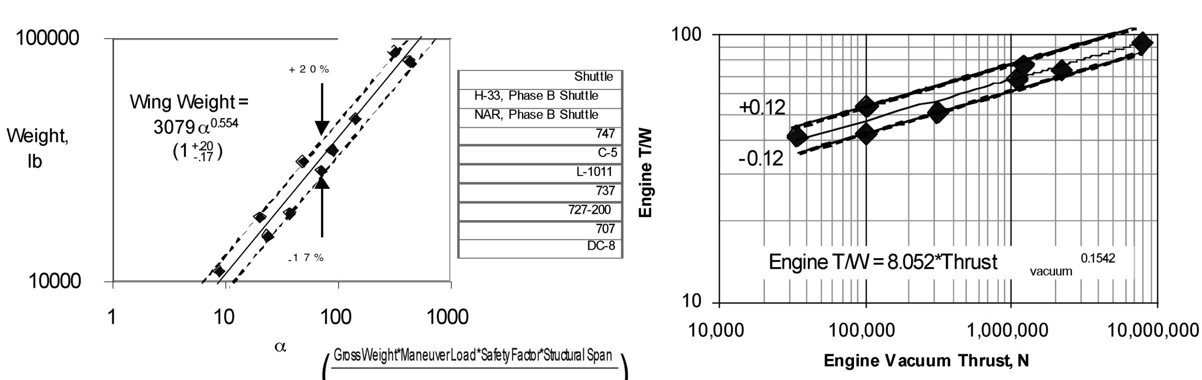

Fig. 2 Example of empirical wing [2] and engine [3] weight estimation and uncertainties

Table 1 Component technology reduction factor and weight uncertainties

	Baseline technology factors	Weight estimation uncertainty bounds	
Wing	0.20	−0.17	0.20
Tail	0.20	−0.70	1.06
LH2 tank	0.15	−0.18	0.41
LOX tank	0.15	−0.51	0.49
Body	0.20	−0.36	0.64
Gear	0.15	−0.15	0.21
Propulsion	0.05	−0.12	0.12
TPS	0.05	−0.30	0.30
Subsystems	0.20	−0.30	0.30

Table 2 NASA design margins for spacecraft

Prephase A	25–35%
Phase A	25–35%
Phase B	20–30%
Phase C	15–25%

Table 3 Shuttle weight growth

Space shuttle growth phase C/D (1972-1983)	
Wing	0.27
Tail	0.14
LH tank	0.13
LOX tank	0.13
Body	0.03
Gear	0.06
TPS	0.01
Propulsion	0.12
Subsystems	0.50
I_{sp}, s	−2.5

randomly samples each of the distribution functions and executes the vehicle-sizing model. After several trial simulations, it was determined that similar results were produced with approximately 800 iterations.

III. Results

A. Technology Effects on SSTO and Bimese Concepts

The effects of vehicle weight growth (or performance shortfalls of advanced technology) on future launch vehicles is modeled simply as percentage dry mass weight increases over the baseline (Table 1). The overall impacts on the gross weight and dry weight (gross weight minus payload and ascent propellant) are shown in Figs. 4a and 4b. These results were generated by multiplying each of the component weights by one plus the technology factor and sizing the concept and component weights to meet mission requirements (see Sec. II.A, System Weights and Sizing).

Figure 4a shows that the Bimese concept is significantly lower in gross weight than the SSTO concept because of the staging effects and also is significantly less sensitive to changes in technology weight factor. In Fig. 4b, the Bimese dry weight is still less sensitive to technology changes. This reduced weight sensitivity is key for the Bimese to tolerate design uncertainties.

However, the dry weight for the Bimese is unexpectedly close to the SSTO at the baseline technology factor of zero. This closeness of dry weight is due to the reduction of system efficiencies, such as tank packaging, with the resulting smaller stages and the duplication of many component weights in the booster, such as crew, crew provisions, payload bay, and other subsystem weights that do not size down as vehicle size and weight are reduced. Also, the assumed technology weight reductions in the baseline vehicles (Table 1) could be optimistic if the initial operational capability date is less than 20 years from today.

B. Uncertainty Analysis

Figure 5 is a Tornado chart showing the sensitivity of each of the variables based on an integrated average of the uncertainty times the impact of the variable. Propulsion is ranked number one because it is the largest weight component (27% of dry weight as shown in Fig. 5) and has an above-average uncertainty of ±30%. This risk

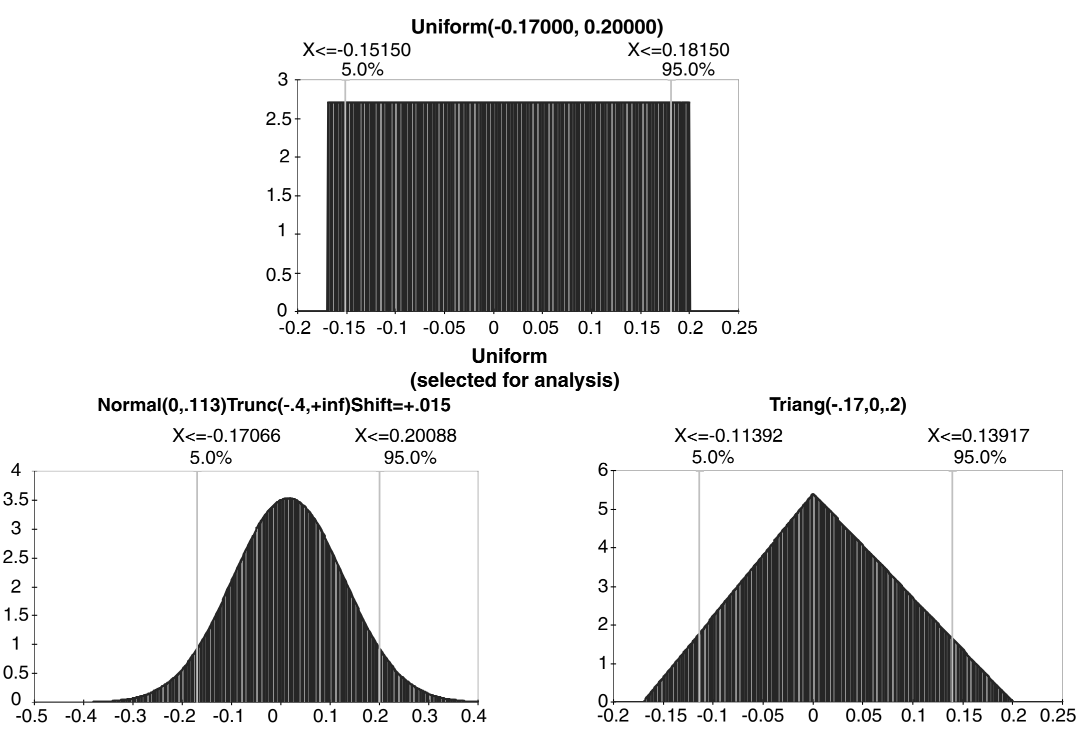

Fig. 3 Uncertainty distribution function options.

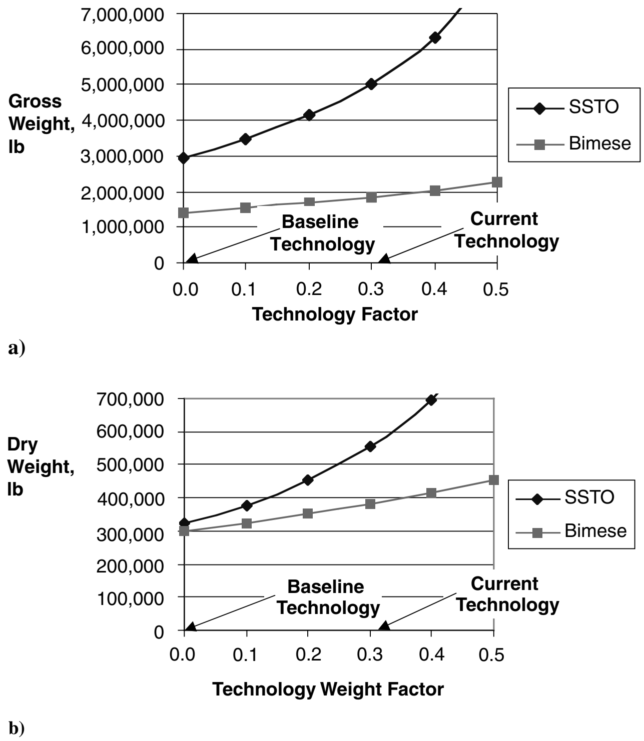

Fig. 4 Effect of technology factor on a) gross weight, b) dry weight.

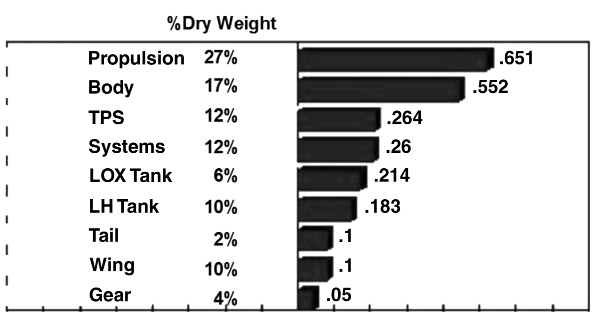

Fig. 5 Variable sensitivity on dry weight.

prioritization chart could be used as a guide for investment spending in a technology development program.

The accumulated probability results are shown in Fig. 6 showing the range of dry weights resulting from the total uncertainty band for each weight component. The mean is 340 klb, and at 95% probability, the weight is 426 klb resulting in a 25% difference.

In previous deterministic vehicle studies, the results are based on the mean with a vehicle growth factor on the empty weight. Many of these studies have assumed 15% growth margins. As shown in the results (and NASA's recommended margins in Table 3), additional margins should be considered.

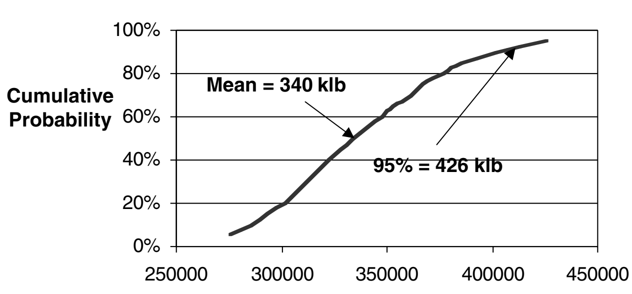

Fig. 6 Cumulative probability on dry weight.

However, in the case of a launch vehicle, weight growth cuts directly into payload performance. As an example for the SSTO concept, if the vehicle is designed for the 340 klb mean and there is a 25 klb (7.3%) overrun, which is equivalent to the payload, the vehicle delivers no payload to orbit. Also note the advanced technologies employed leave no contingencies to reduce the weight like the space shuttle orbiter. When the X-33 had a 35% weight overrun and the hydrogen tank was to be replaced by a heavier aluminum tank because of a test failure, the program was cancelled because the technology demonstration mission could not be performed. Also, commercial transports that are based on small increments in design and technology evolutions, have weight growth factors from 5 to 15%.

With the space shuttle development cost approximately $44 billion in present year dollars and the A380 subsonic transport projected development costs of $12 billion, the investment for a new reusable launch vehicle will be billions of dollars. Knowing the extreme risk of either significant reduction in payload or significant increase in cost to mitigate the weight growth in program development, the prudent probability selection for design weight probability should be greater than 50%. Thus, selecting 70, 80, or perhaps even 95% design probability does not seem unreasonable.

At a 95% probability for the present study, the result is a weight growth of 25% from the mean or 3.5 payloads as shown in Table 4. However, the reduced sensitivity nature of the Bimese results in a growth of only 9% or approximately 1 payload.

Adding the engine specific impulse, drag, and volume, the SSTO concept climbs to a growth of 31% or 4.3 payloads at 95% probability (Table 5) and the Bimese at 15% or 1.8 payloads (Table 6).

Table 4 Weight growth due to weight uncertainties (payload = 25,000 lb)

		Δ dry weight	
Cum. probability	Dry weight, lb	Mean	Payload
	SSTO		
Mean	339,884	0.00	0.0
60%	348,204	0.02	0.3
70%	361,014	0.06	0.8
80%	376,621	0.11	1.5
90%	401,815	0.18	2.5
95%	426,189	0.25	3.5
	Bimese		
Mean	311,331	0.00	0.0
60%	313,693	0.01	0.0
70%	318,849	0.02	0.2
80%	324,216	0.04	0.4
90%	332,721	0.07	0.8
95%	339,689	0.09	1.0

Table 5 Effect of adding I_{sp}, drag, and volume uncertainties (payload = 25,000 lb)

		Δ dry weight	
Cum. probability	Dry weight, lb	Mean	Payload
	SSTO		
Mean	339,884	0.00	0.0
60%	348,204	0.02	0.3
70%	361,014	0.06	0.8
80%	376,621	0.11	1.5
90%	401,815	0.18	2.5
95%	426,189	0.25	3.5
SSTO: $I_{sp} = \pm 5\%$, $drag = \pm 15\%$, $volume = \pm 5\%$			
Mean	349,035	0.00	0.0
60%	355,326	0.02	0.3
70%	371,730	0.07	0.9
80%	392,926	0.13	1.8
90%	430,897	0.23	3.3
95%	456,781	0.31	4.3

Table 6 Effect of adding I_{sp}, drag, and volume uncertainties for SSTO and Bimese (payload = 25,000 lb)

Cum. probability	Dry weight, lb	Δ dry weight Mean	Δ dry weight Payload
SSTO: $I_{sp} = \pm 5\%$, $drag = \pm 15\%$, $volume = \pm 5\%$			
Mean	349,035	0.00	0.0
60%	355,326	0.02	0.3
70%	371,730	0.07	0.9
80%	392,926	0.13	1.8
90%	430,897	0.23	3.3
95%	456,781	0.31	4.3
Bimese: $I_{sp} = \pm 5\%$, $drag = \pm 15\%$, $volume = \pm 5\%$			
Mean	311,331	0.00	0.0
60%	315,583	0.01	0.2
70%	324,489	0.04	0.5
80%	335,920	0.08	1.0
90%	348,252	0.12	1.5
95%	356,791	0.15	1.8

The analysis includes most of the known unknowns, but history shows there may be a number of unknown unknowns that should be modeled. Thus, the SSTO concept is much more sensitive to weight growth as compared with the Bimese, and thus should have more weight margin and management control than the Bimese. In spite of the fact that the Bimese reduces weight growth risk, the total system cost must be compared with the much larger SSTO concept.

IV. Summary

A process was developed for estimating the uncertainty of weight prediction for future space transportation systems. The impact of shortfalls in technology projections for the future launch vehicles was modeled as simple dry weight percentage increases to determine dry weight and gross weight sensitivities. In addition, weight modeling uncertainties were developed for single-stage and two-stage reusable launch concepts from historical weight-estimating relationships which are typically used during the concept definition phase of a system. Other uncertainties were assumed for specific impulse, drag, and subsystem volume to determine their relative sensitivities. The uncertainties were integrated into the systems analysis model, and a Monte Carlo simulation was conducted.

With the assumed technology projections, the two-stage and single-stage vehicles were very close in dry weight; however, as the technology weight factor increases (simulating shortfalls in technology projections), the two-stage system is much less sensitive to the weight growth as compared to the single-stage system.

With the uncertainty model, results showed there is a 31% weight growth (4.3 payloads) uncertainty in dry weight between a 50 and 95% probability for the single-stage-to-orbit concept. The Bimese concept (two identical single-stage concepts mated and sized for two-stage performance) resulted in a reduction in both weight (12%) and uncertainty (15% or 1.8 payloads) as compared to the SSTO concept. The top four drivers of the weight uncertainty are propulsion, body structure, thermal protection system, and subsystems.

References

[1] Stanley, D. O., Engelund, W. C., Lepsch, R. A., McMillin, M., Wurster, K. E., and Powell, Richard W., Rocket-Powered Single-Stage Vehicle Configuration Selection and Design., *AIAA/AHS/ASEE Aerospace Design Conference* AIAA Paper 932-1053, 1993.

[2] Haefeli, R. C., Littler, E. G., Hurley, J. B., and Winter, M. G., "Technology Requirements for Advanced Earth-Orbital Transportation Systems," NASA CR-2866, 1977, p. 14.

[3] Wiley, J. L., and Pranke, L. K., *Human Spaceflight-Mission Analysis and Design*, McGraw–Hill, New York, p. 787.

[4] Rohrschneider, R. R., "Development of a Mass Estimating Relationship Database for Launch Vehicle Conceptual Design," AE8900 Project, Georgia Inst. of Technology, Atlanta, GA, 26 April 2002.

[5] Ryschkewitsch, M. G., NASA Mission Design Process: An Engineering Guide to the Conceptual Design, Mission Analysis, and Definition Phases, NASA Engineering Management Council, 22 Dec. 1992.

J. Martin
Associate Editor