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Principles from Two Small SRM-Based Launcher Design/Developments

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Abstract

Two solid rocket motor based launch vehicles are in design and development at MSFC, a 4-stage Earth-to-Orbit vehicle cube-satellite launcher, and the 2-stage Mars Ascent Vehicle for the Mars sample return mission. Trade studies that refined these designs have benefited from developments in understanding sensitivities and approximate modeling. The development and practice of that has demonstrated 3 principles:

PRINCIPLE I: Identify mission-specific key design drivers — particularly interactions (and focus knowledge-development on those).

PRINCIPLE II: Agility — don't lock down your derived requirements early — keep trading, and characterizing sensitivities.

PRINCIPLE III: Use multi-fidelity tools to improve "understanding value generated per modeling effort cost."

The designer should expect different design solution techniques to derive from different mission needs. This paper shows how two launch vehicle designs developed and adapted to those needs.

Keywords: Launch vehicles, solid rocket motors, trade space

Nomenclature

Isp	specific impulse
ΔV	change in velocity
A_t	throat area
C_f	thrust coefficient
F	thrust
k	number of stages in the vehicle
m	mass
m_p	propellant mass
$\dot{m}$	mass flow rate
n	burn rate exponent
P	pressure
P_{ref}	reference pressure
$\dot{r}$	burn rate
$\dot{r}_{ref}$	reference burn rate
t	time

Acronyms/Abbreviations

CEL	Cubesat Earth Launcher
ERO	Earth Return Orbiter
GLOM	gross liftoff mass
MAV	Mars Ascent Vehicle
MSR	Mars Sample Retrieval
RCS	reaction control system
SRL	Sample Retrieval Lander
SRM	solid rocket motor

1. Introduction

A perennial challenge in space system and launch vehicle development is balancing going into detail enough to answer the salient development questions in a timely manner, but not becoming paralyzed by the immensity of getting all the answers. Sometimes this leads to a project languishing in the early stages because funding appears needed to get beyond a textbook level of detail. Other times early projections are wildly optimistic, only for hopes to be dashed later. Early projections are perhaps wildly pessimistic at times too, but in those cases probably nothing more is heard from the idea. Other times the purveyor of a new system may overly refine the area he is most familiar with, while leaving other areas woefully undeveloped. Often in any of these cases, a single point design is early selected as representative of the whole endeavour.

The author has previously posited the benefits of a trade space mentality, vis-à-vis a point design mentality [2]. The over-arching principle is to gain a lot of understanding for a little modeling effort, in order to enable decisions along the way. The value of the next chunk of effort will be based on how urgent and important are the decisions it enables. Therefore, the practitioner of this is constantly refining, as with the detailed levels of the systems engineering process iterations, but uses tools to (a) assess more cases with fewer resources, and (b) estimate which realms should more urgently be refined and which can wait.

2. Launch vehicle architectures

The vehicles to be considered are both composed of all small solid rocket motor stages. The Mars Ascent Vehicle is a 2-stage launch vehicle to be used for Mars Sample Return. The Cubesat Earth Launcher (CEL) is a 4-stage launch vehicle for earth-to-orbit launch of cubesats. The payloads for both these are less than 20 kg, so this derives small stages for each as well: the smallest two stages for each (Table 1) are on the order of 50 and 200 kg of propellant respectively, with the larger stages of the CEL 500-700 kg. This small size means stages of any type would be subject to the mass efficiency penalties of scale, as described in previous works by the author [1,2]. They are small enough that minimum volume and even minimum mass solutions (especially for MAV) are available with solid rocket motors.

The Mars Ascent Vehicle (MAV), as part of a potential Mars Sample Retrieval (MSR) campaign, is intended to transport samples from the Mars surface to an orbiting Earth return vehicle. MSR is envisioned as utilizing a series of three Earth launches. The first launch consists of a rover that collects Mars rock samples and deposits them for future recovery (Mars 2020). The second and third launches, potentially as early as 2026, would deliver the MAV as part of a Sample Retrieval Lander (SRL) and the Earth Return Orbiter (ERO). The MAV would descend to the Mars surface onboard the lander. The SRL would retrieve the samples and insert them into the MAV before returning to the ERO and Earth.

Due to Mars' lower gravity, the ideal ΔV required for the MAV mission is only 3990 m/s, including losses, for the reference configuration. This ΔV is low enough to require just 2 stages. For the earth-launched CEL, the ΔV required is estimated at 9300 m/s. Given first stage constraints, a 4-stage architecture was derived. The recent reference cases for these two vehicles are summarized in Table 1.

Table 1. Summary of Reference Cases for Two Launch Vehicles, MAV and CEL.

			MAV	CEL
	Payload Target	kg	16	10
	Injected Mass	kg	**64**	**24**
	Total/*n*th stage ΔV	m/s	3990 / **1690**	9300 / **3110**
Stage *k*	Non-prop inerts	kg	**34**	**2**
	inert	kg	14	12
	propellant	kg	54	51
	Comments		Accuracy-driven	Inert Mass-driven, burntime-limited
Stage *k-1*	Non-prop inerts	kg	14	**28**
	inert	kg	46	25
	propellant	kg	216	226
	Comments		Boost-sustain reqd	Burntime-limited, control systems here
Stage *k*-2	Non-prop inerts	kg		10
	inert	kg		77
	propellant	kg		540
	Stage *k*-3 propellant	kg		650
	Comments			Set by partner

3. PRINCIPLE I: Identify mission-specific key design drivers — particularly interactions (and focus knowledge-development on those).

Sometimes this is as simple as a review of the top-level requirements: What is this mission to accomplish, and what are its key quality values? These don't need to be derived down into stage-specifics yet; rather trades in architecture space will show what impact they have. Table 2 shows some of the main drivers for each mission. Notice their particular distinctives from each other.

The initial architectural decisions are additionally informed by ideal-rocket multi-stage sizing activity and empirical mass modelling [1]. Included in these looks are sensitivities to ascertain the effect of development uncertainties and see if early decisions can make the system more optimal, or less likely to be perturbed in the long run. A key to this is the effect of increasing inert mass in the uppermost stages, along with an estimate of what the non-propulsion inert masses might be (for example, control systems).

Table 2. Mission Values and Decisions.

Mars Ascent Vehicle (MAV):	***Cubesat Earth Launcher (CEL):***
limit GLOM, overall length, diameter	limit individual stage length, diameter
have high reliability	limit max acceleration
limit orbital uncertainty	have low cost
launch to a single orbit	attempt medium reliability
	allow launch to multiple orbits
"derived architectural decisions"	
Use pedigreed, refined motor manufacturer/methods	Use "low-cost" small manufacturer
use thrust vector controlled motors plus RCS system	invoke spin-stabilization with steering between burns design to "bend the cost-performance curve" (can't ignore performance, because mass also costs) Optimize for mass by putting control system on 3rd stage instead of upper stage
Drives saving 2nd stage mass at the expense of 1st stage mass	max accel proves a driving limitation

The MAV upper stage is unusually heavy on masses that are not the motor inert mass and not the payload [2]. These masses include the reaction control system (RCS), the avionics package, the structural masses housing them, the vehicle skin around the spherical motor, and thermal control components. Therefore, while accurately estimating solid rocket motor (SRM) inert masses was important as always, assessing whether propulsion system changes would have any effect on the non-propulsion inerts was even more essential.

Identifying what to consider took place in two ways: aerodynamic control/performance and mechanical. Figure 1 illustrates the vehicle performance caused by the first stage and RCS affecting each other. As Ref. 1 described, a long enough burn time was required in order to limit the dynamic pressure at burnout so that the load on the RCS would not be so great. Because of the vehicle diameter, burn times of this duration are not typical. To explore how to provide this, the following steps were undertaken:

1. Survey of existing motors – found STAR15G to stretch and scale, use to inform mass fractions. This identified that the boost-sustain motor will have a lower mass fraction than a standard center-perforated motor, and vehicle sizing estimates were updated. Previous efforts had defined full end-burning motors, which had an even lower mass fraction.
2. Understand effect of motor gross parameters on burn times and thrust levels.
3. Look at grain design features that could generate a boost-sustain and further customize.

Within the variables identified, there were several degrees of freedom for optimization, within the motor propellant grain and operating pressure, and regarding motor length, diameter and nozzle dimensions, subject to trajectory-defined thrust constraints and overall vehicle dimension constraints. The optimization of this motor design over this space is described in Principle 3.

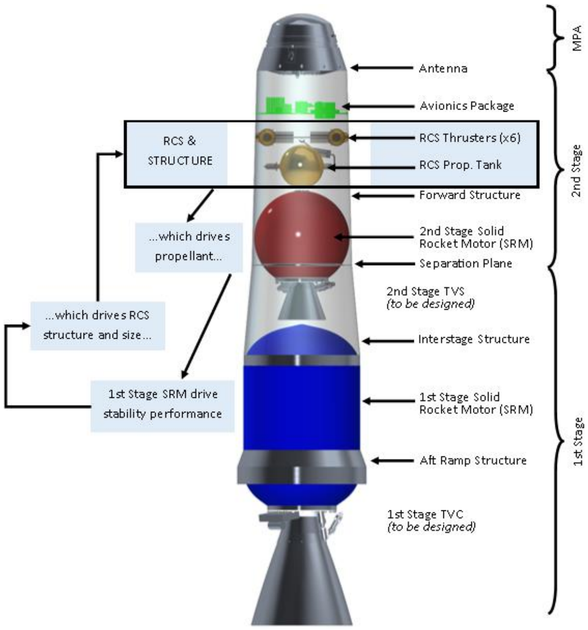

Figure 1. MAV solid propulsion vehicle.

On the CEL, a key driver proved to be an acceleration limit, especially on the 4th stage. Scaling equations were derived from bulk motor internal ballistics relationships to understand the effect of this. These equations also enable thrust traces to be scaled at constant max acceleration, whether they have full grain design modelling with them or not. Controlling the acceleration on these motors proved a particular challenge because the reference burn rate ($\dot{r}_{ref}$ in Eq 1) was already at the minimum producible at the vendor.

Within the CEL design space, the propellant mass is a key free variable at the system level, so even if a motor has been designed to high fidelity, it is important to have tools that can answer questions like "what if we add 20% more propellant to that motor?" The following equations estimate this at constant reference burn rate, attempting to keep max acceleration constant, and assuming St. Robert's law holds (Eq 1). Solid rocket performance equations are defined parametrically, so that time gets adjusted along with the other performance parameters

$$\dot{r} = \dot{r}_{ref}\left(\frac{P}{P_{ref}}\right)^{n} \quad (1)$$

To preserve constant max acceleration, assume that thrust must be unchanged, just spread through more time in the case of increasing the propellant mass. Errors in this assumption are merely the addition of motor inert mass and/or payload that might go with the larger motor.

$$\frac{F_2}{F_1} = \frac{P_2 A_{t_2} C_{f_2}}{P_1 A_{t_1} C_{f_1}} = 1 \quad (2)$$

$$\frac{P_2}{P_1} = \left(\frac{m_{p2}}{m_{p1}}\right)^{-\frac{1}{n}} \quad (3)$$

$$\frac{t_2}{t_1} = \frac{m_{p_2}}{m_{p_1}} \quad (4)$$

Since mass flow rate, like thrust, is proportional to pressure and throat area,

$$\frac{\dot{m}_2}{\dot{m}_1} = \frac{A_{t_2}}{A_{t_1}}\left(\frac{m_{p_2}}{m_{p_1}}\frac{A_{t_1}}{A_{t_2}}\right)^{\frac{1}{1-n}} = 1 \quad (5)$$

Therefore, to this order of fidelity, finding the same acceleration trace for a different propellant mass means setting the new nozzle throat area according to Eq (6):

$$\frac{A_{t_2}}{A_{t_1}} = \left(\frac{m_{p2}}{m_{p1}}\right)^{\frac{1}{n}} \quad (6)$$

Errors in the ballistics assumptions for this are if the thrust coefficient decreases because the nozzle exit does not increase proportionally with the throat, or if the throat erosion is significant and not consistent at the two throats.

Compare this to a more typical "constant pressure" assumption, Eq (7) – (11):

$$\frac{P_2}{P_1} = \left(\frac{m_{p2}}{m_{p1}}\frac{A_{t1}}{A_{t2}}\right)^{\frac{1}{1-n}} = 1 \quad (7)$$

$$\frac{F_2}{F_1} = \frac{m_{p_2} C_{f_2}}{m_{p_1} C_{f_1}} \quad (8)$$

$$\frac{\dot{m}_2}{\dot{m}_1} = \frac{m_{p2}}{m_{p1}} \quad (9)$$

$$\frac{t_2}{t_1} = 1 \quad (10)$$

$$\frac{A_{t_2}}{A_{t_1}} = \frac{m_{p2}}{m_{p1}} \quad (11)$$

Scaling at constant pressure would maintain the motor case thickness and probably the thicknesses of case and nozzle insulators.

With constant pressure, the throat only has to increase proportionally with propellant mass. In the constant acceleration case, assuming a reasonable burning rate exponent of 0.33, the throat area must increase with propellant mass cubed! This will have system repercussions by either driving up the nozzle mass, length, and diameter in order to maintain Isp, or by driving down the Isp at the same nozzle envelope.

4. <u>PRINCIPLE II</u>: Agility — don't lock down your derived requirements early — keep trading, and characterizing sensitivities.

Our Jet Propulsion Lab leaders were uncompromising in refusing to flow down requirements at this early "preliminary architecture assessment" maturity. Yes, there was a definition of orbit and mass goal, but we were encouraged to question both of those too as required to judge the feasibility and sensitivities of each vehicle concept.

What's needed then, is "early and often" identification of interactions between (traditionally separated) disciplines and components. That way, we define an actual multi-variate trade space, and then identify and refine only the most sensitive interactions initially. Another benefit is that once something changes you don't "have to do a complete redesign," you just turn the dial and focus on that new portion of the design space.

This forces analysts to be designers and designers to be (0th-order, anyway) analysts. A key example of this is explained more in Principle 3, where reduced-order insulation modelling was combined with motor grain design and optimization tools to save at least 10 lb and 3 inches. In another case, a design of experiments of trajectory runs was conducted for a trade space of 3 motor propellant masses, 2 case masses, and 2 nozzle lengths, affecting both mass and Isp. The outputs of payload margin and stage 2 propellant offload, when subjected to linear regression, delivered both useful partials – e.g., change in stage 1 propellant per change in upper stage inert mass – and an overall predictive model over the ranges tested. This made adjusting for future payload increase and other mass uncertainties trivial.

Having this modelling paid off weeks later when exploring MAV orbital insertion accuracy, which became elevated as the #1 value. Design solutions to control the accuracy required either more non-payload inert mass on the 2nd stage, or increased ΔV. The design space mapping that had been put in place allowed quick projection of the resulting propellant and GLOM increases in order to compare options.

Where there are these interactions, one pass through the integrated design assessment is unlikely to yield the perfect answer. Instead of taking the time of a lengthy Design-Analysis-Cycle to "make it perfect," like during later portions of a development, let all the involved disciplines start at once, with whatever imperfect information they have, but do a quicker pass so they can find sensitivities and surprises. Good, frequent communication with each other is imperative: they can more easily update their assumptions as they see how they have bearing on each other.

5. <u>PRINCIPLE III</u>: Use multi-fidelity tools to improve "understanding value generated per modeling effort cost."

5.1 Mars Ascent Vehicle Motor Design Fidelity Ladder

The fidelity ladder shows the importance of deciding what your independent variables are, tells you when you need to add a fidelity step. This shows the application of the mission values from Table 2, but each next step on the ladder is usually only discovered from the step before.

A couple of these steps admit a bit more discussion: Step 4 proved especially beneficial for the 2nd stage as shown in Figure 2. At one point the project reduced the allowed MAV length, and the current reference design exceeded the new limit. So nozzles were looked at as a place to potentially save length. For the 2nd stage, a high expansion ratio nozzle had been designed, so a nozzle mass per unit length estimate was available. The interstage mass per unit length estimate ended up being even larger, suggesting a new optimization was probable.

Table 3: Fidelity Ladder as Used for MAV

	MAV steps of increased fidelity	*MAV application*
6	- then, running propellant grain designs, what is the best at each setting? - first, understand what are the gross effects **consider effect of pressure on case, insulation and nozzle masses, and Isp**	Compute consistent set of grain designs at different diameters, pressures, and propellant masses
5	Develop more particular design constraints	quantify "boost-sustain-ness"
4	- consider mass on multiple stages and solve for max payload (or minimum GLOM) at constant ΔVs - know likely payload mass and maximize ΔV OR set ΔV and maximize payload mass - estimate Isp of ER calibrated to reference - estimate "local" inert mass change per unit propellant mass change - estimate structural thicknesses/materials **Model length, diameter range from a baseline**	shorten 2nd stage nozzle to maximize ΔV
3	all-solids modified by features similar to certain motors	calibrate to catalog boost-sustain motor
2	all-solids mass modeling	too small for 0.9
1	textbook or searched mass fraction and Isp to use	use 0.9 prop mass fraction

Figure 2 shows a simplified set of model results departing from the baseline of 0 length change. Even though Isp reduces significantly by the peak, the mass savings more than makes up for it. Where the models diverge suggests the next refinement steps to take.

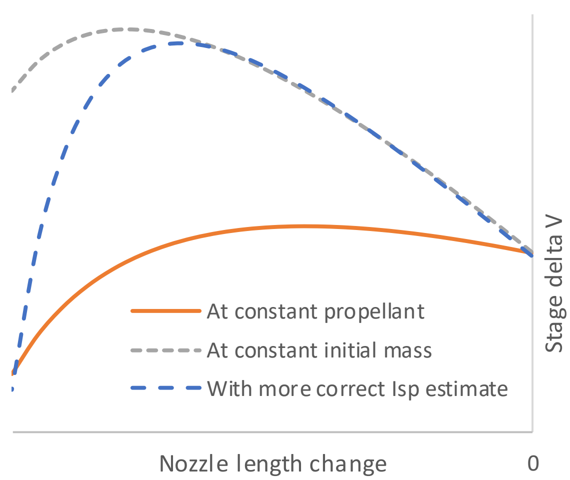

Figure 2. Nozzle length trade study.

Steps 5-6 led to the ability to explore and optimize over the design space of Stage 1 motor grain designs. After all the independent variables and constraints were set up, a optimums were examined for points within some of the independent variables, rather than just jumping to a global optimum of 3 pressures, 3 diameters, and 4 numbers of propellant fins.

Always over a range of propellant masses – as mentioned above, propellant mass must be kept as a design space as long as possible to accommodate other vehicle changes. Design cases of these are shown in Figure 3. Outputs that matter are:

- Total length
- Nozzle mass
- Insulation mass
- Stage total ΔV

Rather than speculate about how to weight the length and masses, a stage total ΔV was derived from the mass models, and Isp model and the ideal rocket equation. So ideal ΔV of the stage was maximized at each propellant mass to determine a suite of "best" designs. That produces a snapshot at a particular stage payload mass of how the stage inert mass and Isp offset each other. Even though full vehicle optimizations to follow might adjust the stage payload (i.e., the upper stage total mass), it is expected that small changes would not drive the optimum far from the computed point. Even then, the final nozzle length is not locked down, so the nozzle length trade can be redone as a last adjustment.

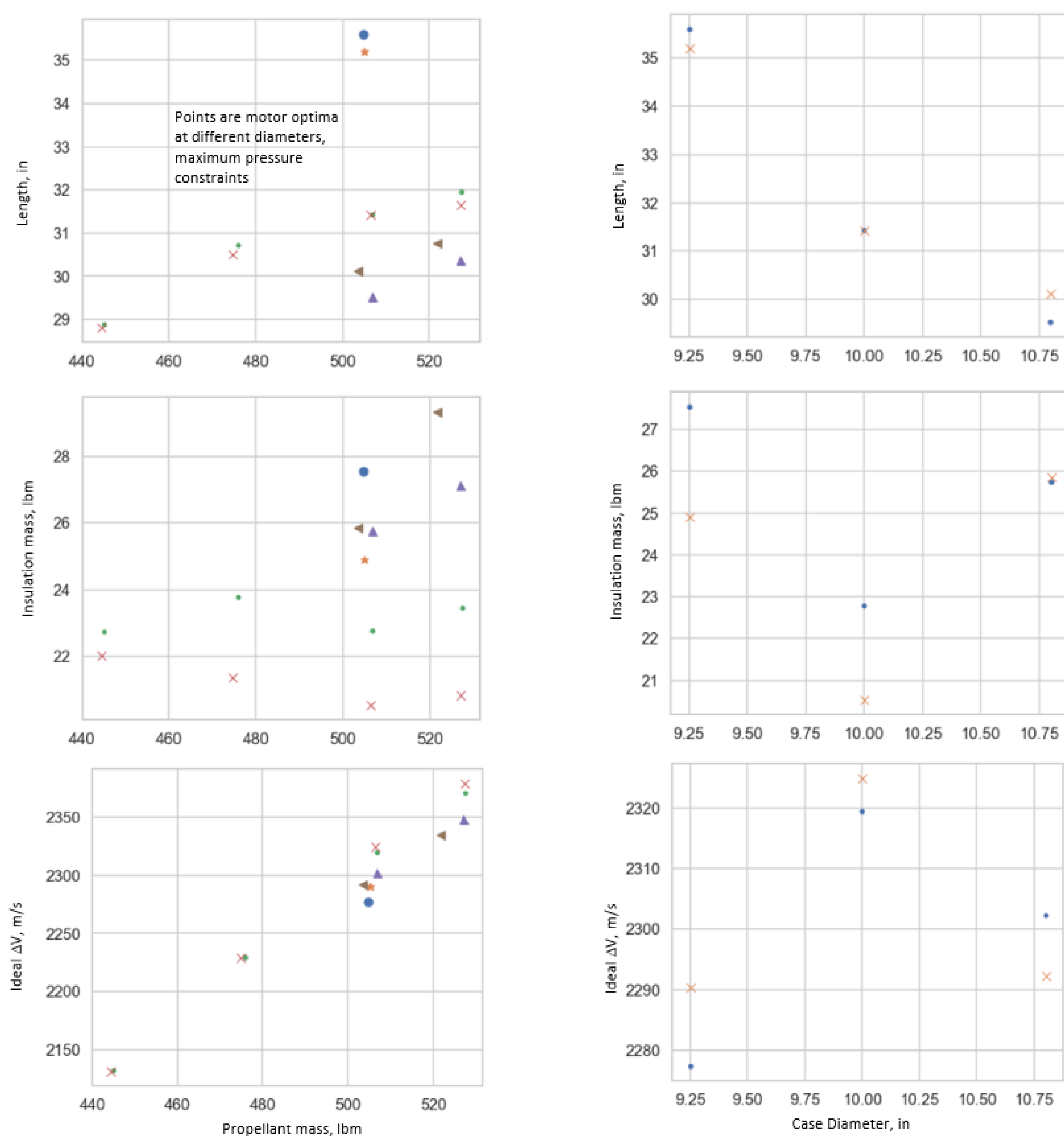

Figure 3. Simplified set of model results departing from the baseline.

Note that fully detailed trajectory optimizations were not in the loop of this part of the optimization. In other programs, often the optimizing engineer adds and adds models to his optimizer until he presents “the one true optimal point!” But that leaves out all information about sensitivities, and if any assumption underlying the first analysis needs tweaked, his answer tends to be “well, I’ll have to run it again.” Instead, in this case the designer has chosen where to peer in to the optimization stack – first, to see how diameters, pressures and number of fins affect things, and then, to see how the single stage can deliver an optimum performance, all the while maintaining results “versus propellant mass.” This enabled the engineer to see that the motor optimization predicted a savings of about 4 inches and 10 lbs of insulation by using its design settings. When a nozzle-propellant interference was found and corrected, it merely corrected all the projections by 1 inch length, rather than invalidating a carefully honed optimal point.

Table 4: Fidelity Ladder as Used for CEL

	CEL steps of increased fidelity	***CEL application***
6	Develop more particular design constraints B	Adjust insulation and nozzle contours to minimize mass given constraints
5	- balance diameter-driven masses with other constraints - generate full grain designs to test ability to customize for acceleration **Develop more particular design constraints A**	Get the most out of the upper stage, which was found to be most driven by constraints
4	- model across multiple stages: solve for max payload assuming constant total ΔV - estimate Isp of ER calibrated to reference - estimate lengthening/ shortening of all stages **Model nozzle/stage mass, Isp to trade**	updates marginal inert mass per unit propellant mass, then enables multi-stage trades
3	all-solids modified by features similar to certain motors	heavier because of acceleration constraint
2	all-solids mass modeling	also be at the lower end for “affordable”
1	Textbook/searched mass fraction and Isp to use	

5.2 Cubesat Earth Launcher Motor Design Fidelity Ladder

Notice by comparing Table 3 to Table 4 that the fidelity ladder for the CEL starts with the same general steps for fidelity increase, but for different reasons, with different customizations. Then at the 4th level the steps themselves diverge because of the different mission needs.

It may appear trivial at first glance to separate out the iterative design workflow to this many gradations, remember these 2 things:

1. Stepping up from 1 to 6 in each program took place over months – meanwhile decisions were enabled and made when a new level was attained.
2. This is only a small set of all the fidelity that could be added. It was the examination of each step in light of program specifics that deferred many possible next steps, thus saving time.

Thus the fidelity ladder stepping model builds maximum understanding for decision making out of minimum effort required to generate the knowledge.

6. Conclusions

Utilize models at the simplest level that tells you what you need, while checking your assumptions. But also don’t throw out/ignore existing knowledge (ie, things that existing high-fidelity models can tell you). SOLUTION: iterate – compute high-fidelity baselines, a few, regularly – then trade study with many low-fidelity runs in between.

From the lowest fidelity, there are many “next level of fidelity” modeling options. Don’t be afraid to test these, but only start implementing the ones that are discriminators for your particular problem. This builds a rapidly-informative ladder of fidelity.

Don’t hold out for “the one great global optimization.” And even if you have it, break it into mini-optimizations, where you peer into the results at certain levels. Wisdom is deciding which levels to peer in.

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References

[1] A. Prince, R. McCauley, T. Kibbey, L. McCollum, B. Oglesby, P. Stefanski, Mars Ascent Vehicle Propulsion System Solid Motor Technology Plans, IEEE Aerospace Conference, Big Sky, MT, 2019, 2 March.

[2] T. Kibbey, Small Launch Vehicle Sizing Analysis with Solid Rocket Examples, 66th JANNAF Propulsion Meeting, Dayton, OH, 2019, 03 – 07 June.

[3] A. Prince, T. Kibbey, A. Karp, A Design for a Two-Stage Solid Mars Ascent Vehicle, AIAA Propulsion and Energy Forum, Indianapolis, IN, 2019, 19 – 22 August.