A Multi-Disciplinary Analysis Framework for the Design of Small Launch Vehicles

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The decisions made in the conceptual design phase have large impacts on the resulting vehicle capability and cost. With the large and varied design space of Small Launch Vehicles (SLVs), it is even more important to have the data necessary to make informed design choices during the conceptual design phase. To enable extensive exploration of the SLV design space, a multi-disciplinary design framework was created. The design framework consists of trajectory, aerodynamics, propulsion, and structures disciplines. The framework then integrates and automates these tools, thus allowing for rapid design space exploration at the conceptual design phase. The outputs of the framework provide preliminary information on SLV size and structural configurations as well as propulsion and aerodynamic characteristics. Finally, the framework provides information necessary for the designer to make informed decisions on what variables lead to the desired performance characteristics and what segment of the design space would benefit from more detailed exploration.

Nomenclature

D	=	Launch Vehicle Diameter (ft)
L_n	=	nose cone length (ft)
$MEOP$	=	maximum expected operating pressure
d_{i_c}	=	inner diameter of the casing
d_p	=	propellant grain diameter
d_{stage}	=	the outer diameter of the casing of a stage
$gamirv_1$	=	Flight path angle of the inertial velocity of the desired orbit [°]
$gamirv_2$	=	Flight path angle of the inertial velocity of the achieved orbit [°]
$gcrad_1$	=	Geocentric radius of the desired orbit [ft]
$gcrad_2$	=	Geocentric radius of the achieved orbit [ft]
i_1	=	Inclination of the desired orbit [°]
i_2	=	Inclination of the achieved orbit [°]
n_f	=	Fineness ratio
t_c	=	motor casing thickness
t_i	=	insulation thickness
v_1	=	Orbital velocity of the desired orbit [ft/s]
v_2	=	Orbital velocity of the achieved orbit [ft/s]

I. Introduction

Between the years of 1995 and 2014 the number of small satellites launched per year (1-500 kg) went up from 20 to 180; out of the 180 launched in 2014, 66% were Nano satellites (1-10 kg)[1]. Most of these small satellites are launched as secondary payloads on medium launch vehicles (LVs) [1]. This secondary payload placement is mainly due to cost because it is prohibitively expensive to launch a 10 kg satellite on LVs capable of carrying over 1000 kg to orbit. Small satellites either have to ride share with bigger satellites or wait for a large enough group of small sats going to a similar orbit. This results in small satellites being placed into either suboptimal orbits or waiting for launch schedules to align with larger launches, resulting in long waiting times. All of these factors lead to a large backlog of small sats waiting to be launched [2]. A launch vehicle dedicated to launching small satellites would improve the responsiveness of small satellite launches and increase availability to space for smaller entities. A small launch vehicle (SLV) is defined by its capability to carry 1-100 kg to orbit. Currently, only about 8% of all small satellites are launched on SLVs [1], but due

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to additional demand there has been an increase in proposed SLV designs [3] [4]. Many of these proposals have failed to provide an actualized design; for some, this is due to the parent companies going bankrupt or rescoping, while others suffered setbacks due to failed test launches stalling the program [4]. These shortfalls show that the development of a cost-effective SLV is a difficult and expensive process.

The development of an SLV starts in the conceptual design phase where a vehicle architecture is chosen; these decisions determine 80 % of vehicular life cycle costs [5]. Therefore, a thorough examination of SLV architecture options during conceptual design is essential to creating a cost-effective SLV. However, the reduced size of an SLV allows for a large architecture space of feasible concepts; this results in having to explore a number of discrete design options during the conceptual design phase. Some of these discrete options include solid vs. liquid fuels, number of stages, lifting body vs. non-lifting body, etc. [6] [7]. Another key option is launch modality: of thirty-four SLVs proposed by the year 2018, five are sea launched, and nine are air launched with the use of airplanes[4]. Air launches provide better fuel efficiency, but have been traditionally limited by the capacity of airplanes [8]. Other concepts have proposed utilizing weather balloons to facilitate air launches [4]. This large architecture space necessitates design methodologies that allow for its exploration, and any design methodology capable of exploring the SLV architecture space will require a sizing environment/framework that is capable of rapidly providing the necessary information for conceptual-level trade studies.

This paper will explore the creation of a framework capable of this exploration. Section two will discuss the SLV multidisciplinary analysis framework and its constituent analysis tools. Section three will present the sample problem which revolves around the design of a four stage solid propellant SLV. Section four will show and discuss the results.

II. Framework

The goal of the framework is to enable rapid exploration of the SLV design space during conceptual design. The specific fidelity and disciplinary requirements of the framework are derived from this need. The basic task of a launch vehicle is to carry a payload to orbit; in order to determine the payload capability of an LV, the vehicle's trajectory to orbit must be calculated and optimized (trajectory discipline). The trajectory is directly affected by mass properties, aerodynamic coefficients, and propulsion characteristics of the vehicle (mass estimation, aerodynamics, and propulsion disciplines). In order to have an accurate estimation of vehicle dry mass, thicknesses of structural components such as interstages, skirts, and pressure vessels must be calculated (structures discipline). The wet mass of the SLV is defined by the interaction between propulsion and structures disciplines which determine the amount of usable propellant in each solid motor stage.

For the aerodynamics and trajectory disciplines, Missile DATCOM and POST2 are used, respectively. The propulsion and structures disciplines are represented in the framework with tools developed at the Aerospace System Design Laboratory (ASDL) at Georgia Tech. For propulsion, the Solid Motor Analysis Code (SMAC) is a physics-based conceptual design tool for solid rocket motors; it is capable of geometric burn simulation, ballistic analysis, and prediction of thrust performance [9]. For structures, the Launch Vehicle Structural Analysis (LVSA) tool is a physics-based tool that focuses on structural dynamic analysis with sizing capability. These tools are integrated into the framework, as seen in Figure 1. Due to interaction between the tools, this becomes a multidisciplinary analysis (MDA) problem with the corresponding connections described in Table 1. Each of the disciplines and their interactions will be further described in the subsequent subsections.

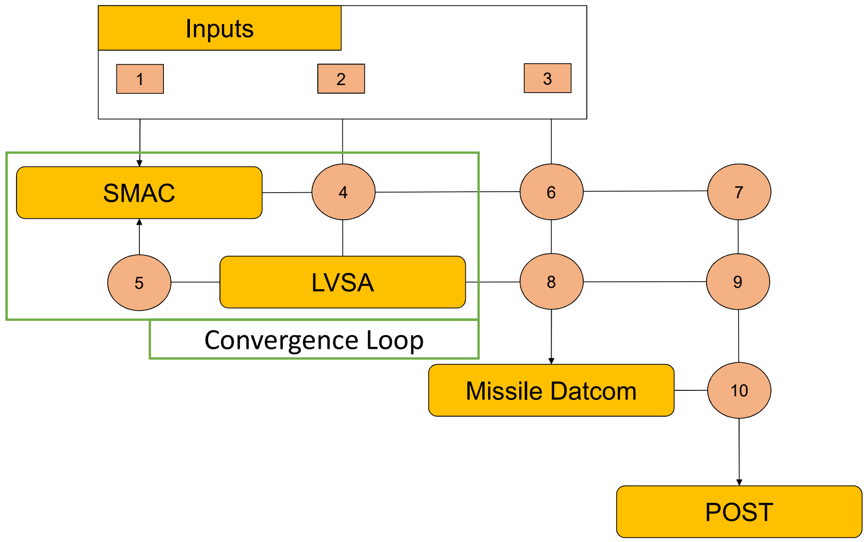


Fig. 1 Process flow diagram

Table 1 Variable Break down list

1	Width of Fin
	MEOP Target
	Propellant
	Grain Type
	Star Points
	Inner Circle Diameter
	Outer Circle Diameter
	Nozzle Length
2	Stage Length
	Vehicle Diameter
	Nose Shape Parameter

3	Vehicle Diameter
	Nose Shape Parameter
4	MEOP
	Insulation Thickness
5	Grain Diameter
6	Nozzle Diameter
7	Engine Deck
	Fuel Weight
8	Complete Stage Length
9	Structural Weight
10	Aerodynamic Decks

A. Propulsion

The propulsion discipline is responsible for sizing the solid propellant, insulation thicknesses, and, in coordination with the structures discipline, the motor casings and nozzles as well as evaluating the propulsion performance. A computationally inexpensive analysis tool developed by ASDL as part of the AIAA missile team competition and master's research, called Solid Motor Analysis Code (SMAC), was selected for the propulsion portion of the SLV framework [10][11]. SMAC was chosen due to its ability to model solid rocket motor performance, and its direct applicability to the type of vehicle being designed: all solid propellant vehicle. SMAC utilizes a forest fire geometric algorithm for burn prediction of propellant grains and is adaptable to various realistic core geometries; a visualization of the geometric burn algorithm is seen in Figure 2. SMAC employs the lumped parameter analysis of unsteady heat transfer to compute the performance of solid rocket motors.

SMAC has input parameters separated into three categories: initial grain geometry, propellant characteristics, and design characteristics. Regardless of composition, solid propellants have an initial geometric configuration called propellant grain. The propellant grain type is an input parameter to be specified in the initial grain geometry (finocyl, star, bates, etc.). Knowing the propellant grain type will determine the progression of the burn and the performance of the solid rocket motor. Other parameters in the initial grain geometry definition include grain diameter, inner/outer diameter of internal geometry, and fineness ratio of the grain. In addition to defining the geometric description of motor, SMAC utilizes propellant characteristics such as propellant density, burn rate coefficient, characteristic velocity,

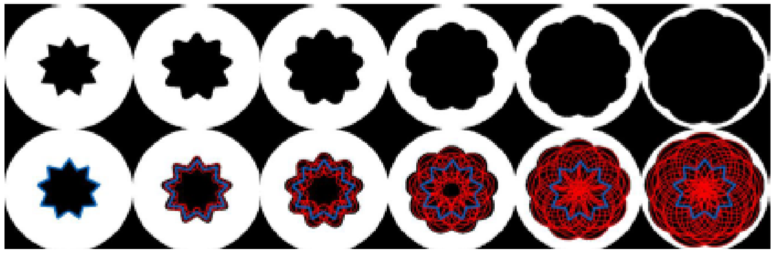


Fig. 2 Illustration of algorithm for Geometric burn [11]

pressure exponent, and the heat capacity ratio as part of the propellant burn calculation. Lastly, the design characteristics are inputs to SMAC in order to determine the operational performance. These are parameters that define the geometry of the nozzle, maximum expected operating pressure (MEOP) of the chamber, ambient temperature, and ambient design pressure.

SMAC is portioned into three modules: geometric analysis, ballistic analysis, and mass estimation. The geometric analysis portion of SMAC simulates a core geometry from the initial grain geometry input parameters. From the core geometry, a matrix of nodes is constructed to define the boundaries of the propellant grain cross section. This identifies the grain type by determining the boundary of where the propellant ends and the internally void region of the propellant grain begins. Once this boundary is defined, the geometric analysis proceeds to simulate a growing circular burn pattern at a specified step increase until the propellant is consumed. Burn area is calculated with respect to the burn progression distance and is used when analyzing the ballistic behavior of the solid rocket motor.

As stated earlier, the ballistic analysis portion of SMAC implements the lumped parameter method for unsteady heat transfer where there is assumed no flow velocity throughout the motor. The mass flow rate of the consumed propellant is determined using the propellant characteristics input parameters along with the current burn area computed in the geometric analysis. The target MEOP specified as an input is used as a determining factor to converge on the nozzle throat area. Then nozzle throat area is used to determine mass flow rate out of the motor. After the mass flow rate of produced mass and mass flow rate out of the motor are defined, the mass accumulation in the motor over the burn time is computed. Utilizing the unsteady lumped parameter method for these calculations is required for the next step within the ballistic analysis module of SMAC which is the ballistic element method. This method differs from unsteady lumped parameter method due to its examination of the internal flow velocity. The ballistic element analysis is higher fidelity because it partitions the chamber volume (continuous in lumped parameter method) into discrete elements. This permits changes in stagnation and static pressures along the motor length. This is an important distinction because the burn rate of propellant is now analyzed at each discrete element which provides insight into the effect burn rate variation has on the motor and burn progression due to erosive burning.

The last module of SMAC uses mass estimating relationships determined with historical regressions to calculate the mass of the igniter, insulation, the motor casing, and the nozzle. The mass of the propellant makes up most of the mass of a solid propellant launch vehicle and is calculated using propellant density and volume. The main outputs from SMAC include burn time, max thrust, specific impulse, and propellant mass. Other important outputs as a function of burn time include thrust, pressure, and mass flow.

SMAC has been validated using several validation cases. One such case is the well-known space shuttle's reusable solid rocket motor (RSRM). Table 2 shows the comparison of experimental values versus SMAC's predicted values for key motor parameters. Other validation cases that vary greatly from RSRM such as small SRMs were executed to prove the versatility of SMAC. The small SRM validation case confirmed to have similar accuracy of results as that of the RSRM.

Table 2 SMAC results comparison with experimental values for RSRM validation case [11]

Parameter	*SMAC Prediction*	*Experimental*
Max thrust (lbf)	3,280,000	3,370,000
Max pressure (psi)	913	900
Burn time (s)	125	125
I_{sp} *(s)*	268	268

B. Structures

The structures discipline is responsible for sizing the body of the vehicle to withstand the loads experienced during the mission. An analysis of loads is needed to size the vehicle structures and determine the overall dry mass. An ASDL developed framework called Launch Vehicle Structural Analysis (LVSA) was chosen to fulfill this need. LVSA acts as a framework for multifidelity multidisciplinary structural analysis and design of launch vehicles. To this point, structural dynamic analysis and analytical sizing have been integrated in LVSA. There are many tool options that could be utilized for the structures discipline in the SLV framework, however, LVSA was favored for its physics-based approach to structural sizing and its flexibility to adapt to most launch vehicle cases (missiles, cores, boosters, small/large launch vehicles, etc.).

LVSA's foundational functionality is its centralized vehicle description. The tool operates primarily on an axisymmetric assumption for component profiles. For these types of geometry, this approach removes the dependency on CAD models for vehicle definition which adds more freedom in conceptual structural design specifications. This also allows for analytical representations of geometric and mass properties of revolved surfaces and structural analysis of shells of revolution within the Python-based tool. The primary steps of vehicle description in LVSA are as follows: profile definition, component definition, assembly definition, and definition of time dependent features such as loads, boundary conditions, flight conditions, etc. Once the vehicle definition is instantiated, modules can be set up to refine the vehicle and gain further understanding of the system.

The geometric description module requires the vehicle to be described as a series of general barrels and domes. Here, barrels are defined as components with two open ends, and domes are defined as components with one open end and one closed end. Since this geometry is axisymmetric, only a two-dimensional profile is needed to define the shape to be swept around an axis. Several common payload fairing profiles are available, such as Cassinian, elliptical, Haack, parabolic, power series, ogive, and torispherical. A custom profile can also be defined using a series of normalized points. A depiction of this can be seen in Figure 3. Profiles are placed by global reference location and rotation angles about X, Y, and Z axes. Dome profiles can also be truncated by providing a different reference diameter. Dome profiles available in LVSA include Blunted Conic, Blunted Tangent Ogive, Parabolic, Conic, etc. and are used to describe storage tanks. Barrel profiles have two categories: straight and curved. The straight barrel profile will have a straight line connecting two open ends with either the same or different diameters. The curved barrel profile is similar to a dome truncated from the closed end. As mentioned, all profiles are assumed to be axisymmetric which creates a better opportunity for

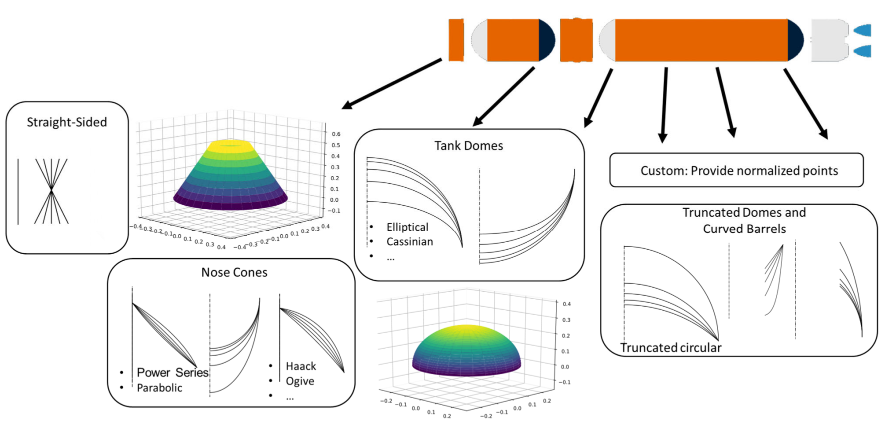


Fig. 3 Depiction of profiles in LVSA

understanding by using analytical equations to provide greater insight into the profile and structural entities. Each profile is defined using a set of input parameters, such as shape parameters, diameters, length, reference location, orientation, etc. with some minor variations depending on the selected type of profile. An example of the profile definition can be seen in Figure 4. The profile definition code uses an adaptive sampling algorithm to determine how many points are needed to define the profile. This algorithm involves estimating evenly distributed locations between profile end points, predicting midpoints between known points, and keeping points with error between predicted/actual. The final points are saved as splines using the SciPy library with the parametric distance along the curve as the input. Curves between profile points and swept around the central axis form bands, described as profile stations. This enables efficient calculation of derivatives and integrals that can be used to find many geometric properties such as centroid, curvature, normal angle and direction, tangent angle and direction, radius of curvature, surface area, and volume.

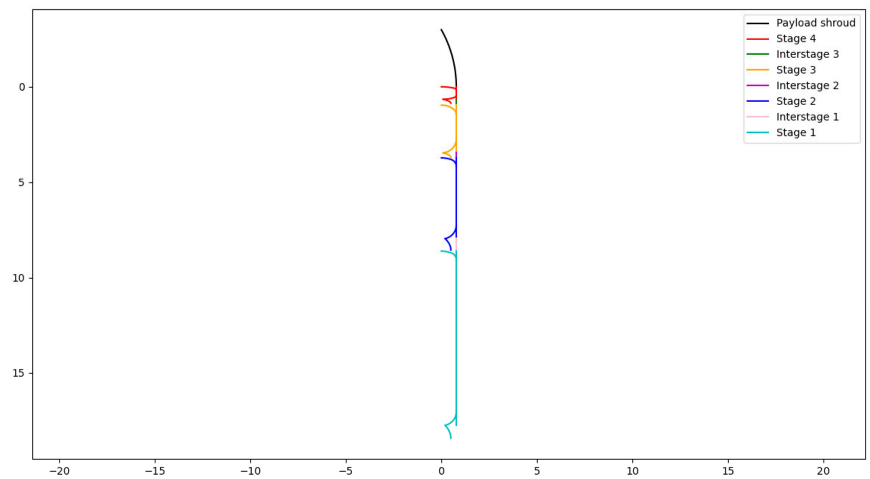


Fig. 4 Example illustrating axisymmetric profile definition in LVSA

While barrel and dome profiles provide geometric properties, the next step is to define structural and application information for the geometry to assign the physical properties of the vehicle components. Components are defined with the associated profile, stiffened or unstiffened panel information, joining regions between panels, and materials. The material properties are such that they can be used when LVSA interfaces with analytical or finite element structural models. At this point in the vehicle definition, the component can be identified as load bearing or not which can be used in loads analysis and structural modeling.

Once the components are created, the relationship between components must be defined. For example, a propellant tank may consist of an upper dome, barrel, and lower dome components. Multiple tanks and interfacing components may define a stage, stages are combined to form a vertical stack, and a vehicle is made up of one or more vertical stacks. Additional assembly classes include motor casings and nozzles. For example, the first stage assembly of an SLV is made up of a nozzle, lower dome of solid propellant tank, barrel of the tank, upper dome of the tank, and barrel of the casing. The stages are then assembled into stacks, and stacks can be assembled into cores and boosters depending on the design of the launch vehicle. The overall vehicle is assembled using the previously defined sub-assemblies. This is illustrated in Figure 5. The components and assemblies in LVSA have the capability to calculate properties such as mass, center of mass, moments of inertia, and volume via splines defined by the profile, material, and thickness information. These properties are necessary when sizing the structures of the SLV.

The last portion of the vehicle definition in LVSA is defining the time dependent characteristics such as flight conditions, propellant levels, staging events, and structural events, all referred to as load cases. The load cases are defined by the flight condition at a particular point in the mission. The current load cases are then assigned to corresponding components and assemblies in order to provide system information, generate models, etc. Once this process is complete, the vehicle is fully prepared for structural sizing. LVSA has several analysis capabilities including beam finite element method (FEM) for dynamic analysis, loads processing which processes aerodynamic and trajectory information to define structural loads as axial, bending, and shear distributions, shell FEM for analysis and structural optimization, and an analytical structural sizing routine based on stress and stability calculations for panels rotated about an axis. This paper utilizes LVSA's structural sizing analysis to perform analytical axisymmetric panel sizing based on stress and

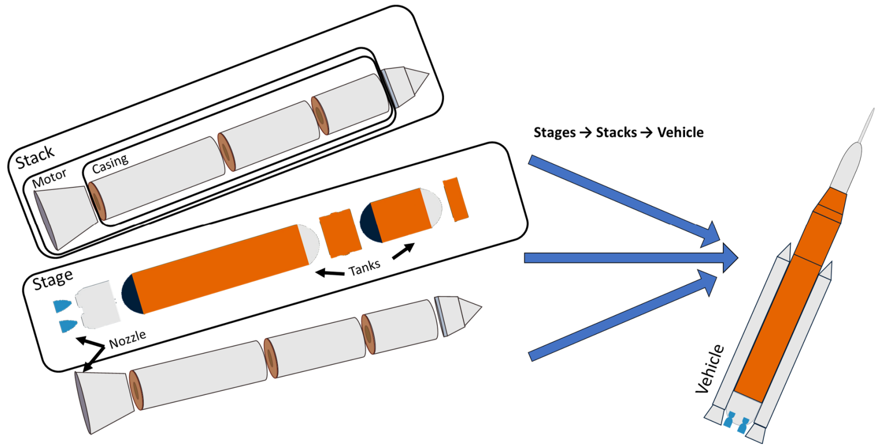


Fig. 5 Illustration of assembly definition in LVSA

stability. LVSA also utilizes data from SMAC to inform the structural sizing for propulsion components discussed in the next section.

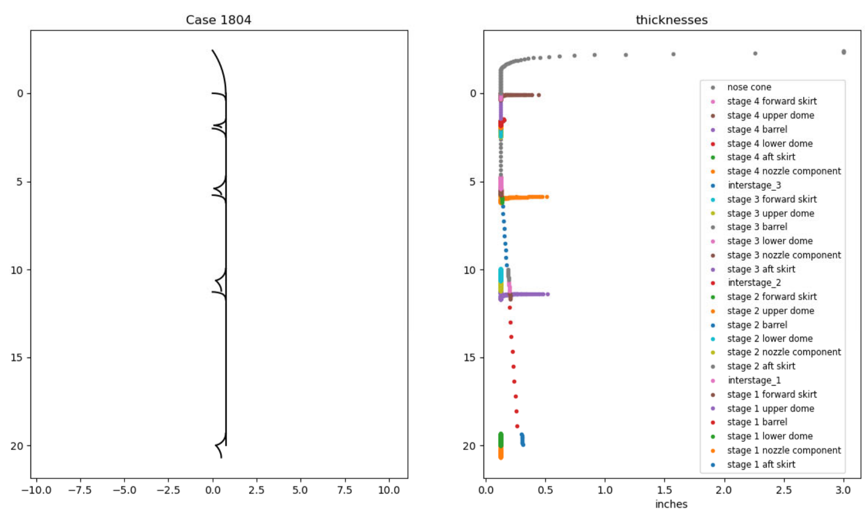


Fig. 6 LVSA profile with sized thicknesses for each component

C. Convergence Loop

The convergence loop, illustrated in Figure 1, manages the integration of SMAC and LVSA. The main point of the convergence loop is to use the data from SMAC and LVSA to converge on the radius of propellant grain in order to determine the allowable space in the motor casing for the propellant. SMAC's inputs and regressions are used to calculate the insulation thickness from insulation density and mass. The insulation mass is calcualted from a mass estimation equation involving propellant mass, burnout time, and propellant length to grain diameter ratio. The Maximum Engine Operating Pressures (MEOPs) and insulation thicknesses for each stage of the vehicle are passed to LVSA. Other parameters, such as nozzle exit diameter and propellant density, which are used for geometrical definition and stiffness calculations, are passed to LVSA from SMAC's computations. Using MEOP data, LVSA calculates motor casing thicknesses and diameters. The motor casing inner diameter is determined by subtracting the thickness from the diameter as seen in Equation 1, where d_{i_c} is the inner diameter of the casing, d_{stage} is the outer diameter of the casing for that stage, and t_c is the casing thickness.

$$d_{i_c} = d_{stage} - 2t_c \tag{1}$$

A check is necessary in order to determine the tolerance of the propellant grain and insulation thickness inside the motor casing. Thus, the grain diameter and insulation thickness determined by SMAC is added together, and the result is subtracted from the motor casing inner diameter as seen in Equation 2, where d_g is the grain diameter and t_i is the insulation thickness. If this difference is less than a specified tolerance, then it is considered converged meaning the propellant grain and insulation thickness occupy the allowable space inside of the geometrical dimensions of the motor casing as calculated by LVSA. However, if it has not passed the check and not converged, then a new propellant grain diameter must be determined for the next iteration in the loop. The new propellant grain diameter (d_p) is defined by subtracting the motor casing thickness (t_c) and insulation thickness (t_i) from the stage diameter (d_{stage}) as shown in Equation 3.The new value for propellant diameter is updated in the next iteration where the calculations and checks will proceed as previously described until each stage has converged.

$$tolerance = d_{i_c} - \left(d_g + 2t_i\right) \tag{2}$$

$$d_p = d_{stage} - 2\left(t_c + t_i\right) \tag{3}$$

D. Aerodynamics

The aerodynamic discipline is responsible for generating aerodynamic information that will be used by POST2. There is a wide selection of tools that handle aerodynamic analysis, and the most detailed of these utilize computationally expensive techniques. This cost is not well suited for design space exploration, so it is important for the selected tool to be capable of analyzing different vehicles at low computational costs. Missile DATCOM is a flexible and robust tool that was designed for aerodynamic analysis of missiles but has also been used for aerodynamic analysis of launch vehicles [12] [13] [14]. Missile DATCOM takes in a generalized physical description of the vehicle and current flight conditions and outputs a wide variety of aerodynamic coefficients (e.g. force, control). The required outputs of the process were determined by the needs of POST2, which requires a look-up table of aerodynamic coefficients (an aero deck) consisting of Mach number, altitude, angle of attack, and the corresponding drag coefficient.

To automate Missile DATCOM, a wrapper was built that runs the designs through appropriate aerodynamic conditions, compiles the data, and writes the aero deck. To account for stage jettison, the wrapper sequentially subtracts lengths of jettisoned stages from the SLV, generating additional aero decks. To allow for coasting phases, aero decks are also generated for "motor on" conditions (i.e. reduced base drag included) and "motor off" conditions (i.e. full base drag). Overall, for a vehicle with four stages, eight separate aero decks will be generated.

E. Trajectory

The trajectory discipline is responsible for determining a design's capability to reach an orbit and for maximizing the payload to said orbit. POST2 is a robust and well tested tool developed by NASA Langley, and it is one of the front-line tools that NASA uses for LV trajectory optimization [15] [16] [17]. It utilizes discrete parameter targeting and optimization to find the best path to the desired orbit. POST2 performs an N-phase simulation which allows for planet and vehicle models to be generalized. To describe the trajectory, POST2 uses a series of events (e.g. takeoff, stage jettison, etc.), where each event can have multiple control variables that POST2 can use in the optimization procedure [15]. The control parameters, in general, describe position and/or behavior of the vehicle during a certain time period (e.g. pitch rate, yaw rate, stage mass, payload mass, initial vehicle position and velocity, etc.). In general, this class of problem has multiple local minimums and infeasible starting points, which causes many of the POST2 runs to not converge on a solution if the initial conditions of the control parameter vector lead to an infeasible trajectory.

Due to the high volume and variety of designs that would be analyzed with this framework, it is impossible to find one set of feasible initial conditions for all of the potential vehicle configurations. In order to address this problem, the framework must search for initial conditions that would lead to convergence. This search was performed by conducting a Monte Carlo simulation over the vector that describes the initial conditions of the POST deck's optimization variables. Monte Carlo simulations have been applied to POST2 previously to increase the confidence that an optimal payload mass has been found [18]. While maximizing payload is still the goal of the presented framework, a variety of different SLV configurations makes it more difficult to find initial conditions that lead to a converged trajectory. This leads to the main change in application, where Zwack et al. used a Monte Carlo to find the most optimal payload mass for a specific vehicle; in the presented framework, a Monte Carlo simulation was utilized to improve the robustness of POST2 in order to analyze multiple launch vehicle designs [18].

III. Sample Problem

To demonstrate capability, the framework was applied to a problem requiring an SLV design capable of inserting a payload launched from Wallops Flight Facility into a 3636.6 by 3651.4 nautical mile orbit at an inclination of 47°. SLV mass is proportional to vehicle cost, so the overall goal is to minimize the SLV mass while maximizing the payload mass to orbit. It is unlikely that a single best solution will be found; instead the solution space will form a Pareto frontier. Furthermore, the data provided by the framework should allow for further analysis of the design space allowing for discovery of important design variables and trends.

The design space of an SLV is very large (variables include body shape, body size, presence and location of control surfaces, number of stages, nose cone shape and size, etc.), but not all of the options are feasible based on the mission needs. For this project the vehicle was defined to be a uniform stage diameter, four-stage solid rocket motor, non-lifting body SLV. The design space for this sample problem is defined by a selection of design variables. Even within the limitations of the sample problem, the design space is too vast to be fully explored. To gain useful knowledge about the problem, a design of experiment (DOE) will be utilized to identify the best parts of the design space to sample in order to obtain useful information about the whole of the design space. The DOE variables and the bounds placed on them will be described in the subsections below.

A. Defining the DOE

To define this vehicle, the stage length, vehicle diameter, nozzle diameter, and nose bluntness need to be specified. The selected variables are continuous and can lead to infeasible designs without limitation. To prevent infeasible designs and keep the vehicle within the size limits of an SLV, bounds were placed on the continuous variables. To create reasonable bounds, vehicles similar to an SLV were used for reference. The vehicles that would most closely resemble an SLV are sounding rockets (SRs). Thus, the designs of existing SRs can be used to define ranges for stage length and diameter. An example of such an SR is a 31 ft SS-550, which is capable of launching 8.8 lbs to orbit [19]. The derived bounds for stage length and diameter are shown in Table 3. The nose cone is defined by a fineness ratio which connects nose cone length to provided SLV radius, shown in Equation 4.

$$L_n = \frac{D}{2} * n_f \tag{4}$$

In Equation 4, L_n is the length of the nose cone, D is the diameter of the launch vehicle, and n_f is the fineness ratio. Next, the ranges for the nozzle lengths of each stage are defined, and the set up is shown in Table 3 and in Figure 7.

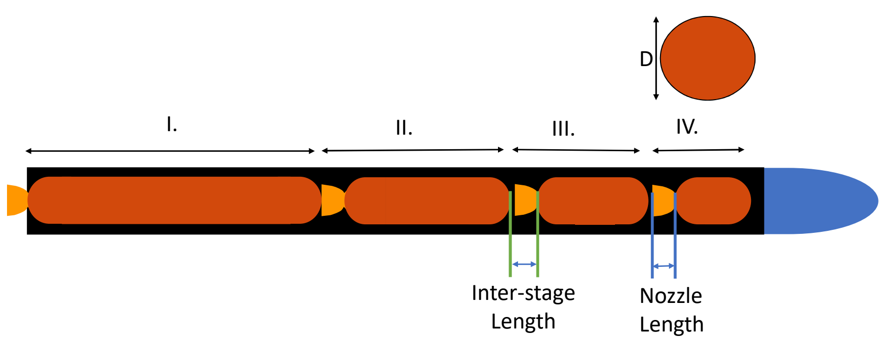


Fig. 7 Launch Vehicle Geometric Definition

For each stage a set of propulsion parameters were defined: outer and inner fuel diameters, width of fuel fin, grain type, propellant, MEOP target, and fuel star points. Four different grain types were available: bates, finocyl, starocyl, and star. From initial sizing runs it was determined that for the first stage, starocyl grain did not lead to designs capable of achieving the desired orbit, so that option was subtracted from the stage one options. Three propellants, for which the necessary information could be found publicly, were used: TP-H-1202, TP-H-3340, and TP-H-1148 [10]. As similar to grain type, preliminary runs showed that stage one required TP-H-1148 for the vehicle to be successful, so it was selected as the only option for that stage. Outer and inner grain diameters define empty spaces within the fuel grain; within the DOE this variable is a fraction of overall grain diameter. This relationship is shown in Figure 8 by the blue and red circles, respectively. The next input is the number of star points, which defines the number of protrusions from

Table 3 Vehicle DOE

	Stage I	Stage II	Stage III	Stage IV
Stage Length (ft)	8-10	4-6	2-4	0.5-2
Nozzle Length (ft)	0.2-0.8	0.2-0.8 1	0.2-0.6	0.1-0.3
MEOP Target (psi)	1000-1500	1000-1200	500-700	500-700
Outer Fuel Diameter (% of total)	0.05-0.1	0.1-0.15	0.1-0.15	0.1-0.15
Inner Fuel Diameter (% of total)	0.05-0.07	0.07-0.1	0.07-0.1	0.07-0.1
Width of Fuel Fin (% of total)	0.03-0.05	0.05-0.07	0.05-0.07	0.05-0.07
Fuel Star Points	2-4	2-4	2-4	2-4
Grain Type	bates,finocyl, star	bates, finocyl, starocyl, star	bates, finocyl, starocyl, star	bates, finocyl, starocyl, star
Propellant	TP-H-1148	TP-H-1202, TP-H-3340, TP-H-1148	TP-H-1202, TP-H-3340, TP-H-1148	TP-H-1202, TP-H-3340, TP-H-1148
		Overall		
Launch Vehicle Diameter (ft)	1.5-2			
Nose Shape Parameter	0.2-0.4			

the center to the outer circle diameter. Finally, the width of the star points is defined. The breakdown of grain type is shown in Figure 8, and the ranges are shown in Table 3.

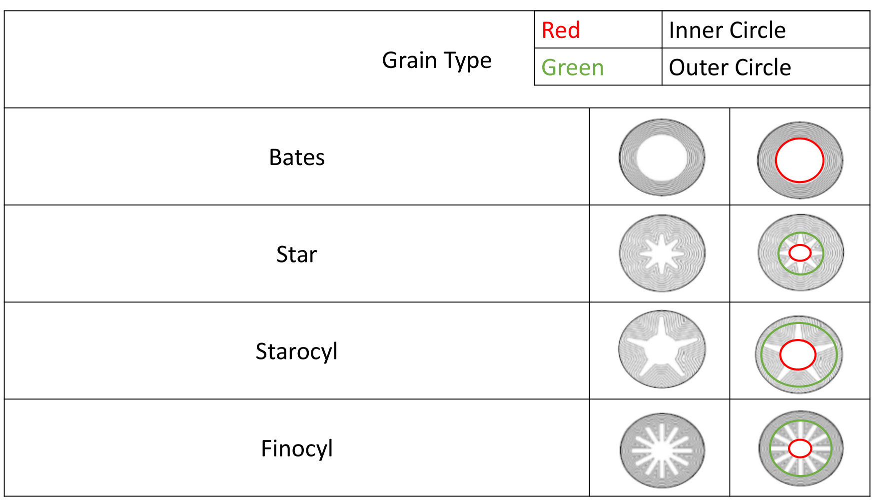


Fig. 8 Different grain types and radius definition

B. Aerodynamics

The inputs are a list of velocities (Mach), altitudes, and angles of attack (AoA). The Mach number array consists of fourteen elements from Mach 0.1 to Mach 26. The Mach range is defined by normal operational velocities of launch vehicles with Mach 25 being the usual velocity LVs reach during takeoff [20]. The altitude array consists of 9 elements from 500 ft to 250000 ft. The maximum altitude is defined by the maximum allowable altitude within Missile DATCOM (250000 ft). At an altitude of 250000 ft, the vehicle would already be in the upper parts of the mesosphere. Thus, any further drag effects could be considered negligible during preliminary analysis. The angle of attack array consists of 17 elements ranging from -20°to 30°. The AoA array was given a wide range to allow POST2 a wider margin that it can search without a need for extrapolation.

C. Trajectory

The trajectory design was made by following the generalized POST trajectory example shown in Waters et. al. [15]. As per example, additional stages were added into the process. Overall POST2 optimization is informed by 12 events: initial conditions, rail clearance, first stage jettison, second stage ignition, second stage jettison, shroud separation, pitch adjustment, third stage ignition, third stage jettison, second pitch adjustment, fourth stage ignition, and fourth stage jettison. A set of initial conditions defines dry and wet mass of the SLV including the initial guess for the payload mass (wstpd5). The initial conditions also define the location of launch (Wallops Flight Facility) by using the azimuth of the launch-centered inertial coordinate system. Finally, the angle at which the SLV is launched is also defined here (pitpc1). Next, in event two the first stage is ignited and the vehicle clears the rails once it reaches an altitude of 50 ft. Once the fuel mass of the first stage reaches zero, POST2 initiates the stage jettison. After waiting for one second, the second stage is ignited, following the same jettison procedure as the first stage. After the second stage is jettisoned, there is a 35 second wait until the shroud separates and the first of the pitching maneuvers occurs. During the pitch adjustment two variables control the event: the pitch rate (pitpc2) and the time since event occurrence (critr). This timer defines the coast phase and gives the optimizer the ability to control the velocity of the vehicles before the next stage is ignited. This control helps with trajectory optimization because with SRMs, it is impossible to directly control when it turns off. Therefore, indirect methods that can control when the stages are ignited allows for the vehicles to slow down to velocities that would enable proper orbital insertions. Ten seconds after the coasting phase is done, the third stage is ignited and then jettisoned. After this, the second pitch adjustment happens. Here, the coasting is defined by true anomaly (critr2). Similar to stage three ignition, there is a ten second timer before the fourth stage is ignited. The jettison of the fourth stage happens when the SLV reaches its final destination. Here a leftover mass variable (mass s 4) is introduced. This is done to add additional flexibility and improve optimizer performance. The allowance for leftover fuel was set up to only be a fraction of available fuel. The low amount should not physically affect the design, but it could significantly improve convergence. The breakdown of events and associated independent variables is given in Figure 9. The ranges for the variables in the Monte Carlo simulation are shown in Table 4.

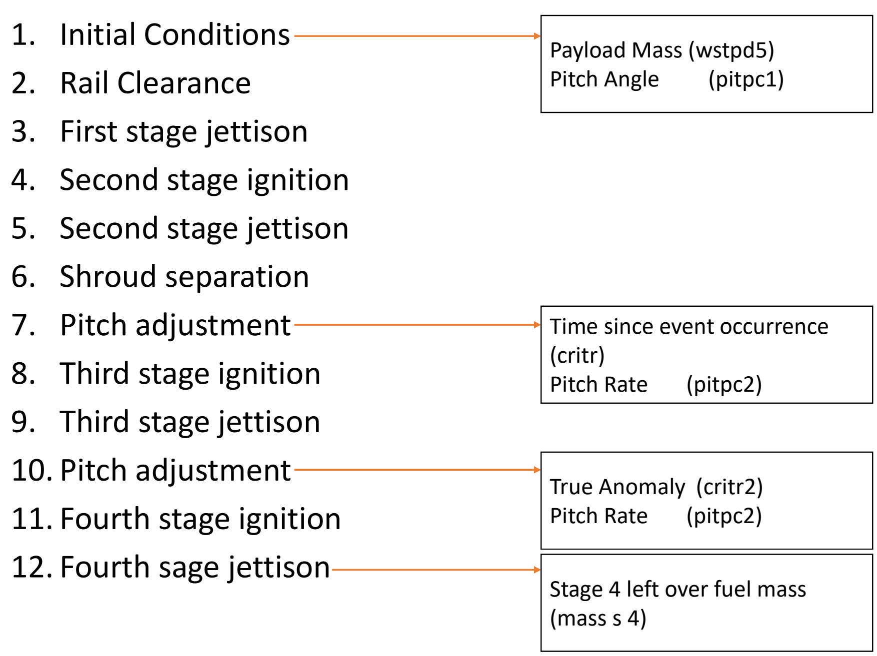


Fig. 9 Connection of independent variables to events

Table 4 Monte-Carlo simulation ranges

	Min	Max
critr (s)	1	80
critr2 (°)	100	185
wstpd5 (lb)	10	80
pitpc2 event 10 (°)	-15	0
pitpc1 (°)	-20	0
pitpc2 event 7 (°)	-20	-5
mass s 4 (lb)	0.5	30

IV. Results

Six thousand different SLV designs were analyzed in the framework as part of the DOE, and all cases that reached the desired orbit were visualized in Figure 10. Each point in this scatter plot represents a unique SLV design. For example, if SLV 4855 is selected, Figure 11 shows the extracted trajectory characteristics of that vehicle. Next, the sizing outputs are shown in Table 5. Table 5 shows all the basic geometric definitions of this SLV. For each stage, fuel and structural masses are calculated as well as thicknesses for rocket motor insulation and rocket motor casing. By utilizing this framework, a designer is be able to quickly analyze different SLV designs and have more detailed information available on demand.

The goal of this sample problem is to identify specific areas in the SLV design space for local optimization in subsequent cycles of design, and the objective of the SLV design problem is to maximize payload to orbit while minimizing the vehicle's wet mass. As expected, the solution space formed a Pareto frontier, shown by the golden diamonds in Figure 10, where additional payload to orbit is traded for additional SLV wet mass. These Pareto points represent the best possible designs within the explored space and are candidates for further development.

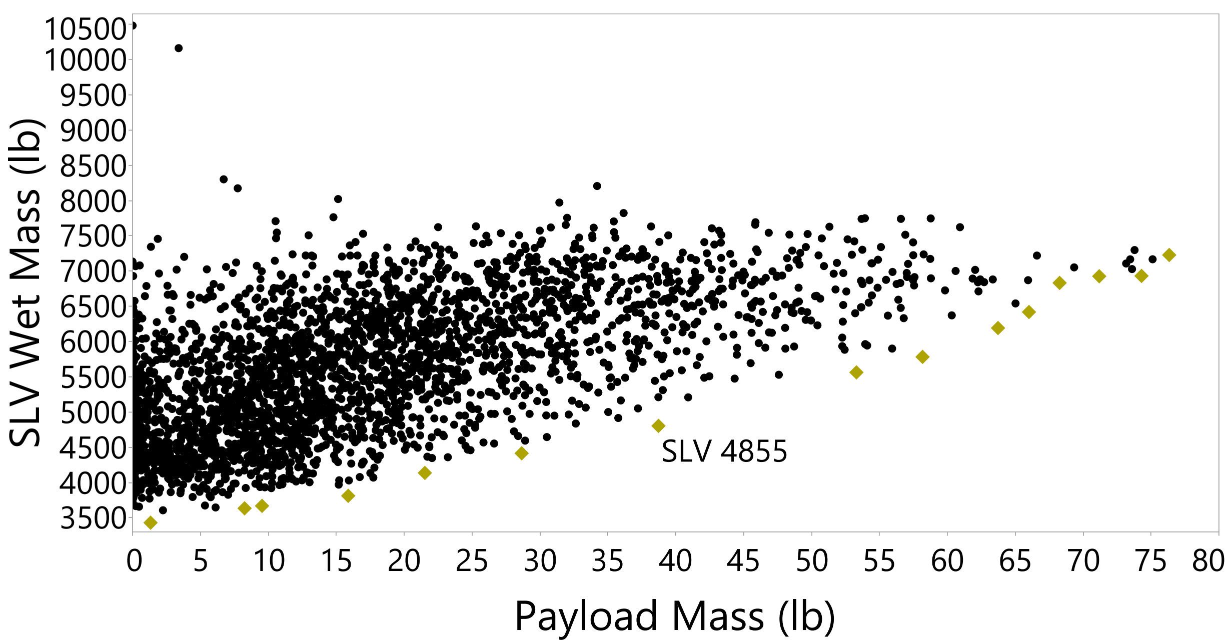
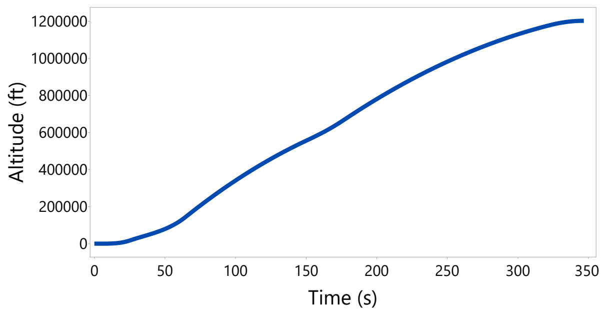
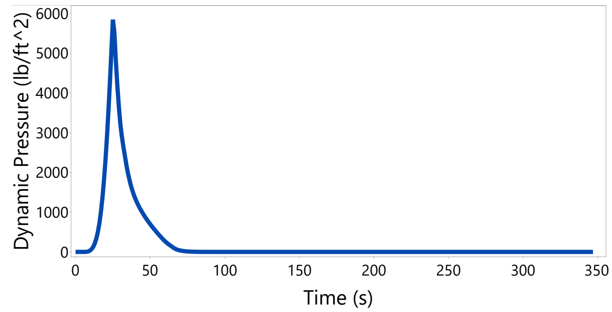


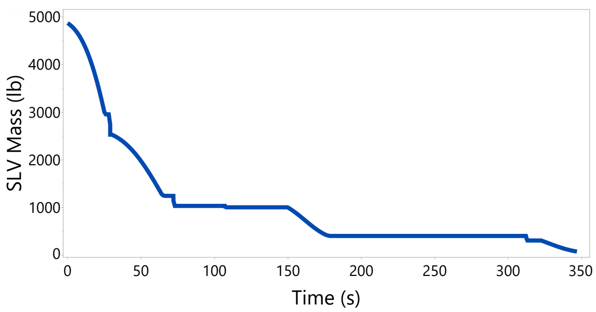
Fig. 10 SLV Wet Mass (lb) vs Payload Mass (lb)



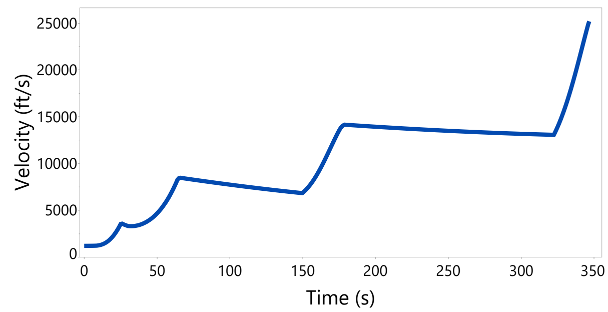
(a) Altitude (ft) vs Time (s)



(b) Dynamic Pressure (lb/ft^2) vs Time (s)



(c) SLV Mass (lb) vs Time (s)



(d) Velocity (ft/s) vs Time (s)

Fig. 11 Trajectory of SLV 4855

V. Conclusion

The primary goal of this work was the creation of a framework capable of multidisciplinary design space exploration to aid in the conceptual design of SLVs. In order to achieve this goal, a set of disciplines that are vital to conceptual SLV design were identified (trajectory,propulsion,aerodynamics, and structures). For each discipline a tool was selected (POST2, SMAC, Missile DATCOM, and LVSA, respectively). Finally, these tools were integrated together in order to create the desired framework.

The framework was then utilized on a sample problem of an SLV capable of carrying a payload into 3636.6 by 3651.4 nautical mile orbit. The framework was capable of rapidly exploring the design space and giving results for the preliminary characteristics of structures, aerodynamics, and propulsion. This led to the identification of Pareto optimal SLV designs that could be carried into further design cycles. This shows the utility of the framework in enabling designers to generate conceptual design data to find promising designs and conduct high-level trades.

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Table 5 Data for SLV 4855

	Stage 1	Stage 2	Stage 3	Stage 4
Stage Length (ft)	9	6	3.23	1.15
Nozzle Length (ft)	0.59	0.23	0.52	0.19
MEOP target (psi)	1174.5	1017	548.8	511.15
Insulation Thickness (in)	0.11	0.12	0.088	0.094
Wall Thickness (in)	0.22	0.15	0.125	0.125
Fuel Mass (lb)	1916	1286	601	232
Structures Mass (lb)	425	210	95	36
Outer Fuel Diameter (% of total)	0.098	0.15	0.13	0.11
Inner Fuel Diameter (% of total)	0.059	0.094	0.096	0.087
Width of Fuel Fin (% of total)	0.03	0.06	0.054	0.051
Fuel Star Points	4	3	2	2
Propellant	TP-H-1148	TP-H-1202	TP-H-1148	TP-H-1202
Grain Type	Star	Starocyl	Finocyl	Star
Overall				
Launch Vehicle Diamter (ft)	1.7			
Nose Shape Paramter	0.23			
Payload Mass (lb)	39			
SLV Wet Mass (lb)	4803			
SLV Dry Mass (lb)	767			

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