17 Annual USU/AIAA Small Satellite Conference

Responsive, Low-Cost Access to Space with ELVIS — an Expendable Launch Vehicle with Integrated Spacecraft

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SSC03-II-5
August 11 – 14, 2003

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Responsive, Low-Cost Access to Space with ELVIS — an Expendable Launch Vehicle with Integrated Spacecraft*

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Abstract. The ELVIS (Expendable Launch Vehicle with Integrated Spacecraft) concept involves: (1) dropping off the upper stage of the launch vehicle as low as possible, with integral low-thrust propulsion taking the spacecraft to its final orbital destination; (2) using the spacecraft bus to provide the avionics functions needed to fly a launch vehicle to orbit so as to avoid the duplication of avionics hardware and software between the satellite bus and the launch vehicle. The result is a reduction in the parts count, weight, and cost of the launch vehicle. There are major benefits associated with early staging — the upper stage can reenter safely without a retro burn, and the mass-to-orbit available from small launch vehicles is significantly increased. The mass gain will depend on the hardware configuration and the orbit destination, but can be as much as a factor of two or more for some low Earth orbits. In addition, the spacecraft bus operates from the time of launch and can begin the mission essentially as soon as the spacecraft reaches its operational orbit or, in some cases, even before. The small spacecraft thus achieves a new level of responsiveness, allowing spacecraft to be launched in response to rapidly changing circumstances. This paper describes a representative ELVIS configuration and performance gains for typical mission destinations, and sample applications that are enabled or made more efficient by the use of this approach. Technical issues and tradeoffs associated with this design will be discussed.

Introduction

The costs of getting to space and operating in space continue to be the barriers that constrict the small satellite market. There are several ways to reduce or remove these barriers that can be used separately or in combination. One is to utilize low-cost launch vehicles that are emerging in the industry, such as the Sprite[1, 2], that will drop launch costs by as much as an order of magnitude and add responsiveness at the same time. Another is to use the avionics in the spacecraft bus to control the launch vehicle and eliminate the need for duplication of hardware. The major benefit from the dual use of the avionics is to have a fully functioning spacecraft at orbit insertion, immediately ready to begin performing a mission. A third option, a technique termed the Modified Launch Mode[3–7] (MLM), can be applied to maximize the system performance and significantly increase the payload mass that reaches the operational orbit. Finally, we can also reduce the systems cost by taking advantage of proven autonomous orbit transfer and control

methods to further enhance the system capabilities and reduce operations costs. These four items are particularly valuable because they can be used in nearly any combination to create an exceptionally flexible system to meet specific needs.

This paper focuses on the second and third options, looking at a specific mission segment niche where a payload is looking for both a ride and a launch. The calculations used in this example assume the Microcosm Sprite as the launch vehicle. Sprite is a launch vehicle with pressure fed engines that has been designed with the objective of placing 318 kg (700 lbm) into a 185 km (100 n. mi.) circular reference orbit due East from the launch site. For the representative missions analyzed, there is a savings on the order of $90K in avionics hardware costs, a 40% reduction in software development costs, an increase in mass to orbit of between 15% and 107%, and a decrease in integration/test costs that depends on the particular launch vehicle, but could be substantial (savings estimated to be about the same as the avionics hardware cost savings).

Dual Use Avionics

With the spacecraft now encompassing the former avionics capabilities of the upper stage, the result can be thought of as a "Stagecraft". Conceptually, this Stagecraft will consist of three basic modules (Figure 1):

- Module 1: Either the traditional launch payload (mission payload plus satellite bus) or just the mission payload;

- Module 2: The avionics bay, which can also serve as the spacecraft bus; supplying power; telemetry, tracking, and command; navigation; attitude control; and a structural/thermal interface for the mission payload;

- Module 3: A restartable upper stage with a low-thrust engine. Low thrust here is defined to have an acceleration of 0.01g to 0.1g, which translates into an engine in the range of 90 N to 900 N (20 lbf to 200 lbf) thrust. Even smaller thrust motors, in the 5 N to 20 N range, may be sufficient. The propulsion module could be included or not included as appropriate. Further, the tanks could be varied in length depending on the needs of the mission. Finally, the propulsion module could be dropped off as in the standard upper stage process or kept along with the avionics to serve as the spacecraft propulsion module.

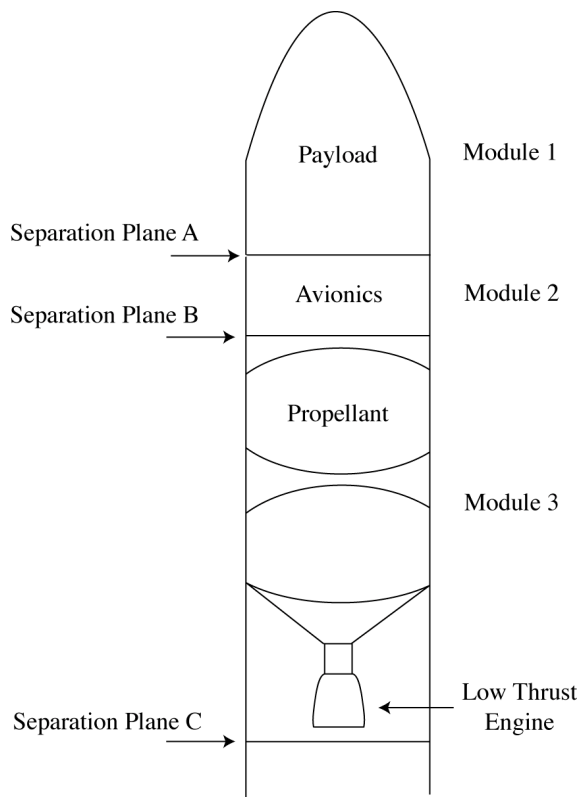

Figure 1: Stagecraft Concept

Although an objective of the ELVIS concept is to eliminate the upper stage avionics, and replace them with space-qualified components as part of the modified

spacecraft, this is not the whole story. There are cost and weight savings due to the elimination of the upper stage avionics. Although the Stagecraft will carry more propellant and have a somewhat heavier tank as a result, the overall system will be lighter or no heavier because less propellant is needed for the third stage. In addition, the batteries for the Stagecraft do not need to be larger than for the spacecraft by itself because the solar panels can be activated shortly after the shroud is jettisoned to allow immediate battery charging.

In addition to hardware savings in the ELVIS concept, the software should also be considered in terms of its contribution to cost savings. We estimate that there is about a 40% reuse factor in terms of software development, particularly in support and infrastructure components. Thus, the software will be available already for the launch vehicle, so the cost of developing software for ELVIS is only 60% of the cost of developing the baseline spacecraft software. Finally, test and integration costs and risks will be reduced simply because there is less to test and integrate.

In terms of the spacecraft avionics configuration, to date most spacecraft are "custom" built, out of necessity. Depending on the mission and the payload, the spacecraft requirements are derived. However, with a launch vehicle, the mission is almost always the same — get the spacecraft to orbit. Thus, a central factor in the viability of ELVIS is to assume a major increase in the variety of possible payloads to have the versatility to be able to respond to a range of possible missions. This assumption is consistent with the umbrella of Responsive Space (i.e., the ability to respond quickly with a combination of launch vehicle and spacecraft to time critical military or science events), which has been gaining considerable notice in the last few years.[8] For example, one scenario that meshes well with the ELVIS concept is the deployment of multiple cameras (or surveillance equipment) in a quick response to a threat. Using the approach that there will be a subset of payloads ready to be integrated into a Stagecraft on short notice allows the development of several baseline spacecraft concepts. It is then possible to create a cost effective development plan for augmenting their avionics with the upper stage requirements.

Table 1 lists components that typically would be part of the third stage, along with estimates of representative direct hardware costs for each. The net hardware cost savings associated with ELVIS is $90K, compared to the conventional avionics path. Although not large when considering large spacecraft, $90K nevertheless is a substantial amount that could be saved in the context of small spacecraft. Also, since most direct costs carry appropriate mark-ups, the price to the customer could be more than twice this amount, plus charges to the customer for additional test and integration work for avionics that is now no longer required. There will also be some weight savings directly associated with the reduced avionics hardware, along with weight savings in terms of propellant and structures. Both cost and weight savings associated with the avionics become significantly more important as the launch vehicle decreases in weight and cost. For upper stage/payload combinations under about 500 kg, the avionics is still sufficiently expensive to warrant adopting the ELVIS concept.

Table 1: Avionics Costs

Avionics Components	Cost ($K)
Flight Computer	10
IMU/GPS	25
Batteries	5
Communications/Telemetry	50
TOTAL	90

If the third stage engines are gimbaled, it is also possible that the Stagecraft could control the stack of the third stage/Stagecraft after the shroud is jettisoned. Then some of the third stage propulsion system associated with controlling the attitude of the third stage could be eliminated. There would then be a direct mass savings and a cost savings both for the components that have been eliminated and the integration and testing associated with the additional components.

Modified Launch Mode (MLM)

The MLM involves using propulsion on-board the spacecraft to eliminate the need of carrying the full weight of a separate upper stage to a high-energy orbit. The MLM was originally proposed by Microcosm for interplanetary missions.[3–5] However, a similar increased payload benefit has been found to apply to missions in the vicinity of Earth. For Earth orbit missions, the upper stage would continue on to the mission orbit, but the stage would become part of the spacecraft such that it is, in effect, integral propulsion. Essentially, either existing propulsive capability is augmented (more propellant and larger propellant tanks), or propulsive capability is added (propellant, a large thruster, and a large propellant tank), so that the spacecraft provides a significant portion of the delta-V requirement for the mission. The launch vehicle, in turn, provides a lesser amount of delta-V than the original mission requirement because a heavier spacecraft is launched. For most scenarios that include a high specific impulse on the spacecraft and a large ratio of final stage burnout mass to spacecraft mass, this change is very beneficial.

The comparable relative payload mass benefits obtained by applying the MLM launch mode both in the Earth orbit and the interplanetary mission context are derived by using the most advantageous sequence of dead-mass drop off, in each of the two very different mission classes. This gain is obtained even with the very great difference, by at least an order of magnitude, in the delta-V requirements of these mission classes. Note that the simple payload gain equation shown in the reference papers[4-6] does not directly apply in the Earth orbit mission case, because of the difference in the specific launch sequence.

The process is illustrated in Figures 2 and 3. Figure 2 depicts the launch sequence for a launch vehicle that uses, in effect, four separate stages. For MLM the Stagecraft performs the functions of a fourth stage. For a conventional, three-stage vehicle, the third stage boosts the spacecraft to its final orbit, and then must be jettisoned, requiring that it be capable of performing a deorbit burn. Figure 3 shows a comparison between the conventional (3A) and MLM (3B) ways of reaching the mission orbit. The conventional sequence involves launching into a parking orbit (185 km circular in this example), transfer to the mission orbit (700 km circular, for example), and the deorbit of the transfer stage. Using the MLM involves launching into an elliptical orbit with a perigee low enough (e.g., 50 km) that the transfer stage (Stage 3) reenters with no need for a deorbit burn after it is jettisoned by the Stagecraft at apogee, which then uses its integral propulsion immediately to begin the transfer to the mission orbit.

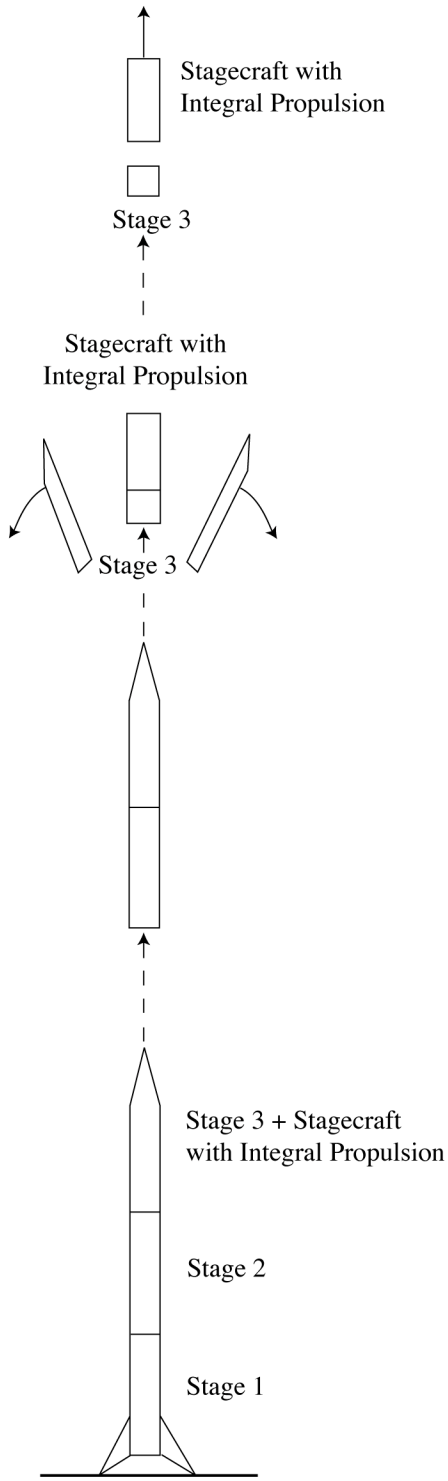

Figure 2: Conceptual MLM Launch Sequence

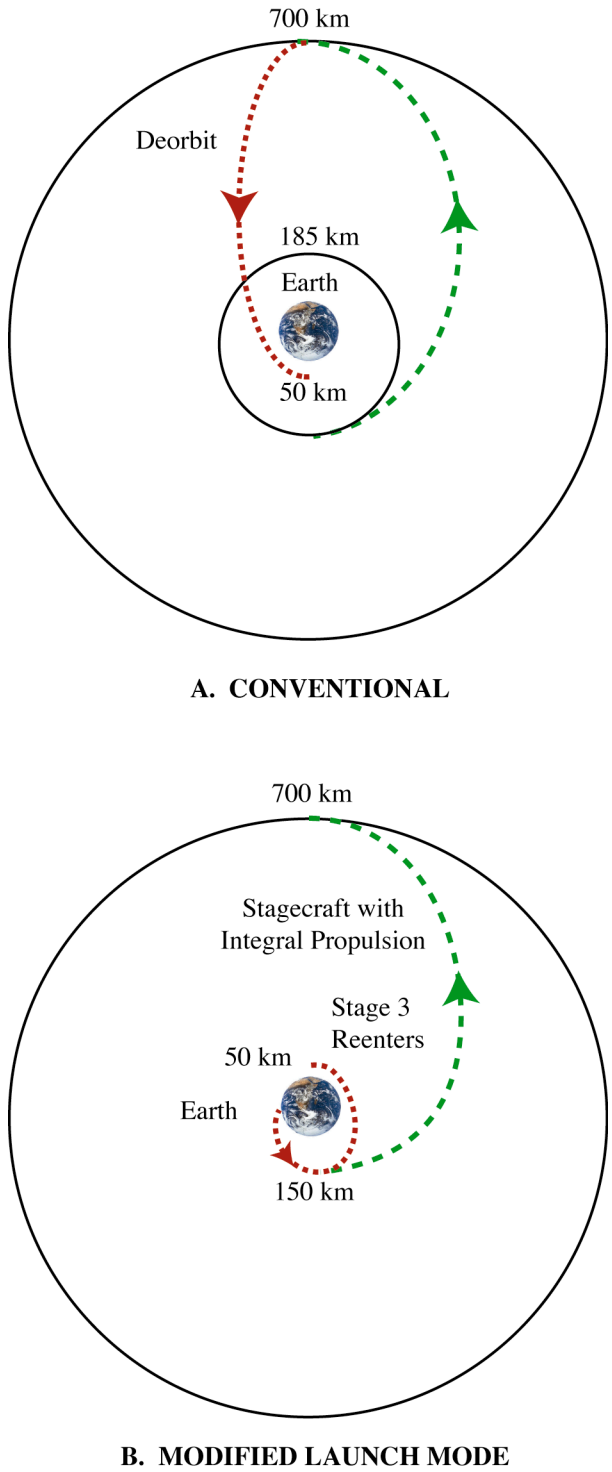

Figure 3: Methods of Reaching Mission Orbit

The MLM represents a paradigm change in the way space missions are designed because it blurs the line between what is part of the launch vehicle and what reaches orbit to accomplish a mission. As a result, the following represent several practical considerations that must be evaluated, but which we believe do not change the fundamental conclusions:

1. Increased propellant, pressurant, and propellant tank weights — increases in these quantities in the Stagecraft are offset by decreases in the corresponding weights for the lower stages — the mass fraction of the Stagecraft now becomes greater, but this trade has minimal effect, with a secondary impact on the attitude control system;
2. Rapid checkout and operation of the spacecraft following separation — not particularly unusual, since the standard launch process also involves rapid sequencing of events involving stage separations;
3. Changes to the launch vehicle to accommodate the MLM — instead of the launch vehicle and payload being considered as essentially two separate and almost unrelated entities (except for a few issues such as power transfer and separation pyrotechnics), each will have to be designed with the other in mind. This is not a significant issue when both designs start with the MLM concept in mind.

For the results discussed in the Applications section, it was assumed that the spacecraft already has the means necessary to execute translational propulsive burns of sufficient magnitude to perform the mission. Normally, such a spacecraft would have the capability for other maneuvers such as retro-braking into an orbit about a planet for an interplanetary mission, orbital plane changes for a reconnaissance satellite, or station-keeping. Such a capability would include thrusters, propellant tanks, pressurization, and a guidance, navigation, and control system for the burn.

For increased payload delivery, the MLM necessarily requires the addition of mass to the Stagecraft. However, the total mass of the third stage and Stagecraft combination remains approximately constant, since the third stage quits sooner in this configuration, but the Stagecraft starts using propellant (for translation) sooner. The added translation delta-V means that propellant must be added to the spacecraft. As a result, there will also be increases in propellant tank size (and mass) and pressurant mass that will be compensated by a reduction in these quantities in the third stage.

Most launch vehicle trajectories require a rapid sequence of burns for maximum performance. This is especially true when attaining orbit initially and when making large “departure” burns near perigee, such as for interplanetary injection. This was the case for which the MLM was originally conceived. For insertion into low Earth orbit, a minimal orbital altitude must be reached during the first orbit or the spacecraft will reenter. Trajectories that require this rapid sequence of burns place checkout and operational constraints on the spacecraft. The spacecraft must execute its large translational burn after a delay of on the order of seconds to a few minutes after separation from the last stage of the launch vehicle. The operational effect of the tight time constraints must be looked at closely, but are not considered specifically here. However, there certainly are precedents for rapid sequencing of events in space missions, especially during the launch phase of virtually every mission to date, which would argue that the operations impact is not excessive for this implementation.

Table 2: Mass Savings Calculation Input Parameters

Parameter Description	Parameter	Comment	Example (Sprite)
Mass to 185 km	M_i	Launch Vehicle Specific	209 kg
Final Third Stage Mass	M_{fs}	Launch Vehicle Specific	281 kg
Conventional Specific Impulse	I_{spc}		323 s
Spacecraft Specific Impulse	I_{spsc}		330 s
Conventional Initial Circular Orbit Altitude	H_{ci}		185 km
Mission Orbit Altitude	H_m		740 km
MLM Drop-off Apogee	DO_a		150 km
MLM Drop-off Perigee	DO_p		50 km
Disposal Altitude	DA		50 km
Earth Radius	R_e		6378.14 km
Earth Gravitational Parameter	μ		398600.44 km^3/s^2
Acceleration of Gravity	g		0.009807 km/s^2
Orbit Raising and De-orbit Mass Penalty	MP	Launch Vehicle Specific	5 kg

As shown in Figure 3B, the MLM utilizes a separation that could be below the circular parking orbit altitude of 185 km. This is the altitude from which the transfer stage would boost the spacecraft to its mission orbit in the conventional scenario.

The mathematical basis for the results that follow is provided in Table 3, with inputs defined in Table 2 and then applied to some orbital geometries that might have scientific or military interest.

Applications

Some specific numerical examples will now be presented that demonstrate the mass gains achievable with the implementation of the MLM applied to the Sprite launch vehicle. Note that the Sprite launch vehicle has been designed to place 318 kg (700 lbm) into a reference circular, due east orbit that has a 185 km altitude, at very low cost. However, a wide range of other low-cost orbital altitude, inclination, and payload combinations are achievable by the Sprite launch vehicle in its current configuration. The reference Sprite orbit and payload mass will now be used to show how implementing the MLM can further expand the system capabilities. Results, summarized in Table 4, include two cases for each mission altitude that correspond to a Stagecraft with two different types of onboard propulsion: (1) monopropellant hydrazine thrusters (I_{sp} = 215 s), and (2) bipropellant thrusters (I_{sp} = 330 s). The Mass Penalty included in the Conventional Orbit Transfer method reflects the upper stage propellant reserve needed for orbit raising and deorbit, which is therefore not available for initial insertion. Drop-off for the MLM is assumed to be at the 150 km apogee of an orbit that has a perigee of 50 km, which will lead to immediate reentry for the upper stage. The shaded areas in Tables 2 and 3 correspond to values listed in the fifth column of Table 4.

Table 3: MLM Mass Savings Calculations

Description	Equation	Example Results
CONVENTIONAL TRANSFER		
Conventional Orbit Circular Velocity Magnitude	(1) $V_c = \sqrt{\mu / (R_e + H_{ci})}$	7.793 km/s
Mission Orbit Circular Velocity Magnitude	(2) $V_m = \sqrt{\mu / (R_e + H_m)}$	7.483 km/s
Total Transfer delta-V	(3) ΔV_{Tc}, absolute value of Eq. 1 minus Eq. 2	0.310 km/s
Transfer Propellant Mass	(4) $\mathbf{M_T} = (M_i + M_{fs})\left\{1 - e^{-\Delta V_{Tc}/(gI_{spc})}\right\}$	46 kg
Disposal Circular Velocity Magnitude	(5) $V_d = \sqrt{\mu / [R_e + (H_m + DO_P / 2)]}$	7.671 km/s
Total De-orbit Velocity Magnitude	(6) ΔV_{Td}, absolute value of Eq. 5 minus Eq. 2	0.188 km/s
De-orbit Propellant Mass	(7) $\mathbf{M_d} = M_{fs}\left\{e^{\Delta V_{Td}/(gI_{spc})} - 1\right\}$	17 kg
Net Conventional Transfer Payload Mass	(8) Initial mass M_i minus Eq. 4 minus Eq. 7 minus Orbit Raising and De-orbit Mass Penalty MP	141 kg
MLM TRANSFER		
Drop-Off Circular Velocity Magnitude	(9) $V_{DO} = \sqrt{\mu / [R_e + (DO_a + DO_P / 2)]}$	7.844 km/s
Drop-off Savings delta-V	(10) ΔV_{DO}, absolute value of Eq. 9 minus Eq. 1	0.051 km/s
Drop-off Propellant Savings	(11) $\mathbf{M_{DO}} = (M_i + M_{fs})\left\{e^{\Delta V_{DO}/(gI_{spc})} - 1\right\}$	8 kg
MLM Transfer Initial Mass	(12) Initial mass plus Eq. 11	217 kg
MLM Transfer delta-V	(13) ΔV_{MLM}, absolute value of Eq. 9 minus Eq. 2	0.361 km/s
MLM Transfer Propellant Mass	(14) $\mathbf{M_{MLMp}} = M_{MLMi}\left\{1 - e^{-\Delta V_{MLM}/(gI_{spsc})}\right\}$	23 kg
MLM Net Payload Mass	(15) Eq. 12 minus Eq. 14	194 kg
RESULTS		
Mass Gain	(16) Eq. 15 minus Eq. 8	53 kg
Mass Gain (%)	(17) (Eq. 16/Eq. 8) × 100	38 %

Table 4: Performance Enhancements Due to the Modified Launch Mode for Two Different Types of On-Board Propulsion Systems

Target Orbit	700 km due East	700 km due East	740 km Sun Sync	740 km Sun Sync	1,300 km Sun Sync	1,300 km Sun Sync
Spacecraft I_{spsc} (s)	215	330	215	330	215	330
Conventional Orbit Transfer (I_{spc} = 323 s)						
Mass to 185 km (kg)	318	318	209	209	209	209
Orbit Raising Propellant (kg)	–51	–51	–46	–46	–81	–81
Deorbit Propellant (kg)	–17	–17	–17	–17	–31	–31
Mass Penalty (kg)	–8	–8	–5	–5	–11	–11
Payload to Destination (kg)	242	242	141	141	86	86
MLM Orbit Transfer						
Mass to 185 km (kg)	318	318	209	209	209	209
Low Drop-Off Savings (kg)	+10	+10	+8	+8	+8	+8
Orbit Raising Propellant (kg)	–48	–33	–34	–23	–57	–39
Payload to Destination (kg)	280	295	183	194	160	178
Payload Gain (kg)	**38**	**53**	**42**	**53**	**74**	**92**
Payload Gain (%)	**15**	**22**	**30**	**38**	**86**	**107**

Note: Bounding calculations have shown that even if the Stagecraft/Third Stage mass combination does not remain approximately constant, the effect is to reduce the payload gains only by between 3% and 7% across the range of missions shown in this table (e.g., 15% reduces to 12% and 107% reduces to 100%).

From the table, it can be seen that a traditional Sprite launch capable of placing 318 kg in low earth orbit due East can put about 86 kg in a 1300 km Sun synchronous orbit. However, with the MLM, the payload can increase to 178 kg, which is not too much less than the 209 kg that can be delivered to the baseline altitude of 185 km at the same low cost. The bipropellant thrusters add 11 to 18 kg to the payload capacity, which must be traded against using the operationally simpler monopropellant thrusters. Thruster selection is therefore a subject for more detailed trades for specific missions.

Figures 4–6 provide the expanded data set from which the specific examples discussed above were extracted. Besides the results for a 50 km perigee, results for a 100 km perigee have been included in case there are operational reasons why the lower perigee cannot be used. For all cases shown, there is only a small loss (< 1.3%) when the perigee altitude for the third stage is increased from 50 km to 100 km. Clearly, the perigee could be raised even more with only a small additional mass penalty. However, in all cases, there is a significant benefit in terms of payload mass increase by using the bipropellant over the monopropellant thrusters (7%–21%).

The data shown in Figures 4–6 were intentionally cut off once the semi-major axis of the injection orbit for the MLM was 185 km, the altitude of the circular parking orbit. This is the altitude for which the Sprite launch vehicle was optimized. The analysis can be repeated using information on the optimized performance of other candidate launch vehicles. Further, the particular mission has to be analyzed to determine what the best injection orbit should be.

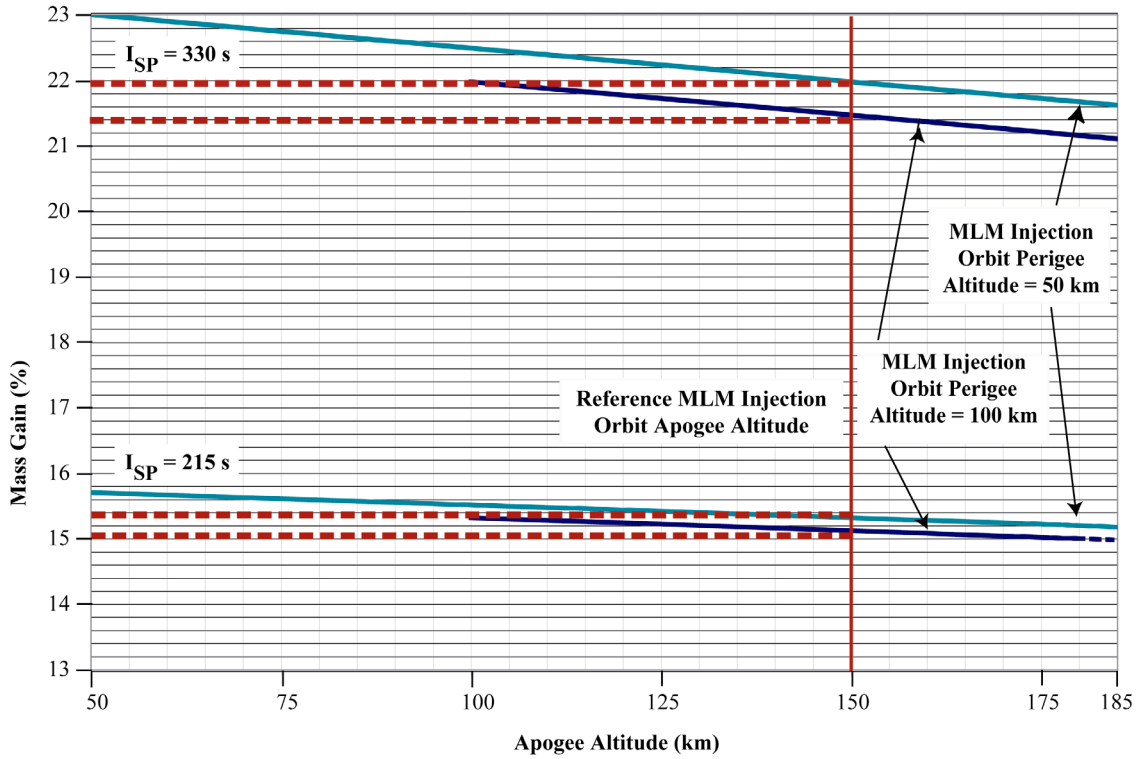

Figure 4: Mass Gain (MLM vs. Conventional) as a Function of Injection Apogee Altitude, Due East Launch, Final Orbit Altitude 700 km Circular

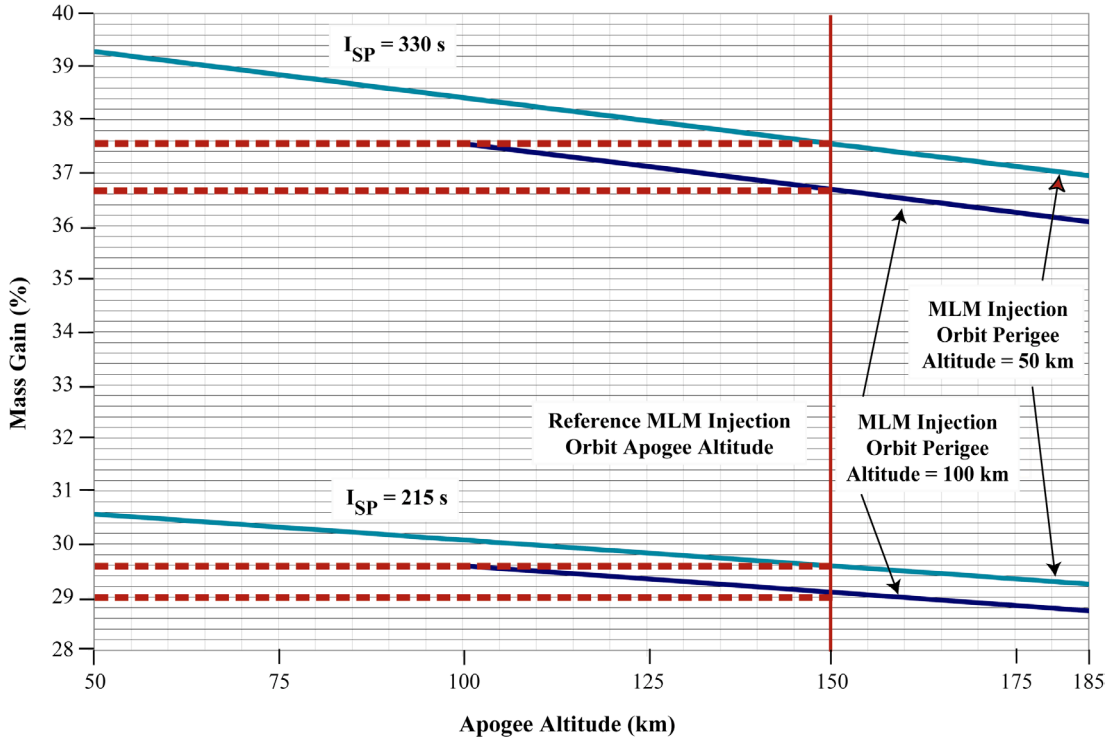

Figure 5: Mass Gain (MLM vs. Conventional) as a Function of Injection Apogee Altitude, Sun Synchronous Orbit, Final Orbit Altitude 740 km Circular

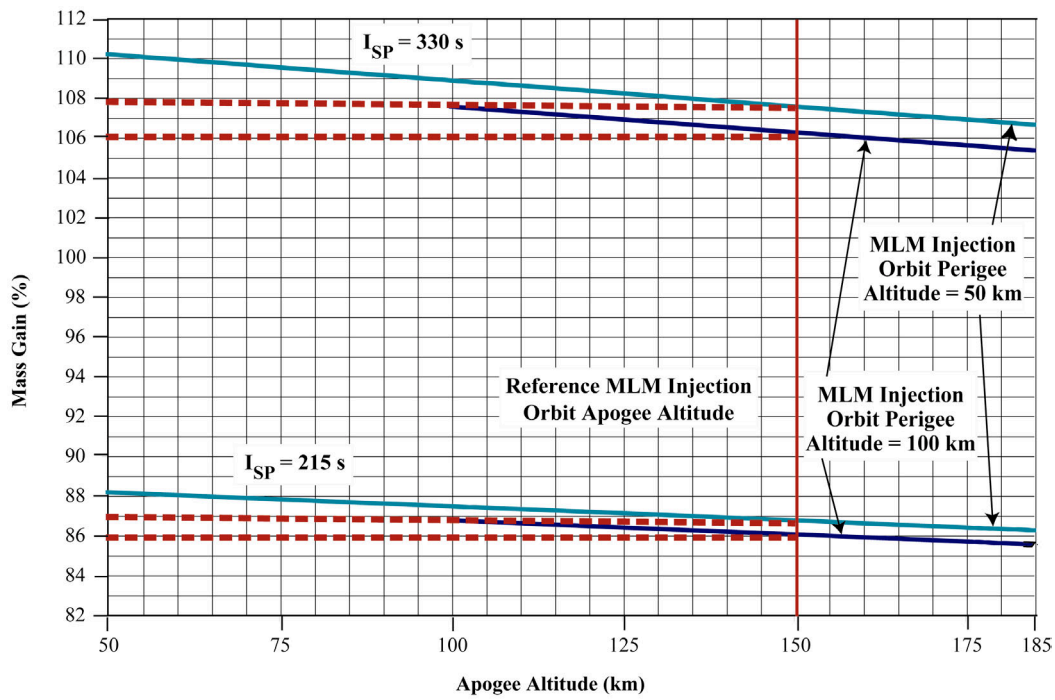

Figure 6: Mass Gain (MLM vs. Conventional) as a Function of Injection Apogee Altitude, High Sun Synchronous Orbit, Final Orbit Altitude 1300 km Circular

Dropping the spacecraft off early requires a reasonable propulsion level to keep from reentering. For the example drop-off orbit of 150 km × 50 km with the spacecraft being dropped off at the apogee of 150 km, there is approximately a 10 kg increase in payload performance. If, for example, a 20 N thruster on a 225 kg spacecraft is assumed, the acceleration will be about 0.01 g or 0.1 m/s^2. Raising perigee from 50 km to 150 km requires increasing the semi-major axis by 50 km, which requires a delta-V of approximately 30 m/s. This would require 300 seconds of thruster burn time. Thus, left off at apogee, the spacecraft would begin gaining altitude after about 5 minutes or 20 degrees of arc, which is very adequate to avoid any risk of reentry or significant orbit decay due to drag. As stated previously, an advantage of this drop-off orbit is that the upper stage would then reenter immediately and would not require a reentry burn or maneuvers.

Summary

The principal system-level advantages of the ELVIS approach are:

1. ELVIS saves mass that can translate into increased payload capability for small launch vehicles. Mass savings could add between 40 and 90 kg or more to the payload capability of a small launch vehicle such as Sprite. 90 kg would be relatively unimportant for a "FatSat". However, 90 kg represents a very substantial payload mass gain and thus translates into major cost benefits for small payloads because for small spacecraft, the bus functions typically represent 75% of the spacecraft mass.

2. The MLM technique typically adds 50% to over 100% to the payload mass delivered to high-energy orbits. De-

pending on the detailed design, it may be possible to give Sprite or other small launch vehicles a modest payload capability to high-energy orbits that would represent a very dramatic cost reduction to the mission.

3. The design is exceptionally flexible, allowing one vehicle to meet multiple mission needs, which means more applications for a single vehicle. This flexibility allows greater economies of scale and reduces costs further. Additionally, each vehicle can be customized to meet specific mission needs because of the variety of payloads that will have been designed to be flown on the same Stagecraft.

4. The low thrust engine for the final stage/integral propulsion minimizes the engine mass that goes all the way to the mission orbit and also minimizes the mass of the final stage control actuators.

The bottom line is — for about $4 million, it would be possible to put a Stagecraft (launch vehicle and spacecraft bus, not including the payload) in a wide range of useful orbits, whenever needed (i.e., a responsive launch within as little as eight hours from the time the Stagecraft arrives at the launch site), and to maintain it in that orbit essentially indefinitely. All the user needs to add is the payload.

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