Low-Cost Planetary Missions Utilizing Small Launch Vehicles and a Novel Launch Mode[1]

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Abstract

The ability to substantially reduce the cost of planetary missions lies not only in the cost reductions to be found in spacecraft design but also in choice of launch vehicle. Smaller launch vehicles are cheaper but are also less capable at lofting sufficiently large masses to interplanetary velocities, while excessive miniaturization of spacecraft subsystems typically leads to higher cost. Microcosm has identified a new launch technique using traditional vehicles and technologies to circumvent these restrictions for many solar system exploration missions. For the purposes of this paper, we named it the Modified Launch Mode (MLM). This method can substantially increase the payload mass capabilities of all launch vehicles for high energy (high C_3) missions. It is particularly advantageous in missions that require onboard propulsion at destination and therefore can make repeated use of that system.

The technique has been described elsewhere and only a summary is presented here. This paper summarizes the newly available spectrum of low-cost planetary capability that this technique makes available to the small satellite community using smaller, cheaper vehicles to carry their spacecraft then have been commonly considered as necessary. A multi-faceted trade of launch vehicle capability against spacecraft size will be appropriate to minimize overall mission cost or to enhance mission capabilities within specified cost constraints. Emphasis is placed on the matching of minimum-cost launch vehicles to small spacecraft now being considered for deep-space exploration.

Summary of the Technique[2]

The application of the technique has been described at greater length in an associated paper[3] so only a short summary is presented here. The formulation of the technique arose through the need to get extra mass at very low cost to the Jupiter system in support of a small study. It was realized that a significant fraction of a launch vehicle's energy is spent accelerating the empty structure and associated subsystems of the final upper stage to the same high energy transfer orbit as the payload spacecraft.

A question was then postulated that if one were to use integral propulsion to supplant some of the energy supplied by the launch vehicle's upper stage to the upper stage/payload stack what, if any, would be the advantage in terms of net spacecraft mass to the transfer orbit? One would still use the launch vehicle's upper stage to exhaustion – accelerating a larger mass to a lower energy orbit, for example, $C_3 = 0$, or Earth escape velocity – then use integral propulsion to accelerate the spacecraft to its final transfer orbit.

[2] "Modified Launch Mode", patent pending.

[3] *A Low-Cost Modified Launch Mode for High C_3 Interplanetary Missions*, J.R. Wertz, H.F. Meissinger, S. Dawson, AAS/AIAA Astronautics Conference, Sun Valley, Idaho, 4–6 August 1997.

Through some relatively simple analysis of the question posed, a conclusion can be drawn that a substantial mass gain can be effected when using traditional launch vehicles, their nominal upper stages, and integral propulsion. These gains are present even when using rather conservative integral propulsion performance assumptions in the mass calculations. Meissinger, et al.[i], give some quantitative analyses of the various launch vehicles' performances. In much the same way that staging a rocket gives enhanced mass performance to Earth orbit, a similar 'staging of the interplanetary transfer orbit injection' enhances payload mass to final orbit.

We believe this technique to be new to the literature inasmuch as we have failed to find mention of similar techniques for enhancing planetary mission payload mass. The closest work the authors can find is work in support of the Pluto Express project where three consecutively smaller solid upper stages are stacked atop a Proton launch vehicle to provide additional mass gains over single large upper stage[ii]. This Pluto Express approach could be viewed as a 'quantized' application of the approach summarized here and detailed in reference [i]. This novel approach has now, apparently, been discarded because of the cost of the Proton launch and upper stage costs in favor of a triple Venus – Jupiter Gravity Assist (VVVJGA) trajectory to Pluto to be launched on a Delta or a Russian Molniya vehicle[iii]. The extra Venus orbits add approximately 3.5 years to the trajectory and greatly complicate mission operations. In addition, the extra cost from the engineering needed to cope with the larger thermal loads imposed upon the spacecraft in the vicinity of Venus' orbit must be included in the trade space of mission complexity and cost. These changes would seem to largely move the savings accrued through changing launch vehicles to the operational stage of the mission.

The proposed staging sequence that depends on using integral onboard propulsion in planetary missions has elements in common with the launch procedure for some Earth-orbital missions with high launch-velocity requirements. For example, the AXAF (Advanced X-Ray Astrophysical Facility) satellite to be launched to a 10,000 by 140,000 km Earth orbit, in late 1998, will reach this orbit through a sequence of onboard propulsion burns at apogee and perigee[iv]. Given the very large (4800 kg) AXAF net spacecraft mass, the use of integral propulsion to reach the intended final orbit is essential, given the payload capability limits of the Shuttle/IUS launch vehicle.

Negative C_3, Attitude and Telemetry Acquisition

It has been noted[4] that the need to thrust immediately following launch vehicle separation could place a heavy burden upon the spacecraft in that the normal mode of initial attitude determination and

Extra Δv to escape m/s	Apogee Radius 1000's km	Period days	C_3 km^2/s^2
200	175.98	3.178	–4.36
300	115.69	1.742	–6.51
500	67.46	0.821	–10.75

Table 1 Characteristics of a 'negative C_3' or phasing orbit

[4] Thanks to L. D'Amario of the Jet Propulsion Laboratory for the question.

the set-up of ground-to-spacecraft telemetry link will be forced into too small a period of time. This shortening of available time, it was argued, could invalidate the scheme as unworkable or, alternatively, eliminate any mass gains due to the higher altitude attained before the integral propulsion system's ignition.

If these concerns are real then a neat solution to the problem is offered by restricting the launch-vehicle provided C_3 to a negative value, that is the spacecraft has not attained escape velocity and is instead in a highly eccentric orbit around the Earth. A similar, but more lengthy, scheme was employed during the Clementine mission. This subsequent 'phasing' orbit could be used most effectively as a time period to allow attitude determination and control, as well as commencement of spacecraft telemetry sessions for full system and subsystem check out, prior to the interplanetary injection burn. Another point to be borne in mind when considering the value of a phasing orbit is that the injection burn can now be initiated ahead of perigee, thus splitting the burn about the phasing orbit's perigee. This has the advantage of lessening the impact of Δv losses due to the increasing altitude at which the spacecraft is thrusting. [See Table 1.]

Description of the Technique

Table 2 lists the payload mass at $C_3 = 0$ and at $C_3 = 77.3$ km^2/sec^2 for the Jupiter mission[v] resulting from the conventional and the modified launch modes (MLM) as shown in the two launch vehicle performance curves (Figure 1 & Figure 2). Also listed in the table is the payload mass remaining after insertion into a Jupiter polar orbit with the dimensions of 1.1 by 60 Jupiter radii which requires an insertion burn of 0.783 km/sec. The major payload mass gain due to the different launch mode amounts to 182% for the Taurus, 52% for Delta II, and 34% for Atlas II/Star 48 B launch vehicles. Clearly, the gain depends strongly on the mass and specific impulse of the LV upper stage and on the specific impulse (I_{sp} = 300 sec) that is assumed for the onboard engines used in the modified launch mode.

These results show that a minimum- or a medium-size Jupiter orbiter can be launched by the Taurus XL/S or the Delta II 7925, respectively, by using the MLM launch mode. Alternatively, the Delta LV would be able to launch two separate orbiters of nearly 180 kg each. This would allow simultaneous observations of physical phenomena from different Jupiter orbit locations, thereby greatly enhancing the scientific mission yield. Also, a trade between onboard propellant mass and payload mass at destination is made possible by separating the spacecraft from the upper stage at an intermediate C_3 value, e.g., at $C_3 = 20$ km^2/sec^2, thus reducing the required onboard propellant mass. The final spacecraft mass in this case would be about 385 kg, i.e., 25 percent more than in the conventional launch mode.

In the Jupiter orbiter mission the

Table 2 Payload Performance Improvement In Jupiter Mission by the Modified Launch Mode

Launch Vehicle	Payload at Jupiter Arrival		Payload in Jupiter Orbit	
	Conventional	Modified	Conventional	Modified
Taurus XL/S	60 kg	169 kg	46 kg	130 kg
Delta II 7925	308 kg	468 kg	237 kg	360 kg
Atlas II/Star 48B	530 kg	712 kg	408 kg	548 kg

Propellant loads	Propellant Tank Mass	Engine Mass
< 100 kg	8% of propellant load	2 kg
< 1000 kg	6% of propellant load	5 kg
> 1000 kg	4% of propellant load	10 kg

Table 3 Rule of Thumb for Tankage Mass Sizing

spacecraft's onboard propulsion system to be used at Earth departure will be used a second time for the orbit insertion maneuver. The multiple use of onboard propulsion in this example makes the modified launch mode particularly mass- and cost-effective.

The above data do not reflect the extra mass required for the onboard propulsion subsystem. A suggested rule of thumb for assessing the extra mass to be allocated for 'tankage' over that normally to be found onboard an interplanetary flyby spacecraft has been described[5] as in Table 3.

The use of the above rule of thumb in subtracting from the injected mass is intended to illustrate the differences on a level playing field between a traditional launcher's injected mass and the Modified Launch Mode's 'useful' injected mass. If, however, a mission were to take further advantage of the integral propulsion, such as in a planetary orbital insertion maneuver during an orbiter mission, then the subtraction of this rule-of-thumb mass from a MLM insertion can be clawed back into trade space of MLM versus traditional launch modes without hesitation. Indeed, it would seem for smaller missions, where the propellant tank sizing is most expensive in terms of loss of useful mass, then MLM becomes more practical in situations where the propulsion system is to be used for at least a second time.

[5] As suggested by E. Keith, Senior Propulsion Engineer, Microcosm, Inc.

Generic Payload Gain Characteristics of the Modified Launch Mode

To show the payload gain obtainable by the modified launch mode, compared with the conventional launch mode, the ratio of the final payload mass for each mode is derived as a function of launch energy. In the conventional launch mode the mass ratio between the initial and final mass is expressed by:

$$\frac{m_{C,0} + m_S}{m_{C,1} + m_S} = e^{\frac{\Delta v_1}{g I_{sp}}} = r_1,$$

where:

$m_{c,o}$ = mass of the conventional launch vehicle and spacecraft payload, but without the dry mass of the upper stage,

$m_{c,1}$ = mass of the conventional spacecraft payload on reaching its destination,

m_s = dry mass of the launch vehicle upper stage,

g = gravitational acceleration constant,

I_{sp} = specific impulse of the launch vehicle

Δv_1 = velocity impulse to reach the energy C_3 of the desired transfer trajectory, and

r_1 = initial to final mass ratio.

By eliminating the launch vehicle upper stage dry mass m_s in the modified launch

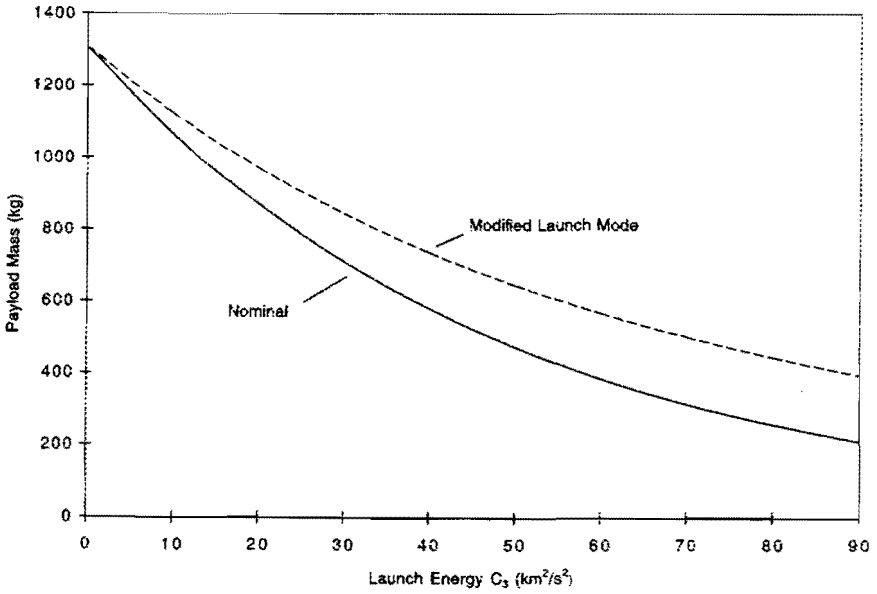

Figure 1 Delta II 7925 Escape Mission Performance

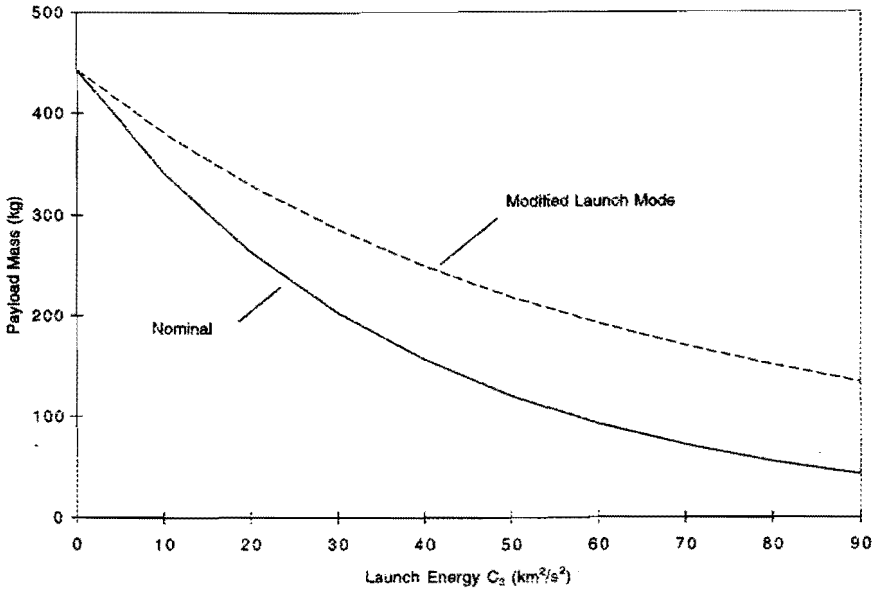

Figure 2 Taurus XL/S & Star 37 FM Escape Mission Performance

mode, the ratio of initial to final mass becomes:

$$\frac{m_{M,O}}{m_{M,1}} = r_1,$$

where:

$m_{M,O}$ = initial mass of the spacecraft in the modified launch mode, and

$m_{M,1}$ = final mass of the modified spacecraft in this mode.

The above charts show the advantages of the MLM method for a range of

traditional launch vehicles. All the charts are 'referenced' to a baseline 'drop-off' C_3 of 0 km^2/s^2, that is the MLM method takes over only when the launch vehicle has already burned to exhaustion at exactly Earth escape velocity.

As a simplifying assumption, the same specific impulse (I_{sp} = 300 sec) is used in both launch modes although the upper LV-stage typically has a somewhat larger I_{sp} value. Also, the masses $m_{M,0}$ and $m_{C,0}$ at the reference point $C_3 = 0$ are assumed to be equal. This leads to an explicit expression for the relative payload mass gain (R) which is derived from the two preceding equations,

$$R = \frac{m_{M,1}}{m_{C,1}} = \frac{1}{1 - \frac{m_s}{m_{C,O}}(r_1 - 1)}.$$

Figure 3 obtained for the Taurus XL/S launch vehicle with $m_{C,0}$ = 475 kg shows representative curves of the payload gain versus C_3 for several assumed upper stage dry-mass values, ranging from 100 to 400 kg. The velocity impulse Δv_1 that is used in defining the r_1-ratio is related to v_∞ and hence to C_3 by the equation

$$V_1 = V_{es,o}(\sqrt{1 + C_3 / V_{esc,o}^{\ 2}} - 1$$

where $v_{esc,o}$ is the escape velocity at the initial altitude. This formula is used in deriving the payload gain curves shown in Figure 3. Each of these curves has a pole at the r_1-value given by

$$r_1 = 1 + m_{c,o} / m_s.$$

where the denominator in the equation for R goes to zero.

As a specific example, the Taurus XL/S launch vehicle with a representative upper stage dry mass m_s = 164 kg, an initial mass $m_{C,0}$ = 475 kg, and a mass ratio r_1 = 2.863 corresponding to C_3 = 77.3 km^2/sec^2 for the assumed Jupiter transfer trajectory, the MLM payload mass gain will be 2.816, i.e., an increase of 182 percent in payload mass over the $m_{C,1}$ value of 60 kg, as previously shown in Table 2.

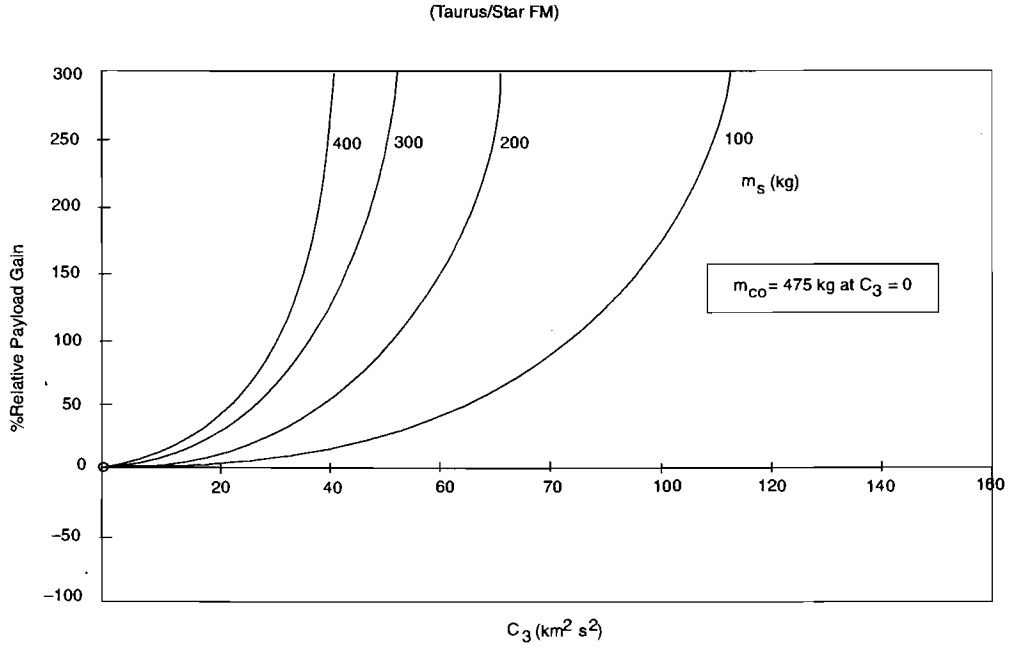

Figure 3 Payload Gain vs. C_3 by Modified Launch Mode

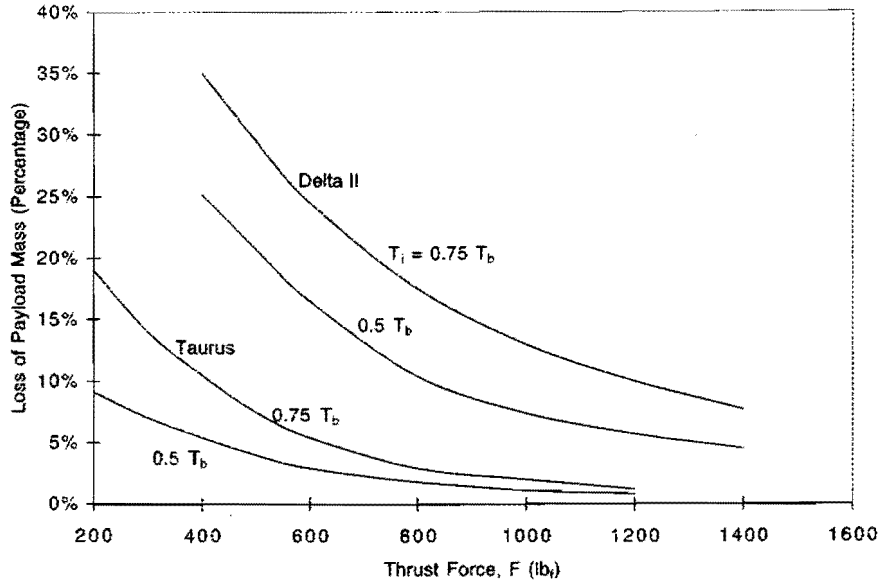

Figure 4 Loss of Payload Mass vs. Thrust Force for Taurus and Delta II Launch Cases

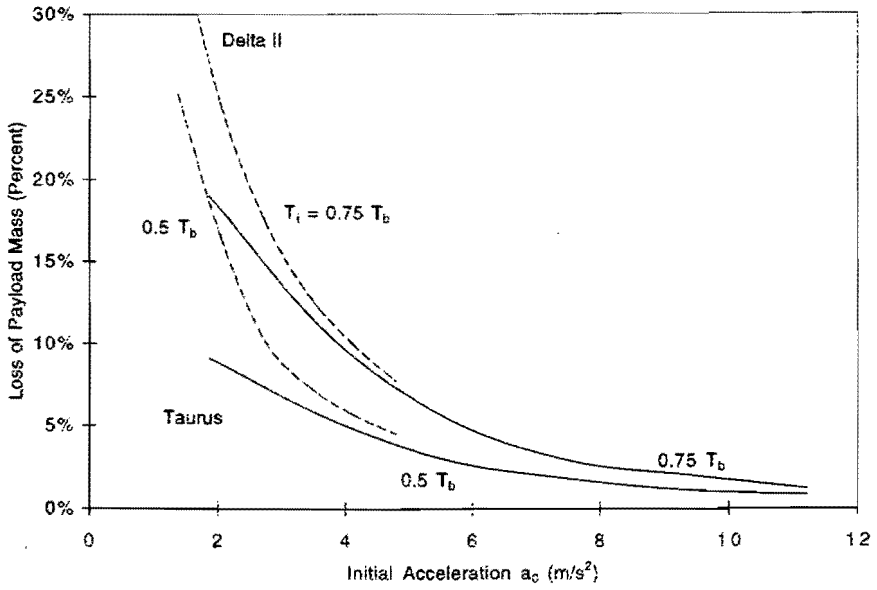

Figure 5 Loss of Payload Mass vs. Initial Acceleration for Taurus and Delta II Launch Cases

Analysis of the Delta-V Penalty Associated with the Modified Launch Mode

The net payload mass gain achievable by the modified launch mode will be adversely affected by the increase in the required departure velocity. This increased velocity is due to the higher burnout altitude resulting from the increased thrust phase duration. In the conventional launch mode the much higher thrust level of the launch vehicle's upper stage assures a very short thrust phase, and hence, minimizes the delta-V penalty. To keep the MLM mode performance penalty small, onboard engines with a sufficiently large thrust level must be used.

A simplified analysis is used to estimate the delta-V penalty, assuming that an *equivalent impulsive* thrust is applied at some point on the escape parabola before the end of the total thrust phase. The propellant mass, and hence, the burn time is derived from the nominal velocity impulse that is produced by the onboard propulsion system. To obtain upper and lower brackets of the prolonged burn-time effect, the assumed equivalent impulsive thrust is applied at a time t_i = 0.75 or $0.5T_b$. Figure 6 illustrates the flight path geometry used in this approximation. The upper and lower limits of the delta-V penalty thus obtained are conservative estimates, with results that depend on the onboard thrust force and the acceleration level. For more precise results, actual ascent flight histories should be computed for several

Table 4 Payload Mass and Mass Penalty Varying With Assumed Impulse Application Time, T_i

T_i (min)	m_f/m_o	Taurus			Delta II		
		m_f (kg)	Δm_f (kg)	Loss (%)	m_f (kg)	Δm_f (kg)	Loss (%)
0*	0.3560	169	0	0	468	0	0
4	0.3393	161.2	7.8	4.6	441.1	26.9	5.8
8	0.3053	145.0	24	14.2	396.8	71.2	15.2
12	0.2747	130.5	38.5	22.8	357.1	110.9	23.7
16	0.2525	119.9	49.1	29.1	328.3	139.7	29.9
20	0.2353	111.8	57.2	33.5	305.9	162.1	34.6

* Idealized condition (Impulsive thrust case)

Table 5 V_{esc1}, ΔV_i and Payload Mass Ratio Varying with Assumed Impulse Application Time, T_i

Impulsive Thrust Time, T_i (min.)	Radial Distance r_1/r_p	$Vesc_1$ (km/sec)	ΔV_i (km/sec)	Payload Mass Ratio, m_e/m_o
0*	1.0	11.052	3.073	0.3560
4	1.15	10.306	3.243	0.3393
8	1.48	9.085	3.560	0.3053
12	1.89	8.039	3.876	0.2747
16	2.29	7.303	4.129	0.2525
20	2.69	6.739	4.341	0.2353

* Idealized condition (Impulsive thrust case)

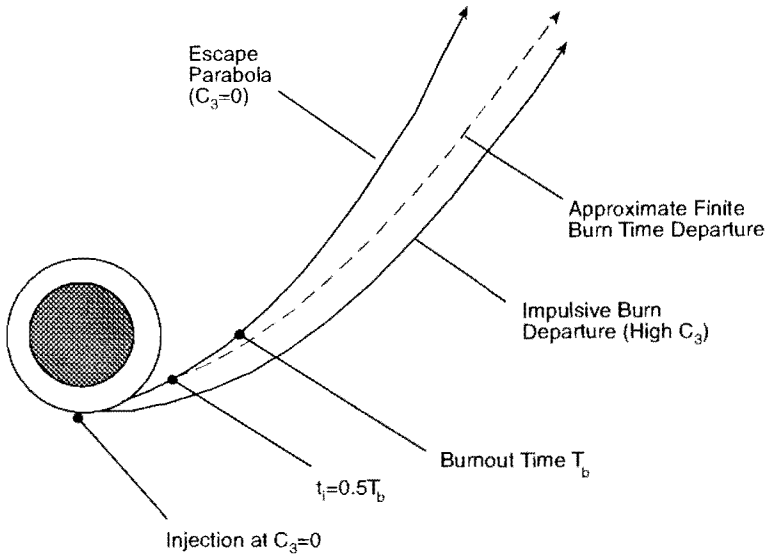

Figure 6 Geometry of Finite Burn-Time Departure Approximation Description of the Trade Space

different initial thrust acceleration levels.

Table 5 gives the values of the radial distance, the escape velocity at that distance, the required velocity increment ΔV_I, and the payload mass ratio $m_f/m_o = \exp(-\Delta V_I / g\ I_{sp})$ that correspond to the assumed impulsive thrust application times t_i. Table 4 shows the resulting payload mass decrease with change in the impulse-application time for the Taurus- and Delta II- launched spacecraft in the Earth-to-Jupiter mission discussed earlier (see Table 2). The payload mass loss is reasonably small only for impulse application times t_i that are lower than 8 minutes.

From these data the variation of the payload loss as function of the onboard-propulsion thrust level is derived for the two spacecraft sizes being considered. The results are shown in Figure 4 for the assumed upper and lower equivalent impulse application times. For the Taurus launch the loss is between 3 and 5 percent at 600 lb_f thrust, for Delta II it is between about 4 and 8 percent for 1400 lb_f thrust. The losses would be unacceptably high for thrust levels below 300 and 1200 lb_f, respectively, in the two launch vehicle examples. Corresponding results are shown in Figure 5, with payload loss as function of the initial acceleration level, a_o. The losses become acceptably small for initial accelerations above about 0.5 g. Thus, a sufficiently large onboard thrust level will reduce these losses to a small fraction of the payload gain achieved.

Description of the Trade Space

The trade space that now opens to designers of interplanetary missions is multi-faceted in its possible implementations.

1. The designer can use the MLM method to increase mass over the nominal mission on the chosen launch vehicle. Alternatively, the mission designer can choose to not use the extra mass but instead fly on a smaller, nominally less powerful vehicle.
2. The designer can use the same launch vehicle as would be chosen traditionally and use the mass margin afforded by MLM to trade on mass optimization versus cost of mass optimization.
3. As an adjunct to Trades 1 and 2, the designer can use some of the mass margin afforded by the MLM method to trade against C_3, thus widening the mission's launch window.

Examining Trade 1 in greater detail, one realizes that this trade is an 'either-or' kind of trade. Either the designer builds a larger, heavier spacecraft, adding instrumentation and capability as desired, and sticks with the nominal launch vehicle, or the designer chooses a smaller, but cheaper, launch vehicle and stays with the nominal capability of the mission. This trade has possibly the greatest allure for mission designers giving maximum leeway in design parameters such as capability versus cost.

Trade 2 is a more subtle overlying trade-space on that of Trade 1. The designer choosing to fly on the nominal launch vehicle and use the extra mass margin afforded by the MLM method can then trade among mass-optimized equipment traditionally utilized on interplanetary spacecraft, and heavier, more standard equipment to be found, say, on LEO spacecraft. Traditionally, interplanetary missions have spent large sums of money on mass optimization of subsystems and instruments. The MLM method can allow designers to trade gained mass for cheaper, less involved engineering using heavier equipment. Additionally, the designer could use some of the mass gained as radiation shielding allowing less tolerant, but more modern and hence faster, electronics to be used.

Trade 3 is an additional overlying trade to be added to the mix generated by Trades 1 and 2. By increasing the mission's launch window increasingly relaxed launch activities could be used so as to allow repeated launch attempts within the same campaign and hence achieve greater mission cost savings in terms of less burdensome on-pad operations.

Example Missions

The earlier introductory section alluded to results for a Jupiter orbiter mission and the gains therein that the MLM method could make available to the end-user. The associated paper [i] covers the same mission and other similar missions such as a Europa orbiter. In addition, the authors present an alternative scenario for a 100 kg Taurus or 250 kg Delta 7925 mission to Pluto.

Fast Mars Transfer

As a simple example of the MLM versatility a hypothetical mission to a C_3 of 30 km^2/s^2 is outlined. The chosen launch vehicle is a Taurus XL/S. Assuming the mission is a fast Mars intercept and Mars is nominally positioned at its average solar distance from the Sun then such a mission would have the following characteristics.

C_3	= 30 km^2/s^2
Time to Mars' average orbital distance from the Sun	= 120.30 days
(assumes no plane change)	= 0.33 years
Baseline Taurus XLS performance to C_3 = 30 km^2/s^2	= 202.8 kg

Table 6 Baseline Characteristics of a 'Fast-Mars' Transfer[vi]

The above mass number compares well with the associated mass numbers calculated for a nominal high C_3 MLM mission. The assumptions used in the MLM model were as above but include the additional mass penalties due to the non-standard use of tankage and heavier rocket engines as discussed in Table 3 as well as an assumed onboard Isp of 300 seconds. The following Figure 7 shows results available from a Taurus XL/S launch vehicle injecting the Mars-bound spacecraft to a range of C_3 values.

As can be seen in the above figure the capabilities of the Taurus have been stretched significantly over that of the nominal mission. This is despite the fact that a substantially lower C_3 has been employed over that C_3 for which the MLM was originally conceived and is indicative of the regime for which MLM should be used. At lower and lower C_3s the relative advantage of the MLM method is largely swamped by the higher efficiencies of traditional launch techniques as empirically noted in the 'rule-of-thumb' given in Table 3. Only by going to higher C_3's does the disadvantage of launching large, heavy, burnt-out upper stages become evident.

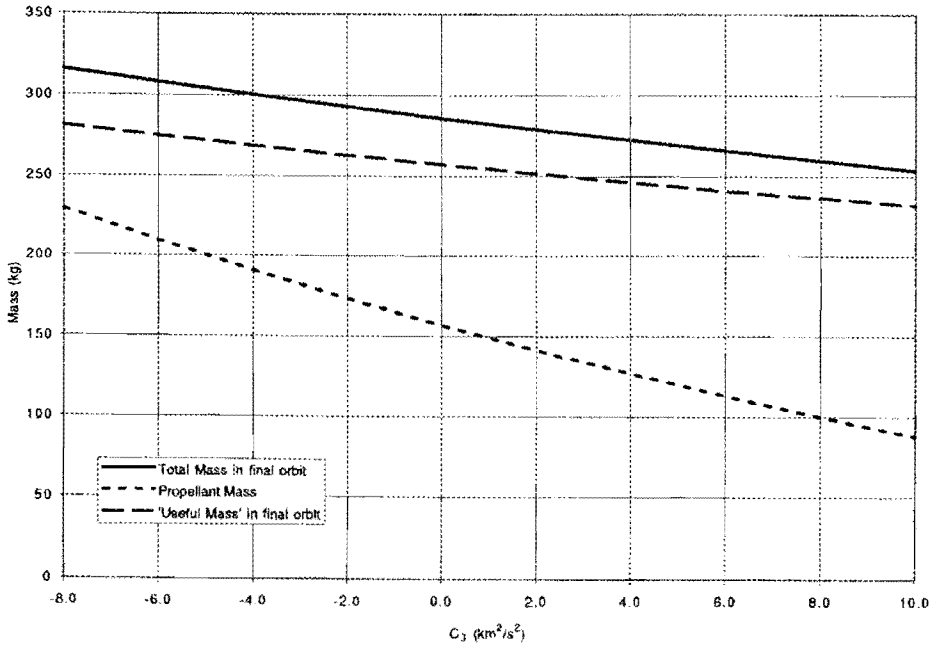

Figure 7* Taurus XL/S Capability to a 'Fast-Mars' Transfer using MLM for a range of launch vehicle-provided C_3 values.

Solar Probe to 2.5 solar radii via Jupiter GA

The use of MLM really comes into its own when considering small, cheap missions to the outer reaches of the solar system. This is especially true if the onboard propulsion system were to be used again, perhaps, in a Jupiter powered gravity assist to one of the outer planets.

For the purposes of this paper a solar probe to 4 solar radii that takes advantage of the MLM methodology was considered. The use of the onboard propulsion system at Jupiter has not been considered here but there would clearly be a trade set up among many factors. Some of those factors may be 1) the difficulty of keeping larger than normal quantities of propellant onboard all the way out to a Jupiter encounter, 2) the extra Δv that could be saved at Earth by performing a burn at Jupiter, 3) the complexity of the maneuver at Jupiter, and 4) the possible radiation shielding advantages from using onboard burns at Jupiter to effect the same 'turn angle' but at a greater jovicentric distance.

A sample trajectory for the mission is as shown in Table 7[vii viii]. This gravity assist maneuver reduces the heliocentric velocity to 2.35 km/s, that is on a highly eccentric (e = 0.9953) orbit about the Sun. Perihelion velocity reaches the staggering value of 390.1 km/s and perihelion distance is only 2.5 solar radii.

Leave Earth to Jupiter	5/29/98	@ a C_3 value of 116.094 km^2/s^2
Arrival at Jupiter	9/4/99	
Jupiter gravity assist turns spacecraft through	130.079°	
Perijove	13.8 Rj	

Table 7 Jupiter Transfer Details

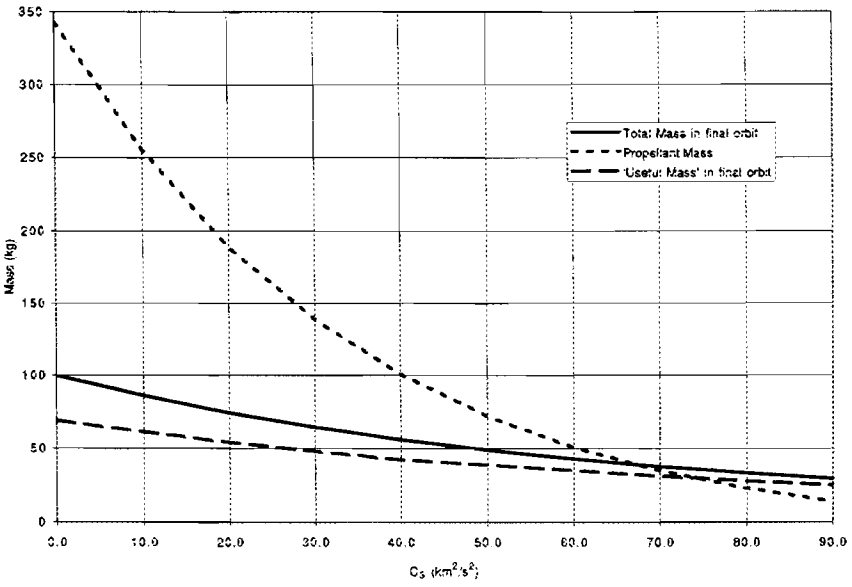

Figure 8* Taurus XLS Mass Performance Using MLM to a 'Solar Probe' Trajectory for a range of Launch Vehicle-provided C_3 Values.

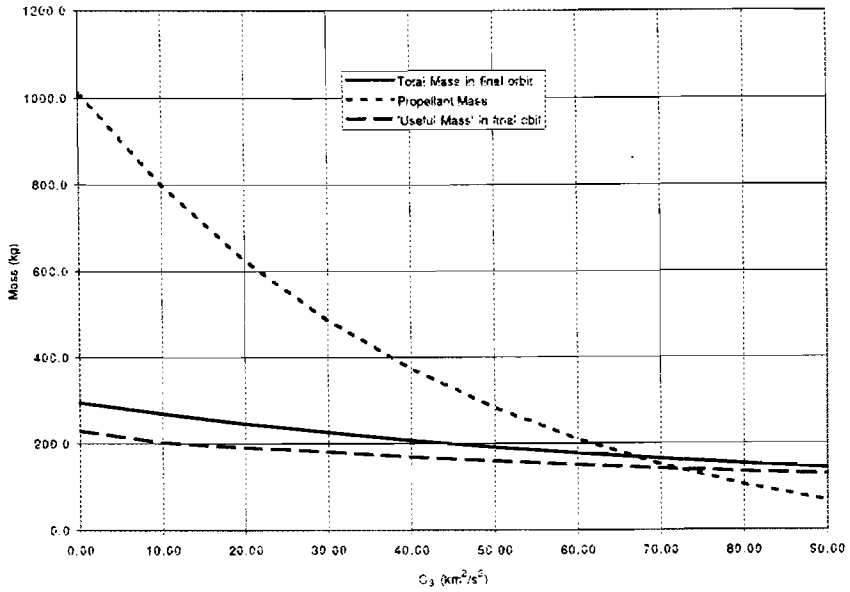

Figure 9* Delta 7925 Mass Performance Using MLM to a 'Solar Probe' Trajectory for a range of Launch Vehicle-provided C_3 Values

Applying the Modified Launch Mode to the desired trajectory produces the following results as shown in Figure 8 and Figure 9. As can be seen the possibilities of using the Taurus XLS for a viable mission to such an extreme C_3 value would seem to stretch the capabilities of spacecraft manufacturers to produce a 50 - 75 kg interplanetary craft capable of withstanding the immense thermal loads placed upon it at such close solar quarters. The Delta 7925, however, would seem to represent a feasible alternative – the 7925 model has the capability of launching approximately 120 kg to this high energy orbit. This value is produced by extrapolating the published data to this extraordinarily high C_3 value and so should be treated with appropriate caution. The 120 kg capability of a 'direct throw' to Jupiter compares with the MLM values of ~ 200 kg for a launch vehicle-supplied C_3 of 10 km^2/s^2. Under MLM approximately 800 kg of the launch vehicle's capability is used to launch the Jupiter-bound spacecraft's propellant. This propellant load, in purely ratio terms, dwarfs conventional tankages and must be kept in mind before such schemes are recommended. In fact, some of the MLM gain might be offset in terms of an upgraded attitude control system that might become necessary to control such an unwieldy spacecraft.

Conclusions

The proposed modified launch mode can provide a very significant gain in payload mass compared with the traditional launch mode. The results show that the payload mass gain for a given launch vehicle can be 50 to 100 percent or more. Conversely, a smaller, lower-cost launch vehicle can often be used for a spacecraft of given mass to reach the desired destination. Generally, this permits a trade between launch vehicle size, payload mass, flight time and other mission characteristics, to achieve the highest cost-effectiveness. Launch vehicle cost is a major factor in this trade, with Taurus in the $ 20 million, and Delta II in the $50 million range compared with the larger, more costly launch vehicles traditionally required to perform high launch-energy planetary missions.

It must be borne in mind that some of these missions require major deep-space maneuvers, such as orbit insertion at the target planet. The onboard propulsion subsystem carried for the modified launch mode would then be used more than once, which makes this launch mode more attractive. The required enlargement of the propellant tank size in such missions only presents a minor loss in net payload-mass. Even if onboard propulsion is to be used only during the launch phase, the extra dry mass and added development cost appear affordable in view of the cost and performance benefits to be gained. The delta-V penalty inherent in the MLM mode has been analyzed and found to be manageable; it can be held within reasonable limits if the initial thrust acceleration is of the order of 0.5 g or greater. Clearly, further study is required in terms of integrating an ascent trajectory so as to fully validate the analytical assumptions regarding the Δv penalty associated with the longer than usual thrust time. However, there are reasons to believe the analysis performed to date is sound and conservative in nature.

A generic analysis of the performance benefit of the modified launch mode shows that the payload gain increases rapidly with the required launch energy C_3. Therefore, this mode is most suitable for high-C_3 missions, e.g., those to Jupiter and beyond. The benefits obtainable in the Jupiter-orbiter, Europa-orbiter, and fast Pluto flyby missions are good examples.

Further studies should be conducted to look at specific candidate missions, their launch vehicle and payload requirements, representative maneuver sequences, and design and cost implications of adding the onboard propulsion capability for implementing the modified launch mode, and to derive the full spectrum of advantages that can be realized.

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[i] H. .F. Meissinger, S. Dawson and J.R. Wertz, "A Low-Cost Modified Launch Mode for High C_3 Interplanetary Missions", AAS/AIAA Astronautics Conference, Sun Valley, Idaho, 4–6 August 1997.

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*In Figures 7, 8, & 9 the abcissa value of C_3 represents the S/C departure energy at time of separation from launch vehicle.