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A Low-Cost Modified Launch Mode for High-C_3 Interplanetary Missions[1]

Hans F. Meissinger , and Simon Dawson , James R. Wertz[+]
Microcosm, Inc.

A novel method of spacecraft injection to interplanetary trajectories is described that can increase the net spacecraft mass by 50 to 100 percent or more in high-C_3 missions. This launch mode employs an unconventional staging sequence, separating the spacecraft from its launch vehicle at a velocity equal or close to Earth escape, i.e., an energy near $C_3 = 0$, and then using onboard propulsion to reach the required final C_3-value. It is particularly suitable for missions that require subsequent deep-space maneuvers for which an onboard propulsion system is carried. The large payload mass gain shown for some representative mission examples is due to applying the required final velocity impulse only to the spacecraft, but not to the launch vehicle's generally large upper-stage dry mass. Delta-V penalties inherent in using a lower than the conventional upper-stage thrust acceleration can be minimized so as not to exceed a few percentage points.

1. INTRODUCTION

The current trend in planning and performing planetary exploration missions is toward major reduction in size and cost of the spacecraft itself, as well as the launch vehicles required for accomplishing the mission. With the exception of missions to Mars, Venus or near-Earth asteroids which require comparatively low launch energies C_3 in the 10 to 20 km^2/sec^2 range, most other planetary missions, e.g., those to Jupiter, Saturn and beyond, as well as to Mercury or a close solar approach demand very much higher launch energies. Even minimum energy missions to Jupiter have launch energy requirements between 75 and 85 km^2/sec^2. A fast, direct Pluto flyby mission may require C_3 levels between 150 and 250 km^2/sec^2, depending on the mission design and the desired flight time.

In the interest of reducing the launch energy and thus lowering the size and cost of the required launch vehicle, mission profiles can be used that include planetary swingbys, e.g., of Venus and Earth to increase the mission energy through the gravity assist obtained at these encounters. The first such mission flown was the Galileo Jupiter Orbiter, launched in late 1989, and successfully performing its complex, multiple-orbit exploration phase of the planet and its Galilean satellites[1]. To achieve this mission within the payload capability constraints of the launch vehicle, the Shuttle and its IUS upper stage, the gravity assists obtained by a Venus-Earth-Earth swingby sequence were essential. The initial launch energy was only 13 km^2/sec^2.

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Chief Engineer, Microcosm; Associate Fellow, AIAA
System Engineer, Microcosm; Member, AIAA
[+] President, Microcosm, Inc., Torrance, CA; Associate Fellow, AIAA

but the consecutive swingbys served to increase it to about 80 km^2/sec^2, by the time of the final Earth encounter, in early 1992, as needed for the Jupiter transfer. Galileo reached its target in late 1995, after a total elapsed time of about 6 years. In addition, the added thermal protection required by the closer solar proximity during the inbound mission phase leads to greater spacecraft design and test complexity, as well as greater mission risk and cost.

In the modified launch mode the launch vehicle and its upper stages are used only to the point of injecting the spacecraft into an Earth-escape trajectory, at an initial energy C_3 at or near zero. This is followed by using "integral" propulsion onboard the spacecraft to add the velocity increment ΔV (delta-V) that is required to reach its destination. In principle, a staging sequence is being followed that differs from the conventional launch mode by applying the final large velocity impulse only to the planetary spacecraft rather than to both the spacecraft and the launch vehicle upper stage. As a result, the spacecraft mass to be launched on a high-C_3 mission can be increased by as much as 50 to 100 percent or more.

As an alternative to increasing the net spacecraft mass in a given planetary or interplanetary mission, the proposed modified launch mode (MLM) also offers the possibility of using a launch vehicle of lesser size and cost for a given net spacecraft mass. Thus, new trade opportunities between spacecraft mass and launch vehicle size, as well as flight time, are becoming available. The principal objective is to accomplish the mission in less time, at a lower overall cost, and without the added complications of repeated inner-planet swingbys.

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[2]. 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.

In applying the modified launch mode two issues of concern must be considered: (a) the need to carry a reasonably large onboard propulsion system for the large initial delta-V requirement, adding extra mass to the spacecraft being launched; (b) an appreciable delta-V penalty associated with the greater thrust phase duration at departure that is due to the lower thrust level of the spacecraft's onboard propulsion system, and the resulting higher burnout altitude.

With regard to (a), note that onboard propulsion often must be carried, anyway, to perform major subsequent spacecraft maneuvers, such as planetary orbit insertion. In these cases there is only a small increment in spacecraft mass and development cost for the larger propellant tanks needed to accommodate the added MLM delta-V requirement. Regarding (b), the delta-V penalty inherent in the longer thrust duration can be limited to a net payload loss of a few percentage points that are minor compared with the achievable payload mass gain. Sufficiently large onboard engines must be used in order to obtain an initial acceleration level of about 0.5 g, depending on specific mission characteristics.

To gain greater assurance that all subsystems on the spacecraft are in proper working condition prior to initiating the onboard propulsion thrust phase, it may be desirable to allow

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Patent pending

extra time for system checkout and, if necessary, for corrective action by ground control after separation from the launch vehicle.[2] To provide this extra time in the launch sequence, the spacecraft separation can be performed to a point slightly before reaching escape velocity. This means the spacecraft will enter into a highly eccentric phasing orbit during which the desired checkout and onboard propulsion system activation can be performed. Several hours later, at a point close to perigee, the spacecraft will then initiate the onboard propulsion thrust phase, starting at a C_3 value slightly less than zero. This sequence has the added advantage of spreading the onboard thrust phase on both sides of perigee, therby minimizing the previously mentioned ΔV performance penalty.

2. PAYLOAD MASS GAIN ACHIEVABLE IN JUPITER MISSIONS

As an example of the payload mass gain that can be achieved by the modified launch mode we consider a direct transfer to Jupiter which under near-minimum energy conditions requires a launch energy C_3 between 75 and 85 km^2/sec^2. In this example a Hohmann transfer is assumed that requires a hyperbolic excess velocity V_∞ of 8.79 km/sec at launch, i.e., an energy $C_3 = V_\infty^2 = 77.3$ km^2/sec^2, based on a transfer between the mean orbit radius of 1.0 AU at Earth and 5.2 AU at Jupiter. Three small-to-moderate size launch vehicles (LVs) are considered for this mission, i.e., Taurus XL/S-Star 37 FM, Delta II 7925, and Atlas II, the latter with and without the Star 48B upper stage. Figures 1, 2 and 3 show their injected payload mass capability as function of C_3. The solid-line performance curves in each of these figures represent the conventional launch mode that uses the launch vehicles' upper stage(s) to achieve the required injection velocity, dashed-lines represent the payload increase by the modified launch mode in which the payload is separated from the LV's upper stage at $C_3 = 0$. For each launch vehicle the spacecraft injected mass is considerably greater than for the conventional launch mode, since the dry mass of the LV's upper stage does not have to be accelerated beyond Earth escape velocity.

Table 1 lists the payload mass at $C_3 = 0$ and at $C_3 = 77.3$ km^2/sec^2 for the Jupiter mission resulting from the conventional and the modified launch modes as shown in the three LV performance curves. 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 a retro-velocity 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 (Isp = 300 sec) that is assumed for the onboard engines used in the modified launch mode.

Table 1
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

[2] This was pointed out by L. D'Amario of the Jet Propulsion Laboratory.

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 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 weight- and cost-effective.

The above data do not reflect the extra mass required for the onboard propulsion subsystem. If onboard propulsion were to be used for the Earth departure maneuver only, e.g., in a Jupiter flyby mission, the allowance for the subsystem dry mass would have to be 50 to 75 kg or more. If Jupiter orbit insertion is required, in addition, then the extra mass allowance to accommodate the MLM mode would reflect only a greater propellant tank size, i.e., a dry mass increase of 10 to 15 kg.

Considering the option of including an eccentric phasing orbit prior to initiating the final departure thrust period, as discused in Section 1, the launch vehicle separation may be set at a point 300 m/s below escape velocity, which corresponds to C_3 = –6.5 km^2/sec^2. The resulting apogee radius is 115,690 km and the phasing orbit period is 1.74 days. The required departure velocity impulse increases from 3.1 to 3.4 km/sec, which is largely offset by the higher payload mass that is injected by the launch vehicle at the neagtive rather than zero C_3 value.

3.GENERIC PAYLOAD GAIN CHARACTERISTICS OF THE MODIFIED LAUNCH MODE

To show the payload gain obtainable by the modified, 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

$$\frac{m_{C,O} + m_S}{m_{C,1} + m_S} = e^{\frac{\Delta V_1}{g I_{sp}}} = r_1,$$

where:

$m_{c,o}$ = mass of the conventionally launched spacecraft at C_3 = 0

$m_{c,1}$ = 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 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.

As a simplifying assumption, the same specific impulse (I_{sp} = 300 sec) is used in both launch modes. 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,

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

Figure 4 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

$$\Delta V_1 = V_{esc,o}\left(\sqrt{1 + C_3 / V_{esc,o}{}^2} - 1\right)$$

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 4. 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 1.

4. MLM APPLICATION TO OTHER HIGH-C_3 MISSIONS

In this section the use of the modified launch mode for an orbiter mission around Jupiter's moon Europa, and a fast Pluto flyby mission are considered.

Europa Orbiter Mission

This mission is of high scientific interest, given the recent discovery of water ice on Europa's surface and the possibility of some form of life existing in the water below. A polar orbit around Europa is envisioned for detailed observations. Upon arrival at Jupiter, three successive major maneuvers (see Figure 5) are performed to achieve the Europa orbit in the most economical way, i.e., with a lower total delta-V than would be required for other insertion sequences. As shown in the figure, the maneuver sequence consists of:

1. Initial insertion into a highly eccentric polar orbit, with a perijove close to the surface and an apojove of up to 150 Jupiter radii. This minimizes the first delta-V maneuver.

2. At apojove a second maneuver will be performed to raise the perijove to Europa's orbital radius. This maneuver also is reasonably small.

3. The Europa orbit insertion maneuver is performed at the time of passing the new perijove. This is by far the largest maneuver of the mission. In Europa-centered coordinates, the difference between the approach velocity at perijove and Europa's orbital velocity becomes the hyperbolic approach velocity V_∞ of the encounter which determines the required orbit insertion retro-maneuver magnitude.

The velocities of this three-step maneuver sequence are given in Table 2 which also shows the ratio of mass in Europa orbit to mass at Jupiter arrival. A key parameter is the apsidal ratio Q = r_a/r_p of the initial, highly eccentric Jupiter orbit with apsidal ratios ranging from 60:1 to 150:1. As shown in the table, the total maneuver velocity ranges from 6.2 to 5.6 km/sec for the apsidal distance ratios assumed. An alternative insertion sequence starting with an initial orbit apojove at the distance of Europa (omitting step 2), would require a three to four times greater initial maneuver delta-V for step 1. Ratios of the final mass, m_f, in Europa orbit to the mass,m_o, at Jupiter arrival also are given in the table, ranging from 0.128 to 0.157 for the assumed values of the apsidal ratio, Q.

Table 2
MANEUVER VELOCITIES OF THREE-STEP EUROPA ORBIT INSERTION AND RESULTING MASS RATIOS

Apsidal Ratio Q	Initial Orbit Injection (km/sec) ΔV_1	Perijove Raising (km/sec) ΔV_2	Europa Orbit Injection (km/sec) ΔV_3	Total Maneuv. (km/sec) $\sum \Delta V$	Resulting Mass Ratio M_f/M_o	Interim Orbit Time (Days) T_i
60	0.783	1.865	3.528	6.176	0.1276	22.7
80	0.662	1.434	3.819	5.915	0.1392	33.9
100	0.584	1.166	4.006	5.756	0.1468	46.6
150	0.500	0.793	4.256	5.550	0.1573	83.4

The time interval, T_i, between the initial Jupiter orbit insertion and the arrival at Europa varies from 22.7 to 83.4 days for the assumed Q-ratios (see Table 2). The elapsed time will be a critical factor in the mission schedule to assure proper timing of spacecraft arrival for Europa-orbit insertion. It is constrained by the synodic period of the interim eccentric orbits and Europa's period of revolution, a period of 4.21 to 3.71 days for the assumed values of the apsidal ratio Q. Earth-launch and Jupiter-arrival dates can be selected readily to conform with this synodic period.

Based on the mass ratios given in Table 2, values for the final mass in Europa orbit are given in Table 3 for several launch vehicles (Delta II 7925, Delta II 7925 H, Delta III and Atlas II with the Star 48 upper stage), using the modified launch mode with $C_3 = 0$ as reference point. The m_o-values are 468 kg for Delta II 7925; 555 kg for Delta II 7925H (a modification that uses the solid strap-on rockets designed for Delta III); 980 kg for Delta III; and 712 kg for Atlas II/Star 48), based on the Jupiter orbiter mission data discussed in Section 2. The assumed launch date is in the years 2004/5, and a Type II transfer trajectory to Jupiter is considered with a transfer time of about 3 years. These launch conditions correspond to minimum values of Earth departure energy and Jupiter arrival velocity based on data from the Jupiter Mission Handbook by Sergeyevsky and Snyder[3]. By using the MLM in this mission concept an estimated 30 to 50 percent increase of the payload mass in Europa orbit is gained compared with the conventional launch mode.

Results given in Table 3 indicate that only the in-orbit mass obtainable by an Atlas II launch (105 to 112 kg), or a Delta III launch (144 to 154 kg), and for the largest assumed Q-ratios, appear suitable for this mission. The lower in-orbit masses achievable in the conventional launch mode tend to rule out the lower-performance Atlas II as launch vehicle.

Table 3
FINAL MASS IN EUROPA ORBIT FOR
SEVERAL LAUNCH VEHICLES (kg)

	Delta II	Delta II H	Atlas II w.Star 48	Delta III
$M_{C3=0}^{(1)}$	1300	1542	2000	2720
$M_0^{(2)}$	468	555	712	979
M_f Q=60	60	71	91	125
M_f Q=80	65	77	99	136
M_f Q=100	69	82	105	144
M_f Q=150	74	87	112	154

(1) Initial mass at C_3=0
(2) Mass at Jupiter arrival

The spacecraft mass in-orbit shown in this table - based on using the MLM - must include an allowance for the onboard propulsion dry mass and some extra propellant for orbit corrections. Also, an allowance for radiation shielding is required to protect against radiation belt exposure during the one-time, initial close Jupiter approach.

Alternative mission concepts currently being considered would eliminate the second of the three orbit insertion steps and thereby save maneuver propellant, but it would increase the mission complexity and add one or more years of extra time before Europa orbit insertion: The characteristics of the spacecraft's orbit about Jupiter would change gradually due to gravitational perturbations from repeated Jovian satellite encounters, known as "orbit pumping".

Fast Pluto Flyby

The modified launch mode can be applied to a fast Pluto flyby mission with Jupiter gravity-assist. A particular advantage is gained by using the "powered swingby" technique which uses only a small delta-V expenditure to raise the V -level at Jupiter departure to 8 or 10 km/sec. This minimizes the Jupiter-to-Pluto transfer time and thereby greatly reduces the overall flight time, while requiring only a moderate amount of launch energy C_3. Thus, a launch vehicle of the Taurus or Delta II 7925 class can be used to achieve the mission with a spacecraft final mass of about 100 or 250 kg, respectively.

Missions with a similarly short overall transfer time and comparable final spacecraft mass at Pluto, but not using the modified launch mode and Jupiter swingby, require a much larger, higher-cost launch vehicle, perhaps of the Proton class.

An alternative fast Pluto flyby mission concept[4] with a moderate launch energy depends on multiple Venus swingbys for gravity assist to raise the mission energy - akin to the Galileo Jupiter mission. As such, the mission profile becomes more complex and requires added thermal-control because of its initial inbound excursions to Venus, in contrast with the proposed direct, initial Earth-to-Jupiter transfer.

Table 4 gives generic characteristics of the Jupiter-Pluto transfer trajectory, listing the eccentricity, transfer angle, transfer time, and the required delta-V augmentation at the powered swingby for (jovicentric) hyperbolic departure velocities $V_{\infty j}$ ranging from 6 to 16 km/sec.

Using Jupiter and Pluto ephemeris data for the first two decades beyond the year 2000 and the data in Table 4, a 9-year transfer trajectory with $V_{\infty J}$ = 10 km/sec was selected with Jupiter departure in Oct. 2004 and Pluto arrival in Oct. 2013. The assumed Jupiter encounter date in 2004 matches a minimum-energy, Type I transfer from Earth launched in Oct. 2002, based on data from the Jupiter Mission Handbook[3]. The total flight time thus becomes 11 years, with an Earth departure energy C_3 of 90.5 $(km/sec)^2$. Table 5 gives additional, detailed mission characteristics. Note that the 136-degree swingby angle change of the jovicentric hyperbolic velocity vector in this mission profile matches the intended Jupiter departure in the direction that is collinear with its orbital velocity. This minimizes the required departure velocity for the intended 9-year transfer to Pluto. (However, the assumed closest Jupiter approach of 2 R_J that would meet this constraint may have to be increased to mitigate the radiation exposure during the flyby).

Table 4
CHARACTERISTICS OF JUPITER-TO-PLUTO TRANSFERS

$V_{\infty j}$ [1] (km/sec)	V_H [2] (km/sec)	$V_{\infty H}$ [3] (km/sec)	e	Transfer Time (yr)	Transfer Angle (deg)	ΔV_j [4] (km/sec)
6	19.05	4.733	1.1315	14.54	125.30	0
8	21.05	10.129	1.6026	10.897	111.09	0.395
10[5]	23.05	13.813	2.1206	8.985	103.43	0.903
12	25.05	16.941	2.6858	7.738	98.58	1.499
14	27.05	19.779	3.2978	6.849	95.24	2.200
16	29.05	22.437	3.9568	6.167	92.81	2.993

(1) Jovicentric hyperbolic velocity
(2) Heliocentric departure velocity
(3) Heliocentric hyperbolic velocity
(4) Powered swing by velocity increment
(5) Selected 9-year Jupiter-to-Pluto transfer, with $V_{\infty j}$ = 10 km/sec

In the proposed mission concept the launch window constraints dictated by the relative orbital positions of Earth, Jupiter and Pluto are of concern if a different launch opportunity is to be selected. The Jupiter-Pluto synodic period is about 12 years, and that for Earth-Jupiter is 13 months. Thus, one or two launch windows about one year apart, separated by 10 to 12-year intervals would be available, with departure conditions at Earth and encounter conditions at Jupiter being highly sensitive to the relative planetary positions in each case.

The possibility of using the low-cost Taurus launch vehicle for this demanding mission class, as well as the simpler mission profile that avoids multiple inner planet swingbys on the way to Jupiter make the proposed launch mode highly attractive for further consideration in planning the projected Pluto mission.

5. 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 diminished by the increase in the required departure velocity that 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.

Table 5
MISSION CHARACTERISTICS FOR SELECTED
9-YEAR JUPITER-TO-PLUTO TRANSFER
(11-YEAR FLIGHT TIME EARTH-TO-PLUTO) [1]

Jovicentric departure hyperbolic excess velocity (selected from Table 4)	$V_{\infty j}$	10.0 km/sec	
Transfer time to Pluto	t	8.99 years	
Transfer angle (Heliocentric true anomaly)	η	103.4 deg	
Matching Jupiter departure and Pluto arrival dates (from ephemeris data)		Oct. 2004 (departure) Oct. 2013 (arrival)	
Selected near-minimum energy transfer trajectory from Earth [1] (2-yr flight time to Jupiter.) (Type I transfer) Departure date		Oct. 2004	
· Launch energy (Earth departure)	C_3	90.5 km^2/sec^2	
· Arrival hyperbolic velocity at Jupiter	$V_{\infty A}$	7.2 km/sec	
· Increase by powered swingby	ΔV_∞	2.8 km/sec	
· Powered swingby velocity increment	ΔV_j	0.700 km/sec	
· Mass ratio M_f/M_o at Jupiter		$e^{-\Delta V/g\ Isp}$=0.797	
Initial and final spacecraft mass (arriving and leaving Jupiter)		M_0 (kg)	M_f (kg)
· Taurus-launch, M_E = 475 kg at C_3 = 0		140	112
· Delta II-launch, M_E = 1300 kg at C_3 = 0		390	310

(1) Based on data from Jupiter Mission Handbook, Reference 3

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 different initial thrust acceleration levels.

Table 6 gives the values of the radial distance, the escape velocity at that distance, the required velocity increment ΔV_1, and the payload mass ratio m_f/m_o = exp (- ΔV_1/ g I_{sp}) that correspond to the assumed impulsive thrust application times t_i. Table 7 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 in Section 2 (see Table 1). The payload mass loss is reasonably small only for impulse application times t_i that are lower than 8 minutes.

Table 6

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)

Table 7

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)

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 7 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 8, 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.

6. 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.

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.

A generic analysis of the performance benefit of the modified launch mode (Section 3) 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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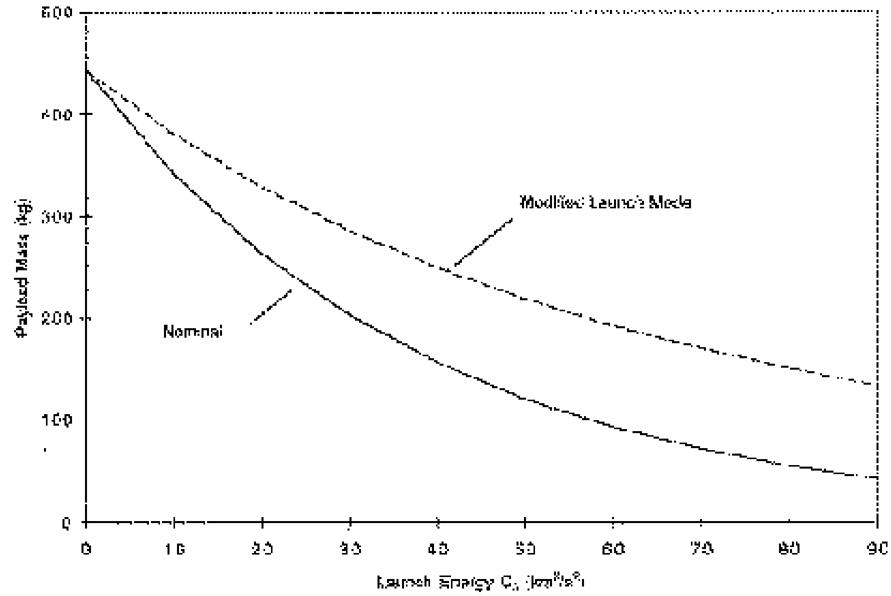

Fig. 1. Taurus XL/S & Star 37 FM Escape Mission Performance

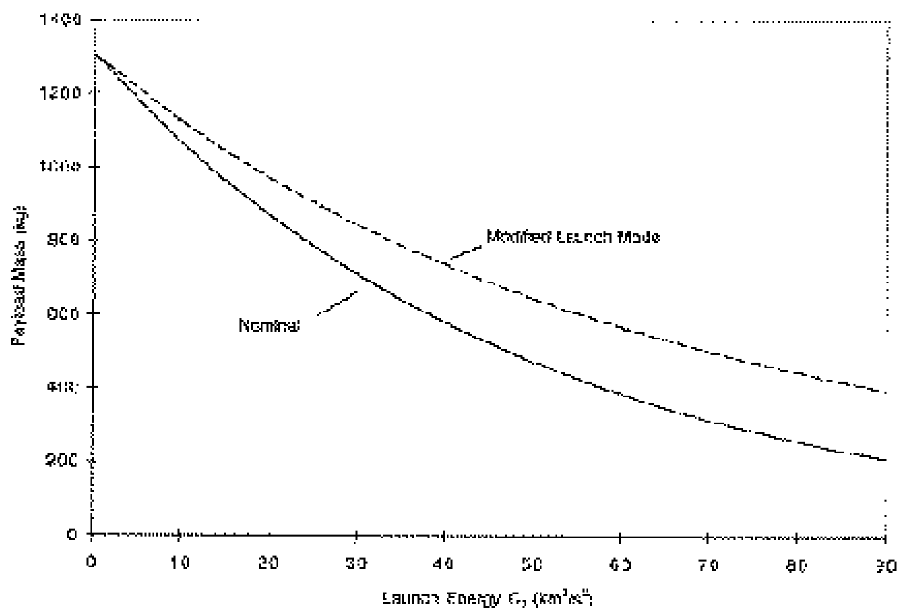

Fig. 2. Delta II 7925 Escape Mission Performance

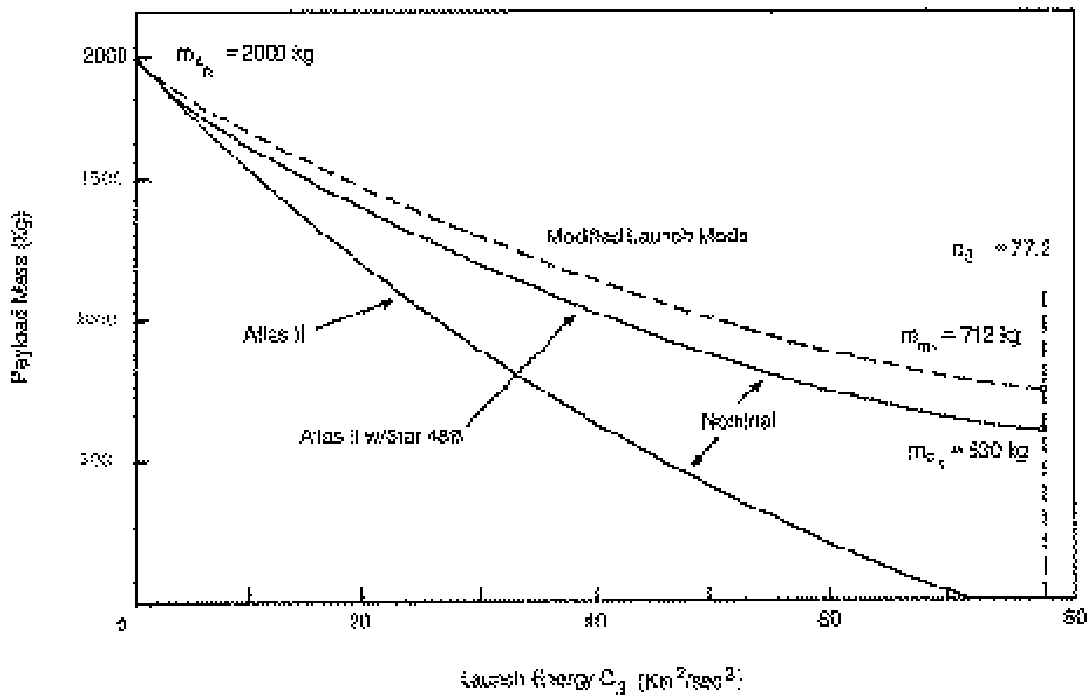

Fig. 3. Atlas II Escape Mission Performance

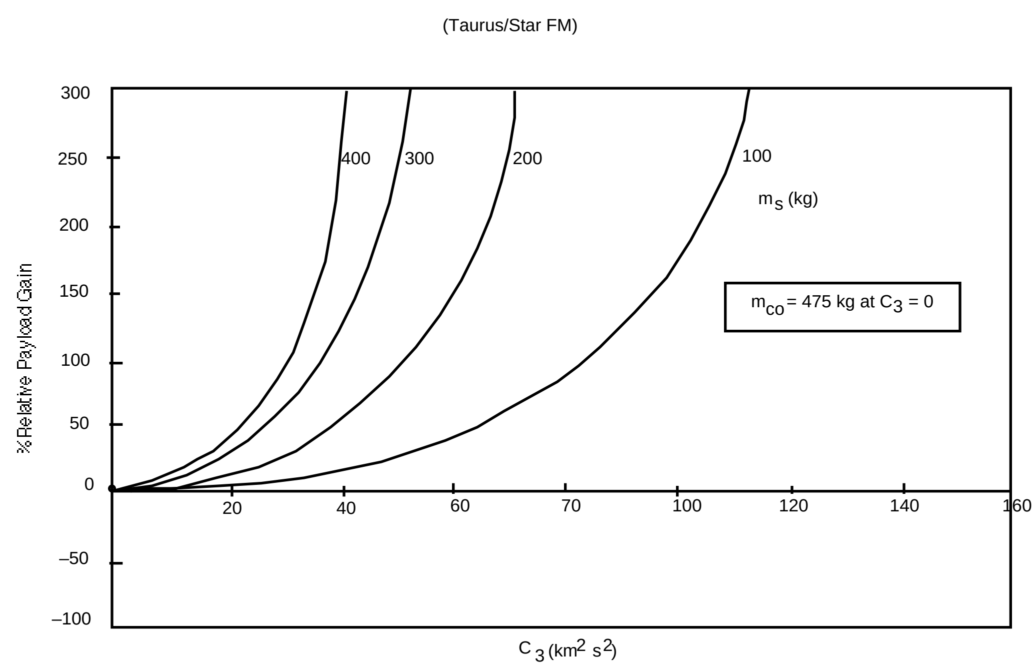

Fig. 4. Payload Gain vs. C_3 by Modified Launch Mode

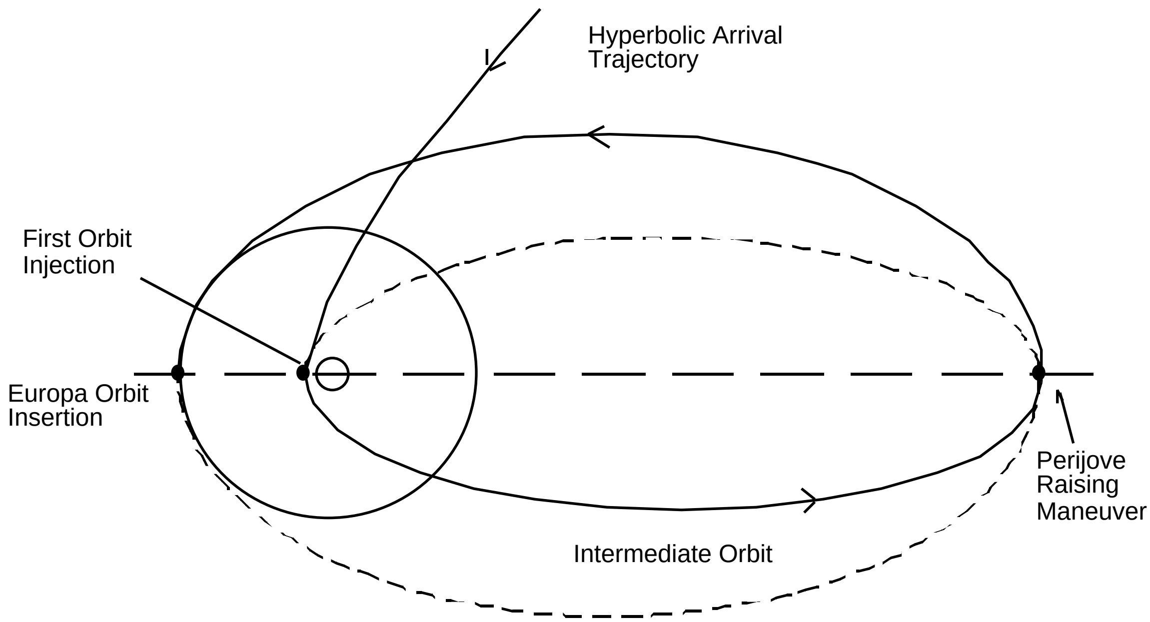

Fig. 5. Europa Orbit Insertion Sequence

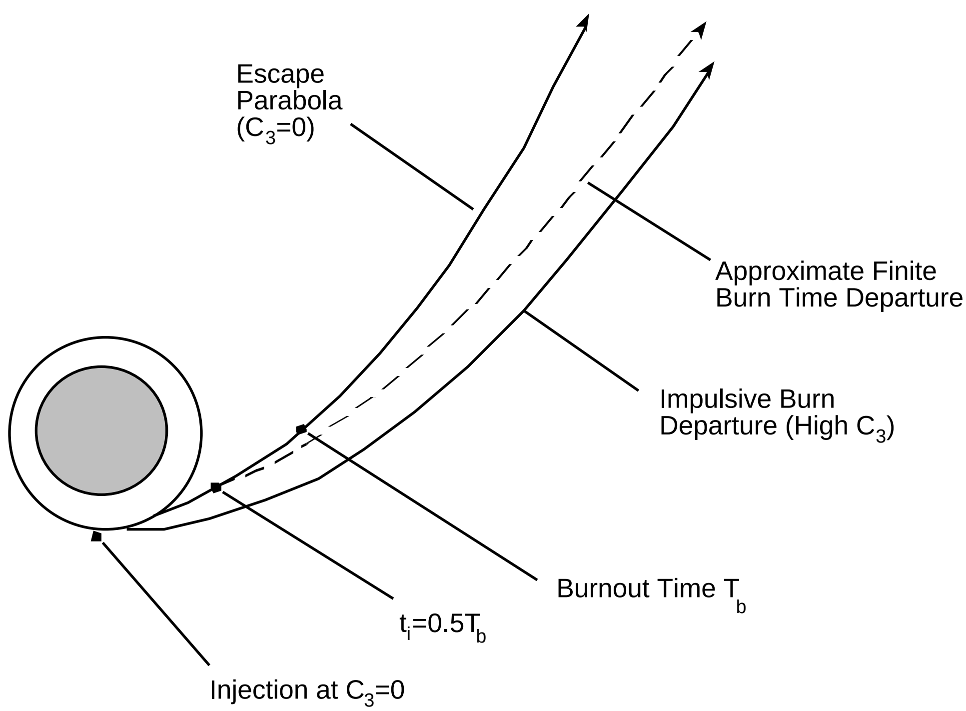

Fig. 6. Geometry of Finite Burn-Time Departure Approximation

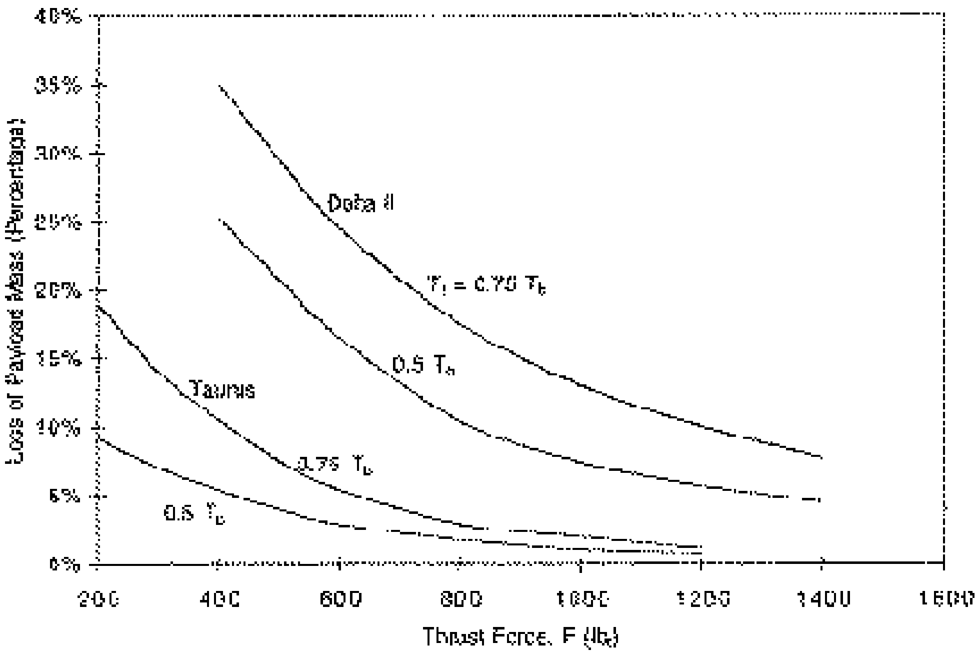

Fig. 7. Loss of Payload Mass vs. Thrust Force for Taurus and Delta II Launch Cases

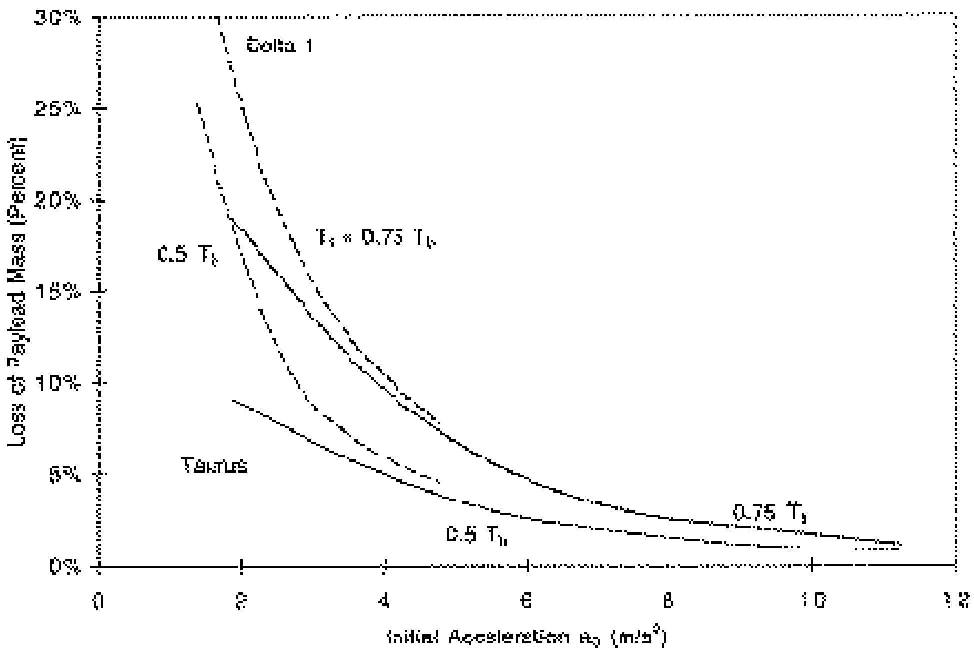

Fig. 8. Loss of Payload Mass vs. Initial Acceleration for Taurus and Delta II Launch Cases