Human Exploration Missions Study

Launch Window from Earth Orbit

MSFC/Alpha Technology, Inc.

FINAL REPORT

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Human Mars Mission

Launch Window from Earth Orbit

Introduction

The determination of orbital launch window characteristics is of major importance in the analysis of human interplanetary missions and systems. The orbital launch window characteristics are directly involved in the selection of mission trajectories, the development of orbit operational concepts, and the design of orbital launch systems.

The orbital launch window problem arises because of the dynamic nature of the relative geometry between outgoing (departure) asymptote of the hyperbolic escape trajectory and the earth parking orbit. The orientation of the escape hyperbola asymptotic relative to earth is a function of time. The required hyperbola energy level also varies with time. In addition, the inertial orientation of the parking orbit is a function of time because of the perturbations caused by the Earth's oblateness. Thus, a coplanar injection onto the escape hyperbola can be made only at a point in time when the outgoing escape asymptote is contained by the plane of parking orbit. Even though this condition may be planned as a nominal situation, it will not generally represent the more probable injection geometry. The general case of an escape injection maneuver performed at a time other than the coplanar time will involve both a path angle and plane change and, therefore, a ΔV penalty. Usually, because of the ΔV penalty the actual departure injection window is smaller in duration than that determined by energy requirement alone.

This report contains the formulation, characteristics, and test cases for five different launch window modes for Earth orbit.

These modes are:

(1) One impulsive maneuver from a Low Earth Orbit (LEO)

(2) Two impulsive maneuvers from LEO

(3) Three impulsive maneuvers from LEO

(4) One impulsive maneuvers from a Highly Elliptical Orbit (HEO)

(5) Two impulsive maneuvers from a Highly Elliptical Orbit (HEO)

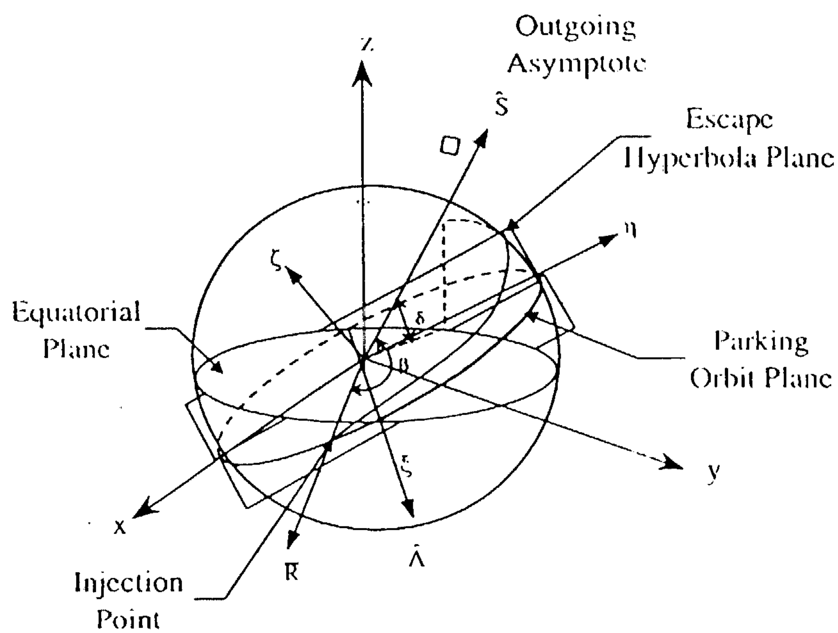
The formulation of these five different launch window modes provides a rapid means of generating realistic parametric data for space exploration studies. Also the formulation provides vector and geometrical data sufficient for use as a good starting point in detail trajectory analysis based on calculus of variations, steepest descent, or parameter optimization program techniques.

NON-COPLANAR ORBITAL LAUNCH GEOMETRY

The basic geometry of the completely general, non-coplanar orbital launch geometry is illustrated in this chart. The parking orbit plane is defined by orbit inclination and orbit ascending node. The escape hyperbolic conditions are defined by the outgoing asymptote vector ($\bar{S}$) right ascension (α) declination (δ) and C_3 (twice the total hyperbolic escape energy per unit mass). The parking orbit ascending node has a regression rate of -5.0 to -7.2 degrees per day for Low Earth Orbit (LEO) The angular orientation (α and δ) of the outgoing asymptote changes much slower, less than 1 degree per day. Because of the LEO regression rate, co-planar launch geometry exists for only a short period of time

High Earth Orbit (HEO) (orbit period ~24 hour) ascending node regression rate is much smaller, ~ -0.10 degree per day. The required plane change between the HEO and the outgoing asymptote does not build up as fast as in the LEO case. The HEO conditions would exist for the SEP architecture And the LEO conditions would exist for the nuclear thermal and chemical high thrust architectures.

Figure 1. Non-Coplanar Orbital Launch Geometry



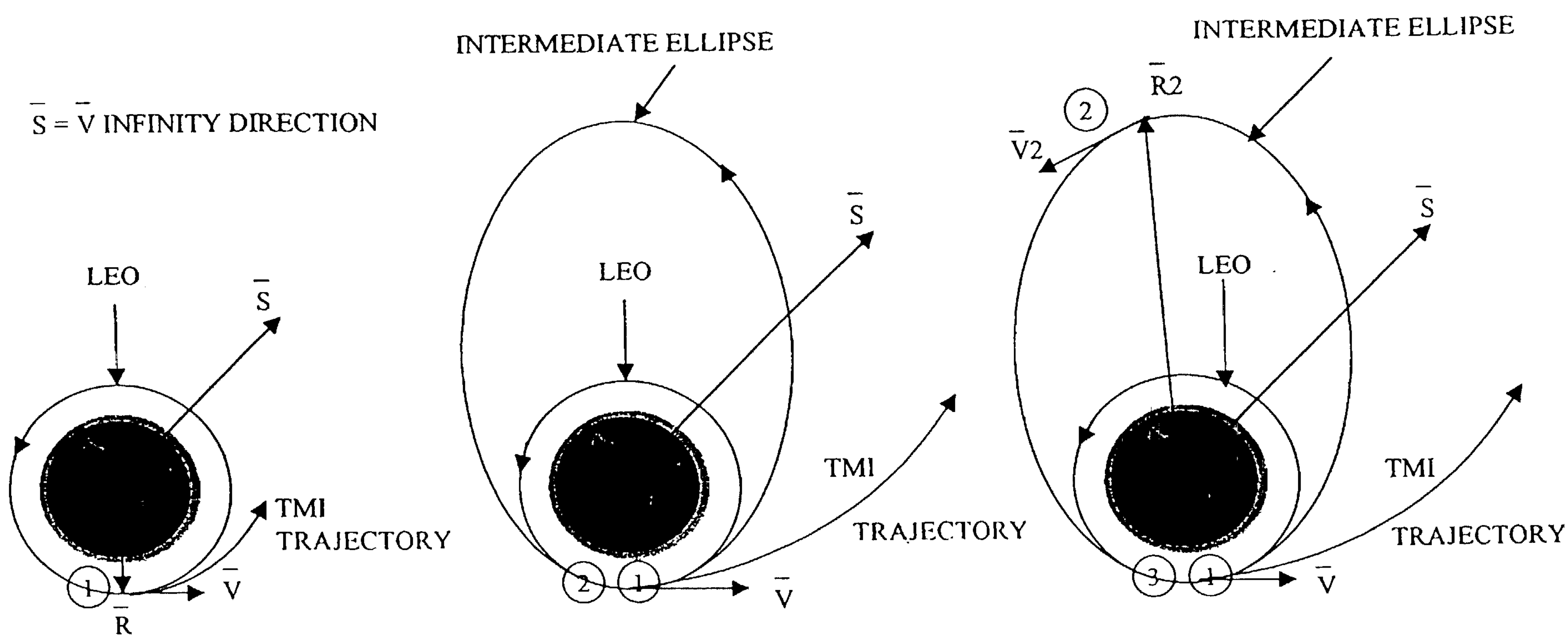
LAUNCH WINDOW FROM LOW EARTH ORBIT (LEO)

This chart shows the orbital geometry for a one, two and three burn maneuver to achieve Trans-Mars Injection (TMI). The one burn maneuver, which includes an orbit energy change combined with a plane change, is made near the LEO perigee. When the thrust to initial weight ratio is relatively high (≥ 0.6) and the required plane change angle is small ($\leq 3.0°$) the one burn maneuver would be efficient to achieve TMI. When the thrust to initial weight ratio is relative low(≤ 0.25) and the required plane change is low ($\leq 3.0°$) the two burn maneuver would be more efficient than the one burn maneuver because the gravity loss would be considerably lower. The first burn would place the Mars' vehicle into an intermediate ellipse (~4.8 hr period) where the vehicle would coast one orbit back near perigee (at point (2)). At this point the second burn is made which includes an orbit energy change and a plane change to achieve TMI.

The three burn maneuver for TMI may be more efficient when the thrust to initial weight ratio is low (≤ 0.25) and the required plane change is greater than three degrees. The first burn places the Mars vehicle into an intermediate ellipse (~4.8 hr period). The vehicle coasts to just past apogee of the intermediate ellipse where a second burn is made to change the intermediate orbit plane to contain the outgoing asymptote vector (S^-). After the second burn the vehicle coasts back to near perigee where the third burn, inplane, maneuver is made to achieve TMI.

The results from all three different types of burn maneuvers need to be determined across the duration of the launch window to identify which is the most efficient on a given day.

LAUNCH WINDOW FROM LOW EARTH ORBIT (LEO)



(1) ONE BURN MANEUVER FOR TMI (INCLUDES PLANE CHANGE)

TWO BURN MANEUVER
(1) BURN INTO INTERMEDIATE ELLIPSE (~ 4.8 HR PERIOD)
(2) TMI BURN (INCLUDES PLANE CHANGE)

THREE BURN MANEUVER
(1) BURN INTO INTERMEDIATE ELLIPSE (~ 4.8 HR PERIOD)
(2) PLANE CHAGE BURN
(3) TMI BURN

LAUNCH WINDOW FROM A HIGH ELLIPTICAL ORBIT (HEO)

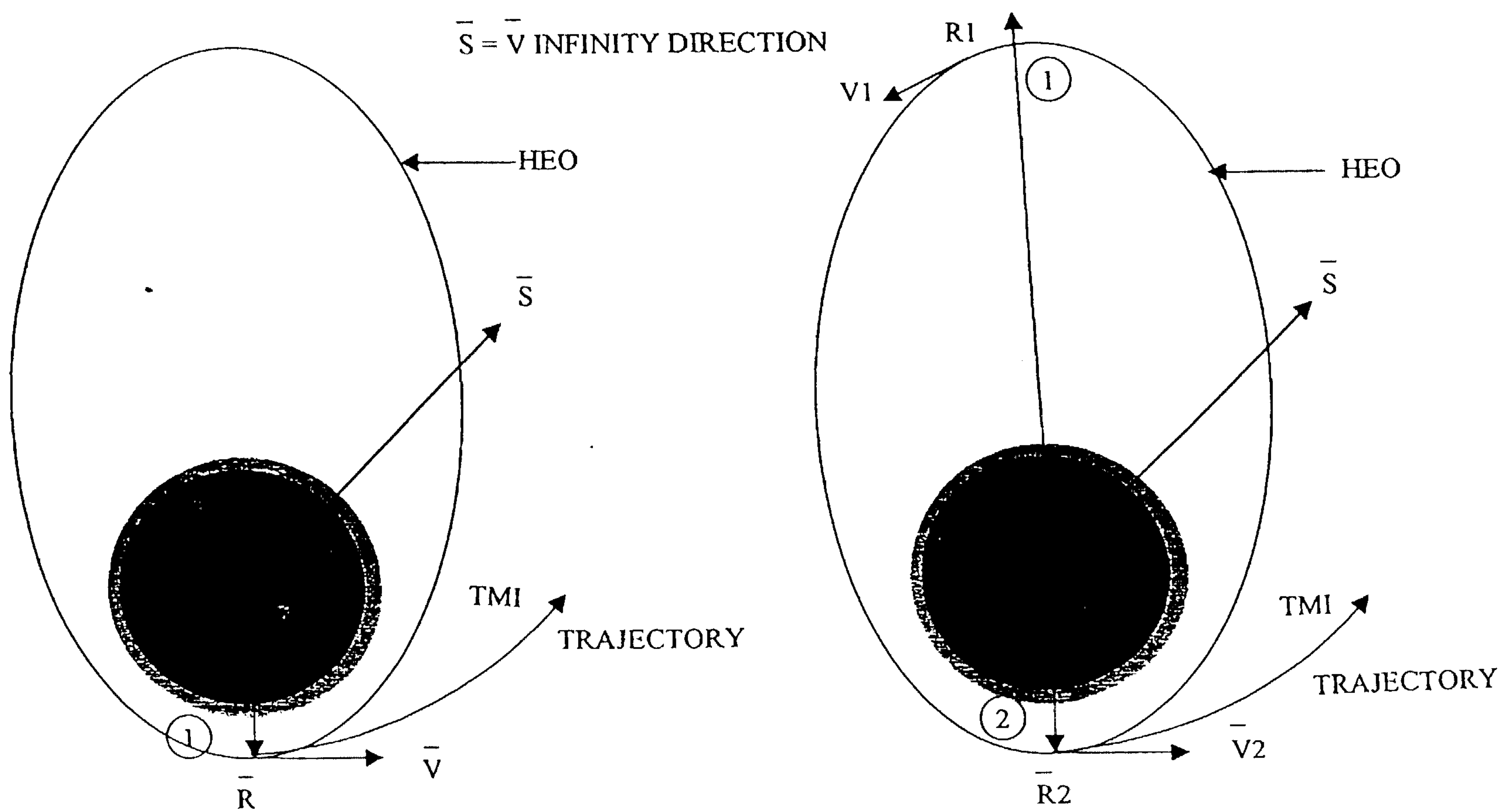
The duration of a planetary launch window depends on the orbital dynamic change between the orbit plane and outgoing asymptote vector ($\bar{S}$) and the hyperbolic energy level required for the earth's Trans-Mars injection (TMI). The parameters that affect the orbital launch window include the parking orbit inclination, perigee altitude, apogee altitude, and the declination and right ascension of the outgoing asymptote vector ($\bar{S}$).

This chart shows the orbital geometry for a one burn maneuver and a two burn maneuver to achieve TMI from a High Elliptical Orbit (HEO). When the angle between the orbit plane and the outgoing asymptote vector ($\bar{S}$) is small, the TMI may be made efficiently with a single burn maneuver. The single burn maneuver will include an orbit energy change combined with a plane change. When the angle between the orbit plane and $\bar{S}$ is large, however, the Δv requirement can be prohibitively large. Thus a more efficient means of attaining TMI is desired. This can be achieved by dividing the TMI maneuver into two burns. The first burn maneuver is made near the HEO apogee, at point (1), which change the HEO plane to contain the outgoing asymptote vector, creating an inplane maneuver for the second burn near HEO perigee, at point (2) to complete TMI.

Both the single burn maneuver and two burn maneuver results need to be determined across the duration of the launch window to identify which is the more efficient on a given day.

LAUNCH WINDOW FROM A HIGH ELLIPTICAL ORBIT (HEO)

$\bar{S} = \bar{V}$ INFINITY DIRECTION



(1) ONE BURN MANEUVER FOR TMI (INCLUDES PLANE CHANGE)

TWO BURN MANEUVER
(1) PLANE CHANGE BURN
(2) TMI BURN

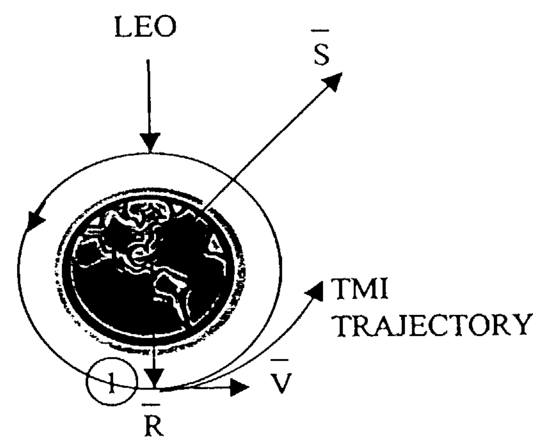
LAUNCH WINDOW FROM LOW EARTH ORBIT (LEO)

One Burn Option (Perigee Burn)

This section presents a one burn option computer simulation for the orbital launch window analyses for out-of-plane conditions based upon a nominal in-plane launch date/conditions and time offset from that date. This situation applies to pre-deployed space systems in an existing orbit, with assembly complete, that is utilizing an early or late departure ($< \pm 2$ weeks) date relative to in-plane date. This existing orbit plane will not contain the V-infinity vector of the Earth escape trajectory, i.e. the right ascension (RA) and declination (Dec) of the injection/orbital - "launch" condition will be off optimal. Therefore an orbit plane change, flight path angle and orbit energy change must be made to achieve the desired Earth escape trajectory.

LAUNCH WINDOW FROM LOW EARTH ORBIT (LEO)

$\bar{S} = \bar{V}$ INFINITY DIRECTION



1 ONE BURN MANEUVER
FOR TMI (INCLUDES PLANE
CHANGE) ± 30° AROUND PERIGEE

ORBITAL LAUNCH WINDOW
ONE BURN SOLUTION
FROM LOW EARTH ORBIT

h_{p1} = 400 km, h_{a1} = 400 km, i = 40°
Use Equation (2) to Determine Ω_0
$\rho_1 - w_1 = 300$, $\Delta\rho_1 = 1$, $k_1 = 120$

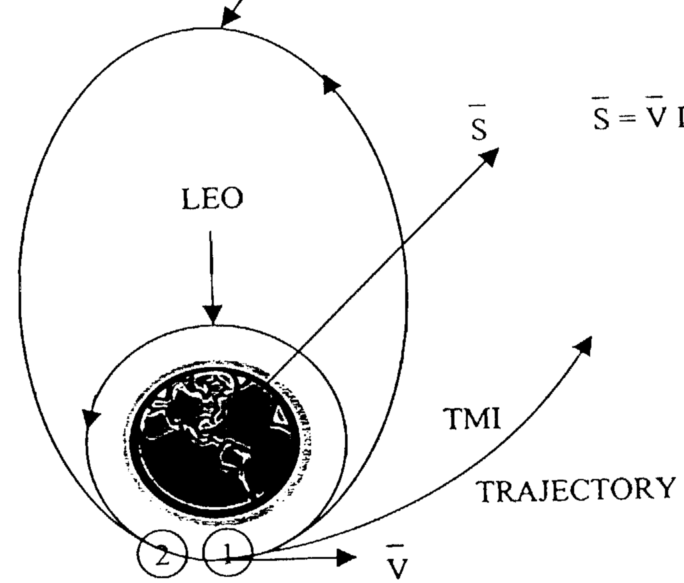
TMI DATE CALANDER	TMI DATE JULIAN	α (R.A.)	δ (Dec)	C_3	V∝
04/05/18	58214	333.20	-41.10	13.9584	3.73610
04/07/18	58216	332.86	-40.60	13.3726	3.65686
04/09/18	58218	332.43	-40.15	12.8367	3.58283
04/11/18	58220	331.93	-39.74	12.3481	3.51398
04/13/18	58222	331.34	-39.37	11.9046	3.45030
04/15/18	58224	330.65	-39.05	11.5043	3.39180
04/17/18	58226	329.87	-38.78	11.1458	3.33853
04/19/18	58228	328.89	-38.52	10.7923	3.28517
04/21/18	58230	328.02	-38.36	10.5504	3.24814
04/23/18	58232	326.95	-38.20	10.3128	3.21136
04/25/18	58234	325.77	-38.07	10..1155	3.18048
04/27/18	58236	324.49	-37.96	9.9589	3.15577
04/29/18	58238	323.12	-37.87	9.8443	3.13756
05/01/18	58240	321.66	-37.79	9.7728	3.12615
05/03/18	58242	320.11	-37.71	9.7462	3.12189
05/05/18	58244	318.49	-37.62	9.7664	3.12513

LAUNCH WINDOW FROM LOW EARTH ORBIT (LEO)

Two Burn Option (Perigee-Perigee)

This section presents a two burn option computer simulation for the orbital launch window analyses for out-of-plane conditions based upon a nominal in-plane launch date/conditions and time offset from that date. This situation applies to pre-deployed space systems in an existing orbit, with assembly complete, that is utilizing an early or late departure (< ± 2 weeks) date relative to in-plane date. This existing orbit plane will not contain the V-infinity vector of the Earth escape trajectory, i.e. the right ascension (RA) and declination (Dec) of the injection/orbital - "launch" condition will be off optimal. Therefore an orbit plane change, flight path angle and orbit energy change must be made to achieve the desired Earth escape trajectory.

LAUNCH WINDOW FROM LOW EARTH ORBIT (LEO)



TWO BURN MANEUVER, ± 30° AROUND PERIGEE

1. BURN INTO INTERMEDIATE ELLIPSE (~ 4.8 HR PERIOD)
2. TMI BURN (INCLUDES PLANE CHANGE)

ORBITAL LAUNCH WINDOW
TWO BURN SOLUTION
FROM LOW EARTH ORBIT

$hp_1 = 400$ km, $ha_1 = 400$ km, $i = 40°$
$hp_2 = 400$ km, $ha_2 = 16533$ km (5 HR Ellipse)
Use Equation (2) to Determine Ω_0
$\rho_1 - \omega_1 = 0$, $\rho_2 - \omega_2 = 300$, $\Delta\rho_2 = 1$, $k_2 = 120$

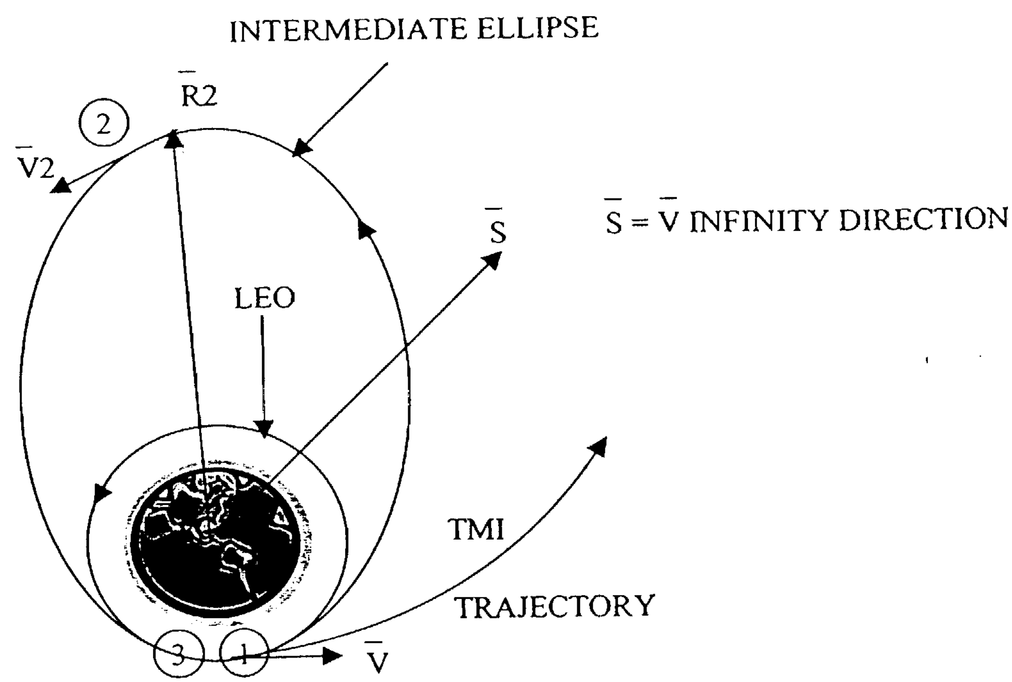
TMI DATE CALANDER	TMI DATE JULIAN	α (R.A.)	δ (Dec)	C_3	V_∞
04/07/18	58216	332.91	-40.72	13.34835	3.65354
04/09/18	58218	332.49	-40.26	12.80931	3.57901
04/11/18	58220	331.94	-39.85	12.31806	3.50971
04/13/18	58222	331.38	-39.49	11.87244	3.44564
04/15/18	58224	330.68	-39.16	11.47035	3.38679
04/17/18	58226	329.90	-38.89	11.11042	3.33323
04/19/18	58228	329.02	-38.65	10.79155	3.28505
04/21/18	58230	328.02	-38.45	10.50907	3.24177
04/23/18	58232	326.95	-38.29	10.27485	3.20544
04/25/18	58234	325.77	-38.16	10.07694	3.17442
04/27/18	58236	324.48	-38.04	9.91998	3.14960
04/29/18	58238	323.10	-37.95	9.80491	3.13128
05/01/18	58240	321.63	-37.86	9.73296	3.11977
05/03/18	58242	320.08	-37.78	9.70584	3.11542
05/05/18	58244	318.45	-37.69	9.72542	3.11856
05/07/18	58246	316.77	-37.58	9.79396	3.12953

Gravity losses, all propulsive, 1 sol at Mars, 40 day stay, NTP

LAUNCH WINDOW FROM LOW EARTH ORBIT (LEO)
Three Burn Option (Perigee-Apogee-Perigee)

This section presents a three burn option computer simulation for the orbital launch window analyses for out-of-plane conditions based upon a nominal in-plane launch date/conditions and time offset from that date. This situation applies to pre-deployed space systems in an existing orbit, with assembly complete, that is utilizing an early or late departure ($< \pm 2$ weeks) date relative to in-plane date. This existing orbit plane will not contain the V-infinity vector of the Earth escape trajectory, i.e. the right ascension (RA) and declination (Dec) of the injection/orbital - "launch" condition will be off optimal. Therefore an orbit plane change, flight path angle and orbit energy change must be made to achieve the desired Earth escape trajectory.

LAUNCH WINDOW FROM LOW EARTH ORBIT (LEO)

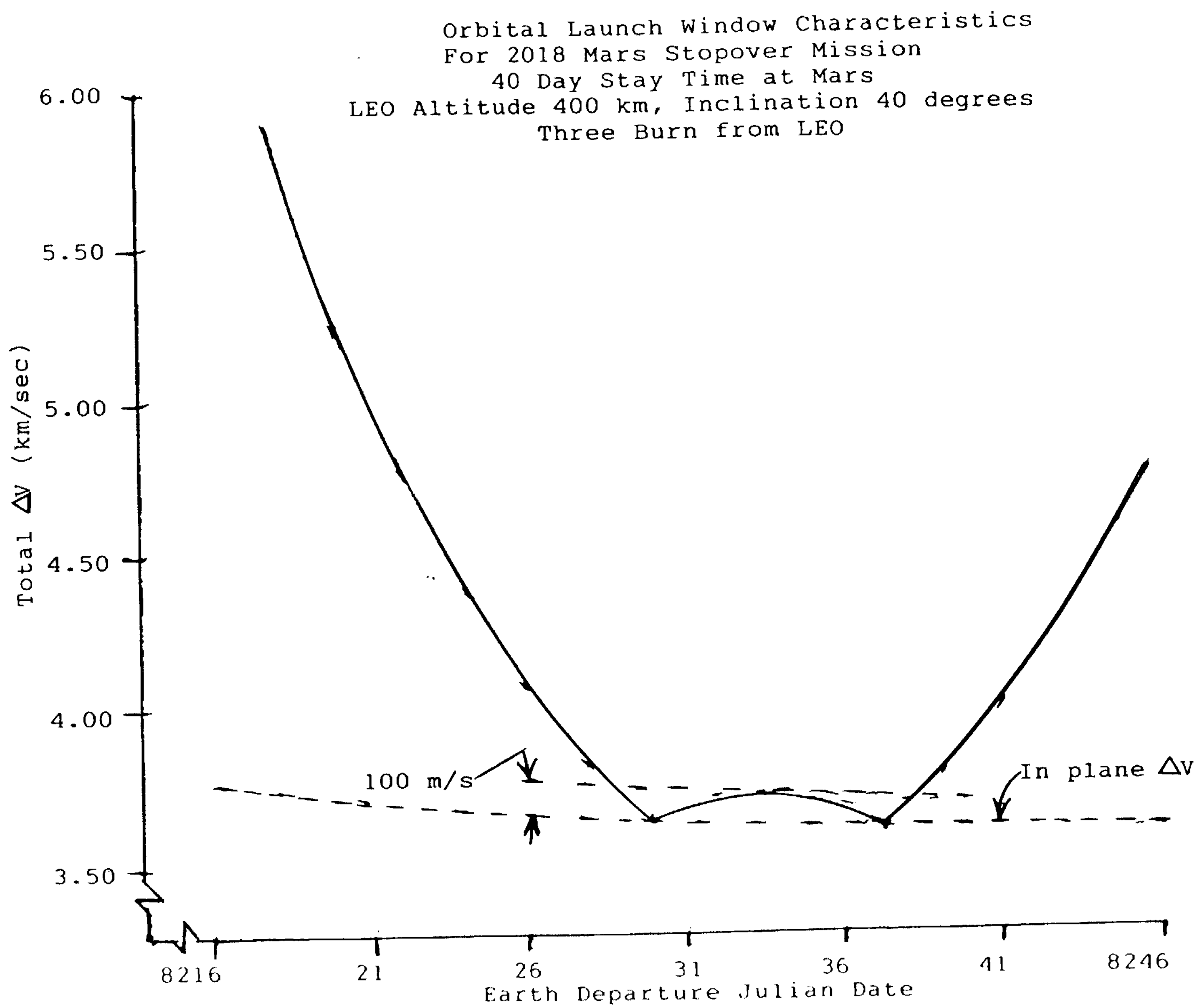


THREE BURN MANEUVER
1 BURN INTO INTERMEDIATE ELLIPSE (~ 4.8 HR PERIOD)
2 PLANE CHAGE BURN
3 TMI BURN
BURNS 1 AND 3 MADE ± 30° AROUND PERIGEE
BURN 2 MADE ± 40° AROUND APOGEE

Orbital Launch Window Characteristics
For 2018 Mars Stopover Mission
Three Burn from LEO

Parameteric orbital launch window data were obtained for a 2018 Mars stopover mission; where the stay time at Mars is 40 days. The parking low earth orbit (LEO) is a 400 km circular orbit altitude with an inclination of 40 degrees. The table presented in this section contains the required injection conditions (α(R.A.); δ(Dec); and C_3) for trans-Mars injection (TMI) for a 30 day in-plane launch window.

The in-plane departure date selected is Julian date 8230 (calendar date 04/21/18). The orbit plane regresses around such that the LEO plane contains the V-infinity vector again on Julian date 8237. TMI departing on any date other than these two dates would incur a plane change ΔV penalty. An out-of-plane launch window ΔV of 100 m/sec would allow an orbit launch window of about nine days, 8229 to 8238.



Orbital Launch Window Characteristics
For 2018 Mars Stopover Mission
40 Day Stay Time at Mars
LEO Altitude 400 km, Inclination 40 degrees
Three Burn from LEO
Total ΔV (km/sec)
6.00
5.50
5.00
4.50
4.00
3.50
100 m/s
In plane ΔV
8216
21
26
31
36
41
8246
Earth Departure Julian Date

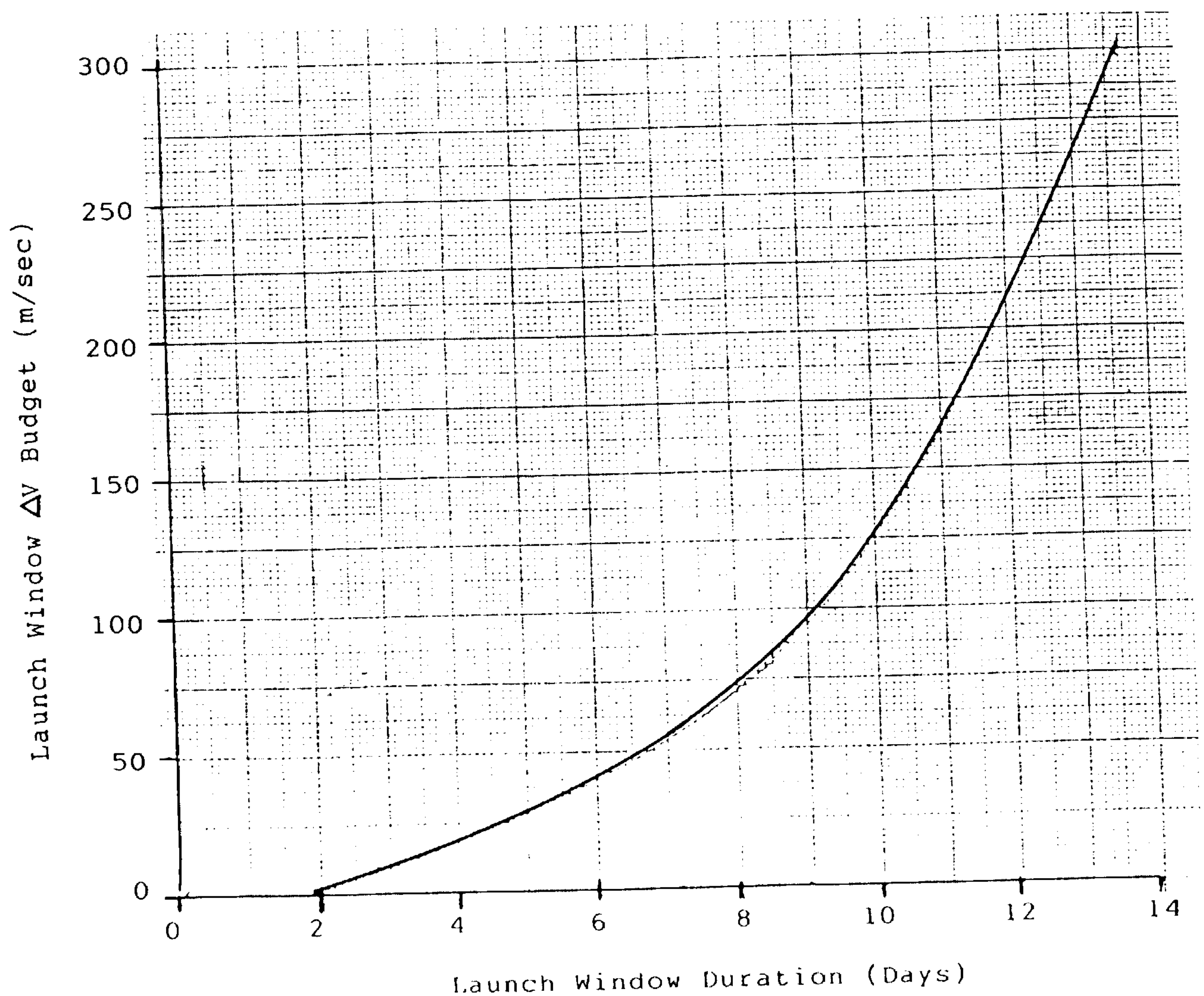
Launch Window Duration
For 2018 Mars Stopover Mission
Three Burn from LEO

Shown on the orbital launch window characteristics graph (previous graph) there are two orbital launch dates which are co-planer with the trans-Mars-injection V-infinity vector. Therefore there would be two days which would not require any orbital launch window ΔV budget.

An orbital launch window ΔV budget of 50 m/sec would allow an orbit launch window of about seven days. A ΔV budget of 125 m/sec would allow an orbit launch window of ten days

Once passed a launch window duration of nine days, the ΔV required for longer launch window duration increases rapidly. This rapid increase is due to the fast build up of the required plane change.

Launch Window Duration
For 2018 Mars Stopover Mission
40 Day Stay Time at Mars
LEO Altitude 400 km, Inclination 40 degrees
Three Burn from LEO



ORBITAL LAUNCH WINDOW
THREE BURN SOLUTION
FROM LOW EARTH ORBIT

$hp_1 = 400$ km, $ha_1 = 400$ km, $i = 40°$
$hp_2 = 400$ km, $ha_2 = 16533$ km (5 HR Ellipse)
Use Equation (2) to Determine Ω_0
$\rho_1 - \omega_1 = 0$, $\rho_2 - \omega_2 = 150$, $\Delta\rho_2 = 1$, $k_2 = 90$
$\rho_3 - \omega_3 = 320$, $\Delta\rho_3 = 1$, $k_3 = 80$

TMI DATE CALANDER	TMI DATE JULIAN	α (R.A.)	δ (Dec)	C_3	$V\propto$
04/07/18	58216	332.91	-40.72	13.34835	3.65354
04/09/18	58218	332.49	-40.26	12.80931	3.57901
04/11/18	58220	331.94	-39.85	12.31806	3.50971
04/13/18	58222	331.38	-39.49	11.87244	3.44564
04/15/18	58224	330.68	-39.16	11.47035	3.38679
04/17/18	58226	329.90	-38.89	11.11042	3.33323
04/19/18	58228	329.02	-38.65	10.79155	3.28505
04/21/18	58230	328.02	-38.45	10.50907	3.24177
04/23/18	58232	326.95	-38.29	10.27485	3.20544
04/25/18	58234	325.77	-38.16	10.07694	3.17442
04/27/18	58236	324.48	-38.04	9.91998	3.14960
04/29/18	58238	323.10	-37.95	9.80491	3.13128
05/01/18	58240	321.63	-37.86	9.73296	3.11977
05/03/18	58242	320.08	-37.78	9.70584	3.11542
05/05/18	58244	318.45	-37.69	9.72542	3.11856
05/07/18	58246	316.77	-37.58	9.79396	3.12953

Gravity losses, all propulsive, 1 sol at Mars, 40 day stay, NTP

LAUNCH WINDOW FROM LOW EARTH ORBIT (LEO)
Three Burn Option (Perigee-Apogee-Perigee)

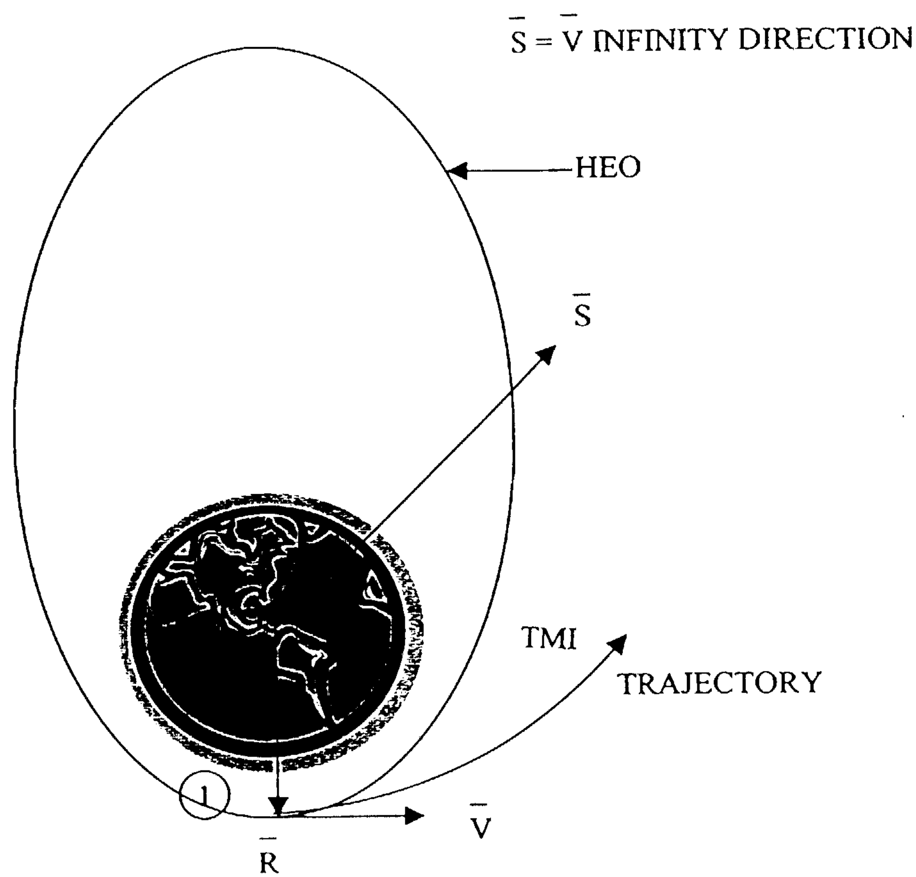
The MAiNe runs for the LEO three burn option are the same as for the LEO two burn option.

LAUNCH WINDOW FROM A HIGH ELLIPTICAL ORBIT (HEO)

One Burn Option (Perigee Burn)

This section presents a one burn option computer simulation for the orbital launch window analyses for out-of-plane conditions based upon a nominal in-plane launch date/conditions and time offset from that date. This situation applies to pre-deployed space systems in an existing orbit, with assembly complete, that is utilizing an early or late departure ($< \pm 2$ weeks) date relative to in-plane date. This existing orbit plane will not contain the V-infinity vector of the Earth escape trajectory, i.e. the right ascension (RA) and declination (Dec) of the injection/orbital - "launch" condition will be off optimal. Therefore an orbit plane change, flight path angle and orbit energy change must be made to achieve the desired Earth escape trajectory.

LAUNCH WINDOW FROM A HIGH ELLIPTICAL ORBIT (HEO)



1 ONE BURN MANEUVER
FOR TMI (INCLUDES PLANE CHANGE)
± 30° AROUND PERIGEE

ORBITAL LAUNCH WINDOW
ONE BURN SOLUTION
FROM HIGH ELLIPTICAL ORBIT

h_{p1} = 800 km, h_{a1} = 120,551 km, i = 40°
$\rho_1 - \omega_1$ = 320, $\Delta\rho_1$ =1, k_1 = 80
Use Equation (2) to Determine Ω_0

TMI DATE CALANDER	TMI DATE JULIAN	α (R.A.)	δ (Dec)	C_3	V∝
04/04/18	58213	333.36	-41.40	14.26497	3.77690
04/06/18	58215	333.06	-40.88	13.65206	3.69487
04/08/18	58217	332.67	-40.41	13.09050	3.61808
04/10/18	58219	332.21	-39.97	12.57773	3.54651
04/12/18	58221	331.66	-39.58	12.11130	3.48013
04/14/18	58223	331.02	-39.24	11.68901	3.41892
04/16/18	58225	330.28	-38.94	11.30937	3.36294
04/18/18	58227	329.46	-38.69	10.97100	3.31225
04/20/18	58229	328.62	-38.50	10.69820	3.27081
04/22/18	58231	327.50	-38.30	10.41527	3.22727
04/24/18	58233	326.37	-38.16	10.19761	3.19337
04/26/18	58235	325.14	-38.04	10.02045	3.16551
04/28/18	58237	323.81	-37.94	9.88461	3.14398
04/30/18	58239	322.39	-37.85	9.79127	3.12910
05/02/18	58241	320.88	-37.77	9.74189	3.12120
05/04/18	58243	319.30	-37.69	9.73833	3.12063

Gravity losses, all propulsive, 1 sol at Mars, 40 day stay, NTP

LAUNCH WINDOW FROM A HIGH ELLIPTICAL ORBIT (HEO)
One Burn Option (Perigee Burn)

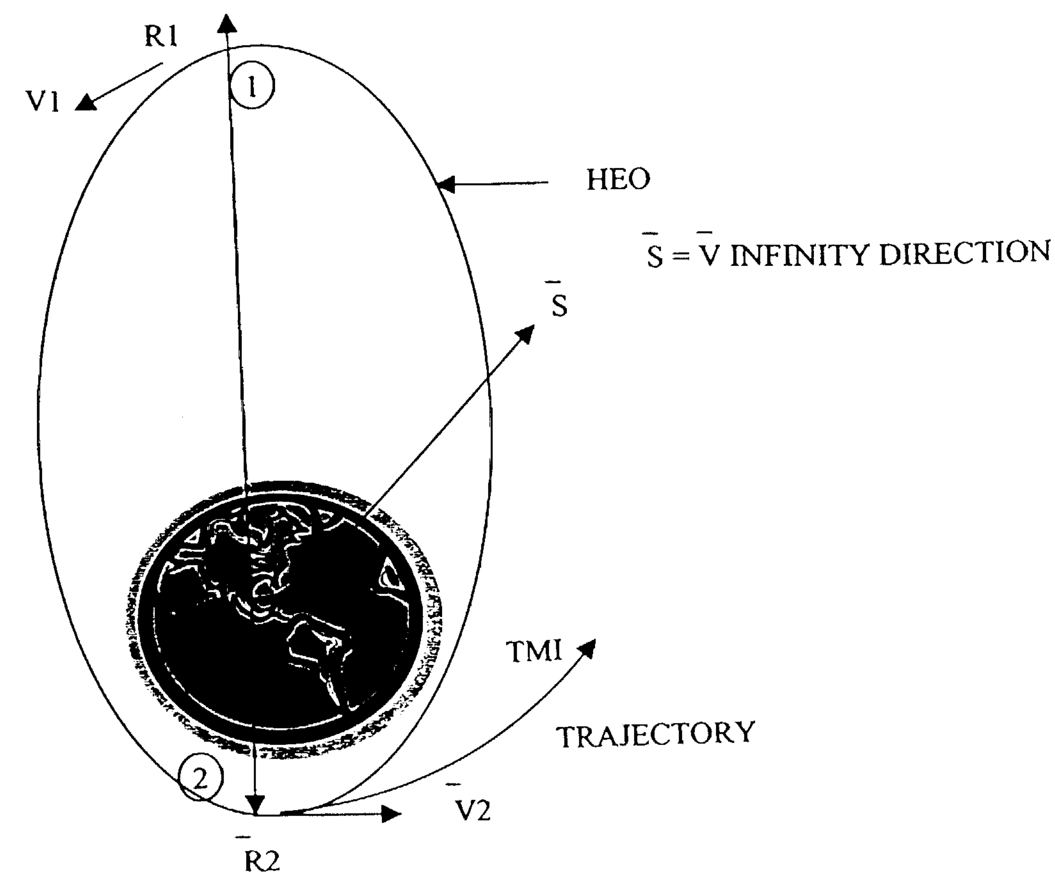
The computer simulation for the HEO one burn option is the same as for the low earth orbit one burn option.

LAUNCH WINDOW FROM A HIGH ELLIPTICAL ORBIT (HEO)

Two Burn Option (Apogee-Perigee)

This section presents a two burn option computer simulation for the orbital launch window analyses for out-of-plane conditions based upon a nominal in-plane launch date/conditions and time offset from that date. This situation applies to pre-deployed space systems in an existing orbit, with assembly complete, that is utilizing an early or late departure ($< \pm 2$ weeks) date relative to in-plane date. This existing orbit plane will not contain the V-infinity vector of the Earth escape trajectory, i.e. the right ascension (RA) and declination (Dec) of the injection/orbital - "launch" condition will be off optimal. Therefore an orbit plane change, flight path angle and orbit energy change must be made to achieve the desired Earth escape trajectory.

LAUNCH WINDOW FROM A HIGH ELLIPTICAL ORBIT (HEO)



TWO BURN MANEUVER

1. PLANE CHANGE BURN, ± 40° AROUND APOGEE
2. TMI BURN, ± 30° AROUND PERIGEE

ORBITAL LAUNCH WINDOW
TWO BURN SOLUTION
FROM HIGH ELLIPTICAL ORBIT

hp_1 = 800 km, ha_1 = 120,551 km, i = 40°
$\rho_1 - \omega_1 = 150$, $\Delta\rho_1 = 1$, $k_1 = 90$
$\rho_2 - \omega_2 = 320$, $\Delta\rho_2 = 1$, $k_2 = 80$
Use Equation (2) to Determine Ω_0

TMI DATE CALANDER	TMI DATE JULIAN	α (R.A.)	δ (Dec)	C_3	V∝
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04/12/18	58221	331.66	-39.58	12.11130	3.48013
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04/20/18	58229	328.62	-38.50	10.69820	3.27081
04/22/18	58231	327.50	-38.30	10.41527	3.22727
04/24/18	58233	326.37	-38.16	10.19761	3.19337
04/26/18	58235	325.14	-38.04	10.02045	3.16551
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05/04/18	58243	319.30	-37.69	9.73833	3.12063

Gravity losses, all propulsive, 1 sol at Mars, 40 day stay, NTP

LAUNCH WINDOW FROM A HIGH ELLIPTICAL ORBIT (HEO)
Two Burn Option (Apogee-Perigee)

The MAiNe runs for the HEO two burn option are the same as for the HEO one burn option.