SPACE VEHICLE DEPLOYMENT FROM SPACE STATION ORBIT

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ABSTRACT

When launching a spacecraft from Earth parking orbit to deep space, it is highly desirable to have the hyperbolic excess velocity vector (v-infinity) contained in the parking orbit plane. Ground launches can force the parking orbit plane to contain the v-infinity vector by using launch azimuth and lift-off time as independent variables. When launching from the Space Station, a new set of variables comes into play. The Station orbit is of fixed inclination but precessing due to the Earth's oblateness. Its plane will seldom (and may never) contain the desired v-infinity vector. Consequently, the departure strategy will usually require multiple burns and include a plane change. Also, the concept of "launch period" will be somewhat different from Earth surface launches. An analysis of the deployment of interplanetary spacecraft from Space Station is described, with emphasis on the effect of the trajectory characteristics on station operations. Several planetary mission types are analyzed for manned Mars missions. In addition, high declination departures of spacecraft on unmanned missions to an asteroid are examined. The constraint of Station orbit nodal position is quantified and the operational implications for station reboost strategy are examined.

SPACE VEHICLE DEPLOYMENT FROM SPACE STATION

BACKGROUND

- PLANETARY MISSION LAUNCHES FROM THE GROUND USE LAUNCH TIME AND AZIMUTH TO CONTROL INCLINATION AND NODE OF PARKING ORBIT
- GROUND LAUNCHES CAN FORCE PARKING ORBIT AND DEPARTURE HYPERBOLA FROM PARKING ORBIT TO BE CO-PLANAR.
- SPACE STATION INCLINATION (FIXED AT 28.5°) AND NODAL POSITION IS A FUNCTION OF FIRST ELEMENT LAUNCH TIME AND NODAL REGRESSION RATE HISTORY (ALTITUDE DEPENDENT)
- SPACE STATION ORBIT ONLY BRIEFLY (AND PERHAPS NEVER) CO-PLANAR WITH DESIRED INTERPLANETARY TRANSFER TRAJECTORY

BACKGROUND

The purpose of this study was to identify the operational requirements and constraints on Space Station *Freedom* resulting from the use of the station to deploy spacecraft on manned missions to Mars and unmanned missions to high declination targets such as asteroids or comets.

Planetary departures from the orbit of a space station fundamentally differ from ground launches. A surface launch to a planetary target allows an orientational targeting choice by careful selection of launch time and ascent azimuth direction; the pre-existing station orbit provides no such options. Further, the orientation of the station's orbit continuously changes due to the oblateness of the equatorial bulge of the Earth, which causes a relatively rapid regression of the station's orbital plane (by about -7.2 deg/day). An orbital launch window occurs every time the regressing orbit plane sweeps over the V-infinity vector of the transplanetary Earth escape hyperbola for the target planet considered. At all other times, energy-expensive plane change maneuvers are required at departure (References 1 and 2).

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SPACE VEHICLE DEPLOYMENT FROM SPACE STATION

OBJECTIVE

- ANALYZE THE ORBITAL MECHANICS AND TRAJECTORY OPTIONS FOR DEPLOYING PLANETARY SPACECRAFT FROM THE STATION
- IDENTIFY THE EFFECT OF DEPLOYMENT TRAJECTORY REQUIREMENTS ON STATION OPERATIONS
  - REBOOST STRATEGY
  - DEPARTURE PERIOD MAXIMIZATION
  - PROPELLENT REQUIREMENTS
- FOCUS ON PILOTED MISSIONS TO MARS
  (NOTE: THIS WORK WAS COMPLETED PRIOR TO THE ANNOUNCEMENT OF THE HUMAN EXPLORATION INITIATIVE)

OBJECTIVE

This report summarizes a study performed at the Jet Propulsion Laboratory in fiscal year 1989. The study examined the trajectory issues involved in using Space Station *Freedom* as the departure site for piloted missions to Mars, and unpiloted missions on high-declination departure trajectories to planetary bodies such as asteroids and comets. Previous studies in this area (References 1 and 2) dealt with a broad range of issues, including assembly of spacecraft at the station, the effects of mission staging on other payloads, and safety. Reference 1 also presented a preliminary examination of the trajectory issues for unmanned spacecraft. This study focusses exclusively on the trajectory issues identified in the previous studies, and examines the effects on station operations resulting from the interaction of the departing spacecraft trajectory and the orbit of the station. Of particular interest is sensitivity of the on-orbit propellant requirements on the misalignment of the station orbital plane and the plane of the desired transplanetary trajectory. The extent to which the station orbital plane orientation can be managed by modifying the reboost strategy also has operational impacts.

SPACE VEHICLE DEPLOYMENT FROM SPACE STATION

SPACE STATION DEPARTURE CASES STUDIED

CASE	DESTIN-ATION	NOMINAL DEPARTURE DATE, EARTH	ARRIVAL DATE, (FLY BY DATE, VENUS)	TRAJECTORY
EXPEDITION #1	MARS CARGO	2001, 4/15	2002, 1/27	EM
EXPEDITION #2	MARS PILOTED	2002, 9/3	2003, 6/15 (2002, 12/29)	EVME
EVOLUTION #1	MARS PILOTED	2004, 5/31	2005, 4/10 (2004, 11/17)	EVME
EVOLUTION #2	MARS PILOTED	2005, 8/22	2006, 2/13	EME
HIGH DECLIN.	EROS	2005, 1/20	2005, 12/25	DEEP-SPACE PLANE CHANGE RENDEZVOUS
HIGH DECLIN.	EROS	2005, 1/1	2005, 12/25	3-IMPULSE FLY-BY
HIGH DECLIN.	EROS	2005, 1/23	2005, 12/25	3-IMPULSE RENDEZVOUS

SPACE STATION DEPARTURE CASES STUDIED

This phase of the space station staging study focused on an assessment of Earth departure penalties for space-assembled and -launched manned and unmanned Mars missions. Specifically, four sample missions were selected from the then-current (Spring 1989) Gateway Case Study repertoire: two exploration mission cases (flights number one and two) and two evolution mission cases (also flights one and two). Table 1 shows the pertinent characteristics of these sample cases – Earth departure dates, Mars arrival dates, Venus flyby dates (if applicable), and trajectory configuration, where E = Earth, V = Venus and M = Mars.

The work reported here was performed prior to the announcement of the Human Exploration initiative. Consequently, flight opportunities in an earlier time frame – 2001-2005 – were analyzed. The general conclusions should be valid, however, over a wide range of piloted Mars missions.

In this study, the first two ''Mars Expedition'' and ''Mars Evolution'' missions described in Reference 3 were analyzed. These missions include both direct Earth-Mars transfer trajectories and Earth-Venus-Mars flyby gravity assist missions. Also included in this mission set are trajectories that adhere to a free return to Earth'' constraint to maximize crew safety. This constraint would very likely also be applied to the Human Exploration Initiative missions to Mars.

In addition to the Mars missions, the study also examined high declination departures from station orbit, specifically, a mission to the asteroid 433 Eros. High declination missions differ from the Mars missions in that the precessing station orbit plane may never be co-planar with v-infinity vector and a relatively large plane change will be required.

Due to the need to limit the pages of this report to a number commensurate with its intent as a summary document, only the results of the analysis of the Mars second expedition mission will be presented. It is representative of the missions studied in that most relevant points can be illustrated in example form. For the complete analysis, the inquiring reader is referred to the FY89 Final task report: Planetary Exploration Departures from the Space Station: Trajectory Effects on Station Operations, JPL D-6896.

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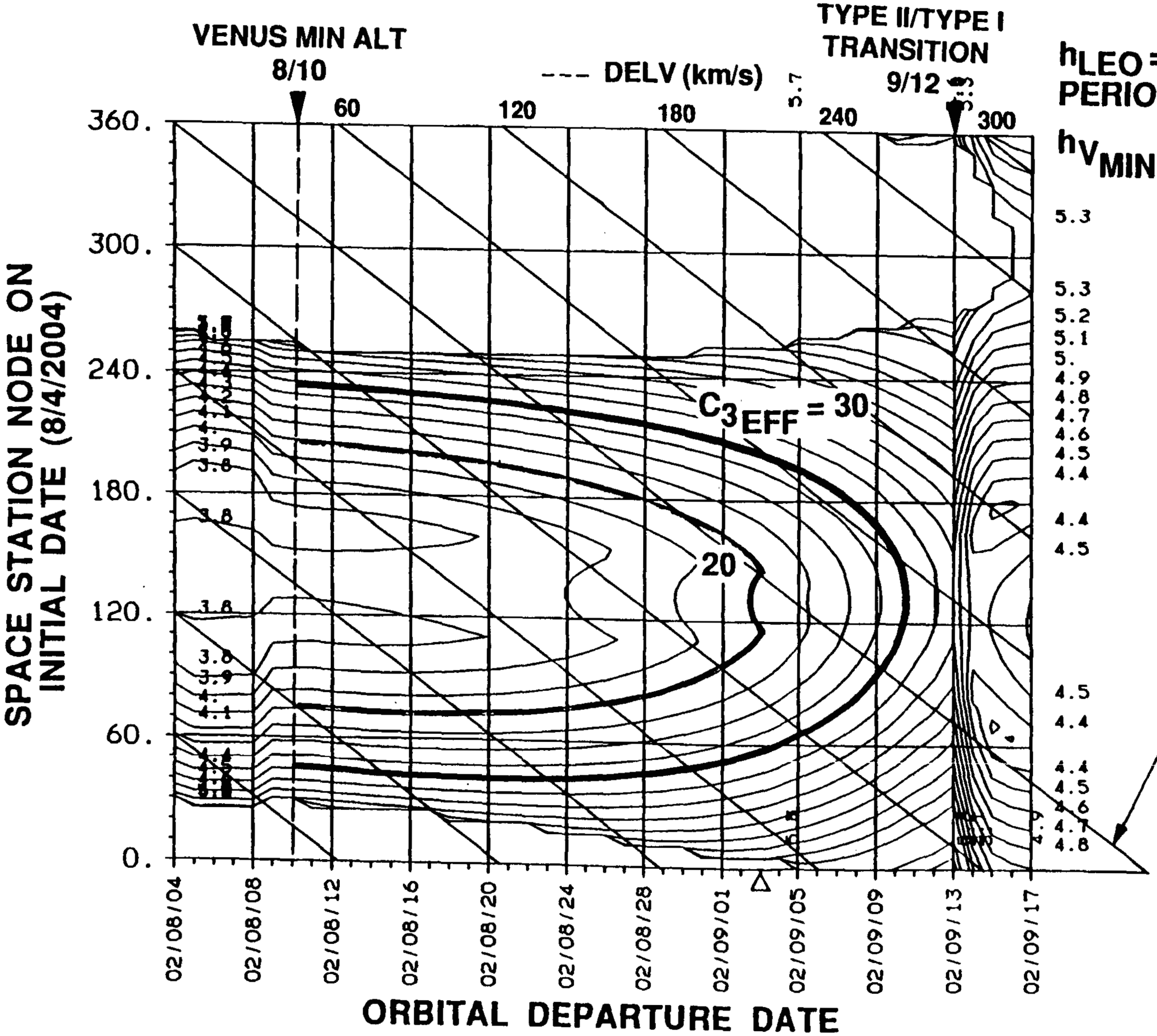
SPACE VEHICLE DEPLOYMENT FROM SPACE STATION
MARS EXPEDITION CASE STUDY FLIGHT No. 2
(3 IMP. INJ., EVME ABORT CAP'Y, MARS ARRIVAL 6/15/2003)



$h_{LEO} = 370$ km
PERIOD 24^h
$h_{V_{MIN}} = 300$ km

NODAL
REGRESSION
$\dot{\Omega} = -7.2°/day$

MARS EXPEDITION CASE STUDY FLIGHT No. 2
(3 IMP. INJ., EVME ABORT CAP'Y, MARS ARRIVAL 6/15/2003)

The orbital departure imposes a plane change penalty on the mission, the implementation of which can be performed as a 3-impulse maneuver: co-planar (with respect to the space station) injection into a 24 hour elliptical parking orbit, a plane change at the high apogee such that the new plane contains the departure V-infinity vector, and a perigee injection burn onto the departure hyperbola.

When evaluated over all possible nodal positions (0-360 degrees) of the space station for each potential departure date, a contour plot of effective C3 can be constructed (shown by bold contour lines in the accompanying figure). Empty, uncontoured regions are ''forbidden'' areas, the result of exceeding the geometric range angle constraint (discussed in more detail in References 1 and 2). The figure clearly shows regions of low (~15 km^2/sec^2) C3, which are forbidden to apsidal 3-impulse maneuvers (Reference 2), and a region of intermediate C3 requirements, sandwiched between the previous two, showing reasonably long departure periods. The thin lines are contours of constant total V_{3IMP} in km/sec. The slanted straight lines labeled ''nodal regression'' represent the continuous shift of nodal longitude of the space station with elapsed time. Hence any departure period from the orbital station of known nodal orientation will lie along one of these slanted lines, as shown. Some periods are short, others long, depending on the nodal longitude at an arbitrary initial reference date and the way the slanted regression line intersects the C3 contours, arranged in a ''horseshoe'' pattern around the forbidden zone. As can be seen in the figure, some departure periods are discontinuous and multiple, while others are uninterrupted over the entire launch period range.

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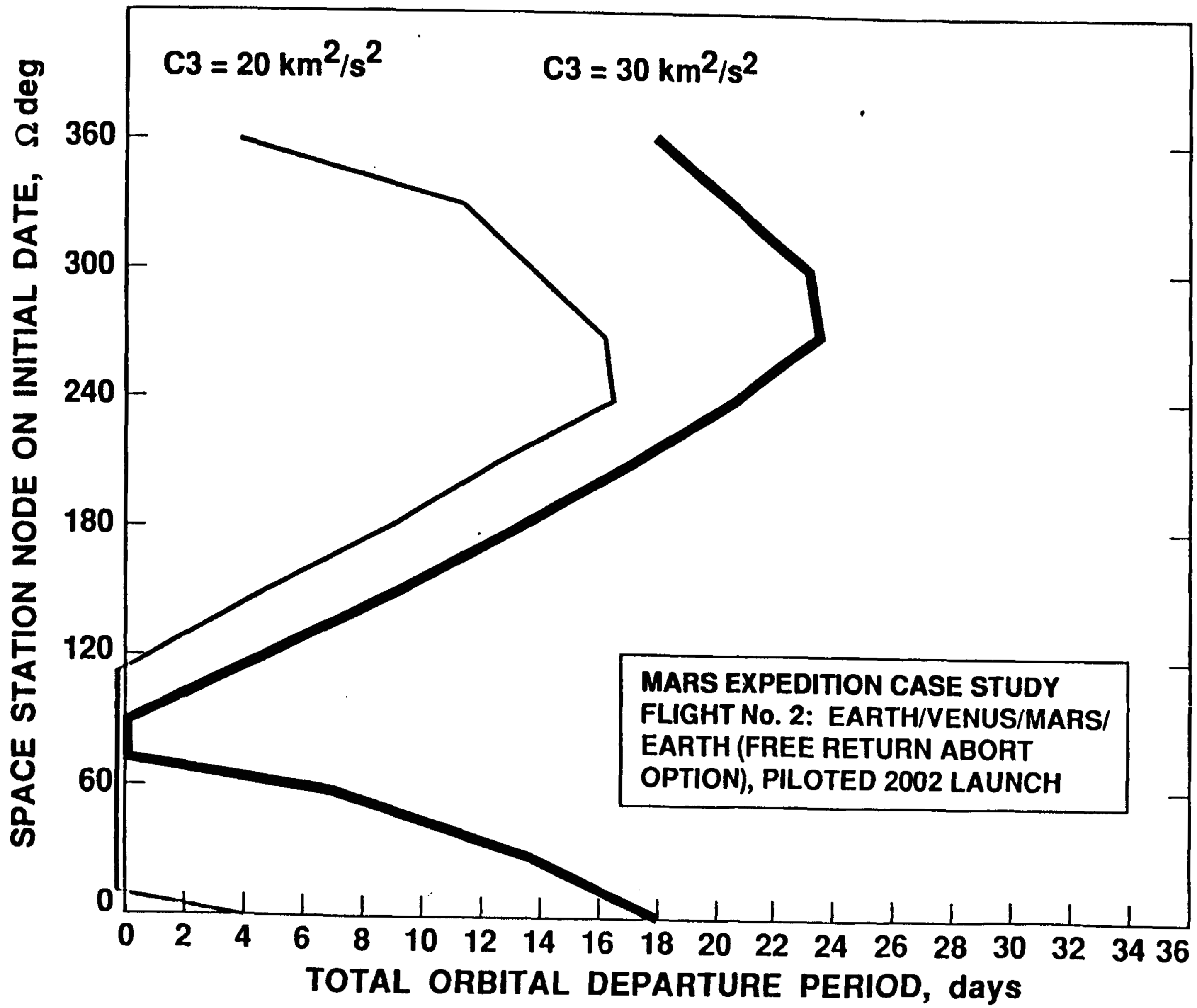
SPACE VEHICLE DEPLOYMENT FROM SPACE STATION ORBITAL DEPARTURE PERIOD AVAILABILITY

(FOR TWO VALUES OF EFFECTIVE DEPARTURE C3 OR PROPELLANT MASS IN m TONS)



ORBITAL DEPARTURE PERIOD AVAILABILITY

The final plot for this mission case shows departure period availability with respect to the station's nodal orientation. Two aspects of this plot should be stressed:

a) If a minimum (single or dual) departure period duration value is specified, one can determine the effective C3 energy required to satisfy that requirement, e.g., for an 20 day guaranteed window, C3 = 30 km^2/sec^2 is needed. This in turn could dictate the maximum allowable payload if an existing injection booster capability is given.

b) If a specific energy is given (i.e., fixed stage and payload), the plot shows the region of the nodal space in which orbital launch *cannot* occur. For instance, if 18 days are required for the departure period in order to make last-minute repairs to the spacecraft, exchange systems or crew, deliver spares from ground or get a second launch off, then, at a C3 = 30 km^2/sec^2, nodal longitudes between 0 and 180 degrees are not allowable. If these are indeed the naturally occurring nodal orientations of the station, then only three options exist:

    1) Go to a higher energy injection stage or stages

    2) Leave some of the payload in Earth orbit

    3) Move the ascending node of the station away from the critical zone. Reboost strategy, as discussed later in this report, is one of the techniques that could be used.

Another aspect of interest is the departure period itself – what is a departure period of any given duration intended to accomplish? Can the number of days between two allowable ''half periods'' be utilized in the waiting process?

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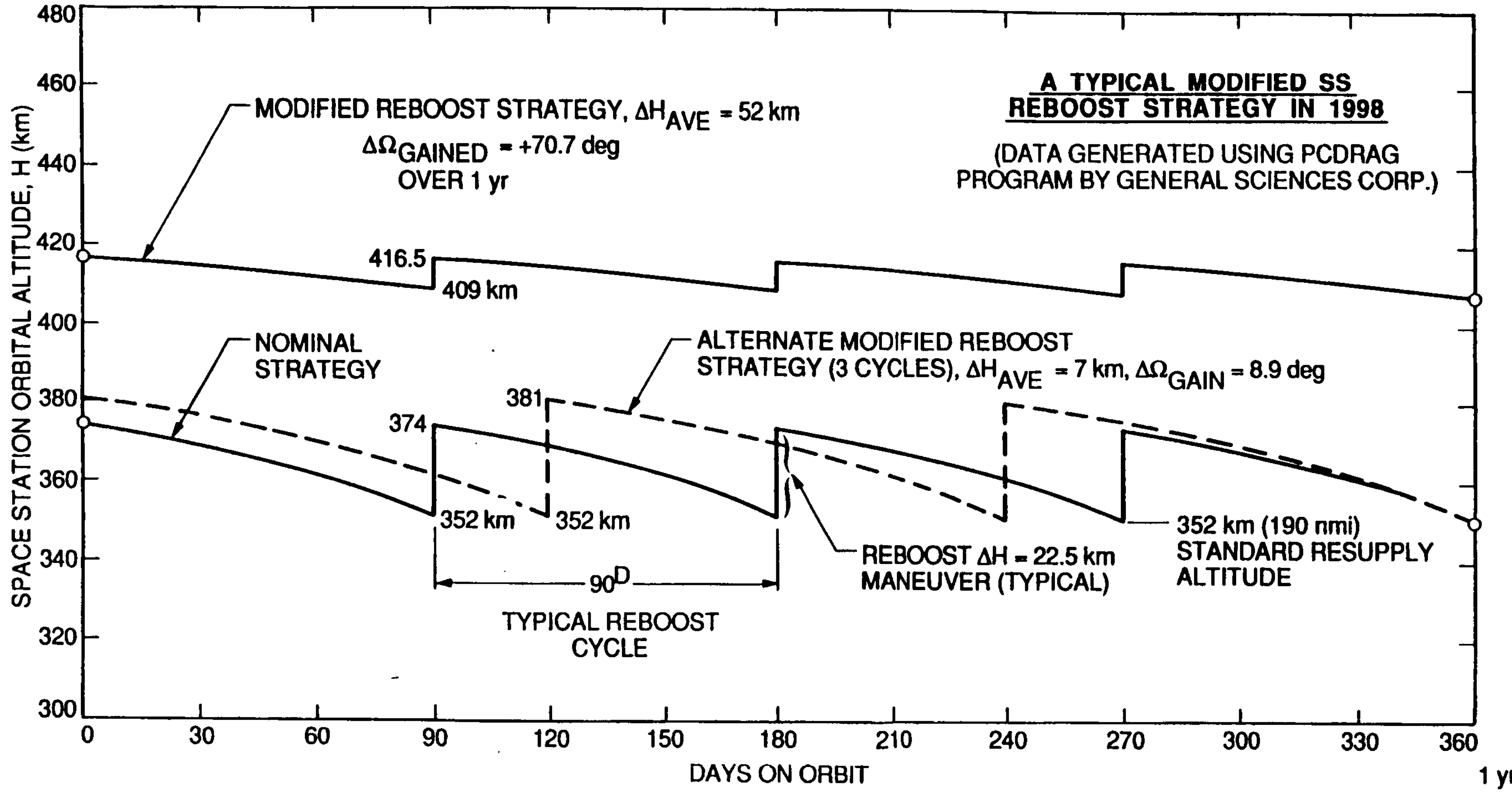
SPACE VEHICLE DEPLOYMENT FROM SPACE STATION

EXAMPLE REBOOST STRATEGY



EXAMPLE REBOOST STRATEGY

It is highly unlikely that a planetary mission departing from the space station will have the luxury of sufficient C3 capability to make its departure independent of the station's nodal position.

Periodic space station altitude reboost maneuvers will be required throughout the station's life. The upper atmosphere produces drag, resulting in the loss of orbital altitude. The amount of drag depends on frontal area mass loading and atmospheric density, which in turn strongly depends on altitude and the state of solar activity – the more active the Sun, the higher the upper atmospheric density and thus higher drag and an increase in the orbital decay rate.

Two example strategies are shown to provide an indication of the sensitivity of nodal regression rate to orbital altitude. Suppose the typical lower/upper altitude bounds of 352-374 km is raised to 409-416.5 km. For a 1998 atmospheric density prediction, both strategies result in a 90 day reboost cycle (four per year), but the higher altitude strategy gains 70.7 degrees of nodal longitude at a year's end. However, the resupply altitude would then be higher, resulting in a lesser payload delivered by the Shuttle. The Shuttle's sensitivity of payload to altitude is approximately -25 kg/km.

The second example strategy shown would raise only the upper altitude bound from 374 to 381 km (same 1998 atmospheric density assumptions). This would result in a change of the reboost cycle duration to 120 days (three per year), and would leave the original resupply altitude unchanged but with a corresponding reduction in frequency of logistics resupply and crew changeout. Also, the gain in nodal longitude would be much less significant: 8.9 degrees change per year. Many other combinations of reboost altitudes are possible, and the reboost strategy may have to be quite complex to accommodate other station activities.

It is quite possible that the strategy to manage the station's nodal position will have to be started well in advance, perhaps years in advance, of the scheduled departure date. Fortunately, the inexorable motion of the planets permits precise advance knowledge of the station's required orbital orientation. This allows sufficient time to adjust the station's nodal rate. The most unpredictable variable in this case will be unforeseen variations in solar activity and the resultant changes in the station's orbital decay rate. Flexibility in reboost strategy would have to be maintained to compensate for these variations.

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PROPELLANT MASS REQUIRED FOR MARS MISSION

(TO INJECT NOMINAL 176 t PAYLOAD)

C3 EFF	ΔV_{TOT} 3 IMP	TOTAL MASS ON ORBIT M_{OO} (t)	PROPELLANT MASS M_{pp} (t)	TANKAGE MASS M_{TANKG} (t) & P/S
10	3.634	449.6	246.20	27.36
15	3.853	478.0	271.76	30.20
20	4.068	508.0	298.78	33.20
25	4.278	539.7	327.36	36.37
30	4.486	573.4	357.62	39.74

h_{LEO} = 370 km

ISP = 460 sec

Tankage Factor = 0.1

PROPELLANT MASS REQUIRED FOR MARS MISSION

One purpose of this study was to determine the penalty for orbital launch missions in terms of total (fueled) mass on orbit required by a typical space station-launched manned payload bound for Mars. Since the amount of propellent required for Earth orbital departure scales linearly with injected payload mass, a standard value of 176 metric tons was assumed for the payload mass for all mission cases (from Reference 3).

As previously shown, the departure period availability, measured in days, greatly depends on the amount of energy available for injection - for higher energies, longer departure periods become available. However, some orientations of the space station's ascending node are very hard to accommodate, thereby driving the effective C3 energy requirement to very high values.

In order to assess the total mass on orbit, M_{oo} from known effective C3 requirements, a table was prepared relating these two quantities. A single stage cryogenic (LOX + LH_2) propellent departure maneuver and injection vehicle was assumed, exhibiting a specific impulse of 460 seconds, with a tank/engine mass factor of 10 percent of the propellent mass. The propellant requirements table shows the total resulting mass on orbit, including the propellent and tankage masses required for different levels of effective injection C3. The use of multiple stages would lower the total mass on orbit somewhat.

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SPACE VEHICLE DEPLOYMENT FROM SPACE STATION

CONCLUSIONS

- NO INSURMOUNTABLE TRAJECTORY BARRIERS TO PILOTED MARS MISSIONS DEPARTING FROM THE STATION.
- CAREFUL ADVANCE PLANNING NECESSARY TO MANAGE STATION NODAL REGRESSION TO BE IN ACCEPTABLE ORIENTATION ON DESIRED DEPARTURE DATE.
- CLOSELY SPACED DEPARTURES (LESS THAN 2 YEARS APART) COULD REQUIRE RELATIVELY LARGE AVERAGE STATION ALTITUDE CHANGES (~100 km) TO ADJUST REGRESSION RATE.
- MANAGEMENT OF THE NODE WOULD IMPOSE A MASS-TO-ORBIT PENALTY DUE TO HIGHER STATION ALTITUDES.
- ANY OTHER ASSEMBLY FACILITY IN LEO WOULD ENCOUNTER THE SAME CONSIDERATIONS REGARDING NODAL POSITION, DEPARTURE PERIOD DURATION AND MASS-TO-ORBIT TRADEOFFS.

CONCLUSIONS

The conclusion of this study is that for piloted Mars missions departing from the station, trajectory considerations appear to impose no insurmountable barriers. Careful attention must be paid to overall system performance, cost and risk optimization (including station operations, ground-to-station logistics, and the departing mission). Astute advance planning would permit the station to be a very advantageous assembly and departure point for manned exploration mission although closely spaced departures could present a problem in the trade-off of nodal position and mass-to-orbit.

It should be noted that any other assembly and staging site in low earth orbit would encounter essentially the same problems as the station regarding nodal regression and the trade-offs required to assure advantageous departure geometries. Nodal regression rates differ with inclination and altitude and it is conceivable that some mission-specific advantage might accrue in having, say, more mass-to-orbit capability (i.e., a lower orbit altitude) or a different inclination. However, in the absence of such mission-specific trajectory constraints, the space station appears quite capable of supporting the trajectory requirements of manned solar system exploration.

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SPACE VEHICLE DEPLOYMENT FROM SPACE STATION

CONCLUSIONS (cont.)

- DEPARTURE PERIOD DURATIONS FOR DIFFERING MISSION TYPES (e.g., E-M, E-M-E, E-V-M-E etc.) AND DIFFERENT OPPORTUNITIES SHOWS GREAT TOPOLOGICAL DIVERSITY
- DEPARTURE PERIOD DURATION IS A VERY ERRATIC FUNCTION OF SPACE STATION NODAL LONGITUDE, Ω, ON DEPARTURE DATE
- IN MOST SITUATIONS, GAPS IN DEPARTURE PERIOD DURATION EXIST IN SOME BANDS OF VALUES, ESPECIALLY AT THE LOWER EFFECTIVE INJECTION ENERGIES, C_{EFF}.
- FOR REASONABLE DEPARTURE PERIOD DURATIONS, THE RESULTING INCREASE IN ON-ORBIT PROPELLANT REQUIREMENTS FOR TRANSPLANETARY INJECTION IS MODERATE (GENERALLY BETWEEN 10 AND 25% OF DEPARTURE FROM OPTIMUM PARKING ORBIT).

CONCLUSIONS (cont.)

The departure window availability for differing mission types (e.g., Earth-Mars, Earth-Mars-Earth, Earth-Venus-Mars-Earth,) and different launch opportunities exhibits great topological diversity. To derive any generalized meaning from this diversity, one needs to consider the ''big picture'' of how such major missions would be planned. As is typical of space missions, the effective C3 capability of the departure booster will be the limiting factor in the trade-off between payload mass and launch window duration. While spacecraft designers always seem to need more mass, mission managers want the longest possible launch window. These competing demands must be reconciled with the C3 limitations of the launch vehicle. For a given launch vehicle capability, it is clear from the graphs of station departure window availability that positioning the ascending node of the station orbit in the most advantageous position can have a profound effect on the length of the departure window. For endeavors of the magnitude of manned Mars missions, this would clearly be part of the mission plan, as no mission is likely to have the luxury of sufficient C3 capability to be totally independent of station nodal alignment.

References

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