Electric Propulsion Performance from Geo-transfer to Geosynchronous Orbits

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Electric propulsion has been broadly accepted for station-keeping and final orbit insertion of Earth-orbit spacecraft. However, operational use for Low Earth Orbit (LEO) to Geosynchronous Earth Orbit (GEO) transfers has not yet occurred. While the performance gain attained from starting with a lower orbit is large, there are also increased transfer times and radiation exposure risk that has hindered the commercial advocacy for electric propulsion stages. An incremental step towards electric propulsion stages is the use of integrated solar electric propulsion systems (SEPS) for GTO to GEO transfer. General analyses of GTO to GEO transfers, from a variety of launch sites, with near-term electric propulsion options are presented. These analyses should provide interested parties the ability to rapidly assess the performance and trip times for such missions and select technical approaches consistent with selected mission constraints.

Nomenclature

F	= thrust, N
g	= acceleration due to gravity, m/sec^2
I_{SP}	= specific impulse, sec
M_o	= initial spacecraft mass, kg
M_p	= propellant mass, kg
$M_{payload}$	= final delivered mass (GEO payload with propulsion system and any added power), kg
P	= power, kW
ΔV	= change in velocity, m/sec
η	= efficiency

I. Introduction

There is a trend in spacecraft requirements towards larger and higher power systems.[1] The cost of solar array power is decreasing with time; just as the efficiency of arrays continues to increase. Figure 1 illustrates these trends in solar array efficiency, spacecraft mass, and the associated GEO power-to-mass ratios. Additionally, electric propulsion thrusters are now quite common on geostationary satellites for station-keeping. There are currently several hundred electric propulsion systems flying on over 100 spacecraft. Electric propulsion applications have also increased recently. Electric propulsion has been used for orbit topping[2] low-thrust transfers on science missions, and an increasing use of EP for near-Earth transfers is likely in the very near-term. All of these factors are favorable to electric propulsion systems, yet there is still little wide range advocacy for LEO to GEO transfers. A technology push continues to be from the technologists rather than a pull from the customers due to unacceptable transfer times or perceived risk. As long as low-thrust transfer times to GEO are within acceptable durations, the GTO-to-GEO transfer is the next low risk logical approach to leveraging the advantages of electrical propulsion. This paper provides top-level analyses of the performance of GTO-to-GEO missions using near-term EP options.

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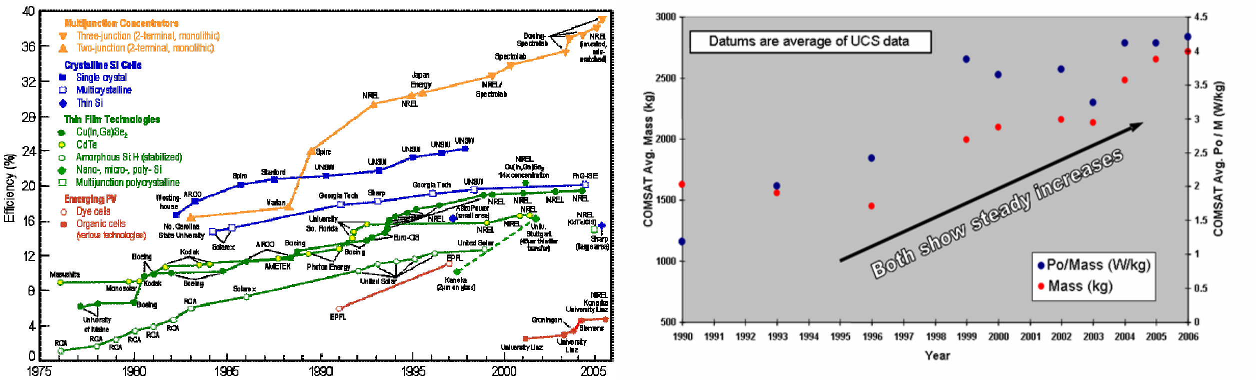

Figure 1. Solar array efficiency (left)[3] and COMSAT power and mass (right)[1] trends.

The option to start with a GTO and use a low-thrust transfer to GEO can significantly reduce the transfer time, and serve as a lower risk option for electric propulsion mission enhancement.[5] The risks associated with low starting orbits include the likelihood of impact with debris and the increased exposure radiation within the Van Allen Belts. The exposure to hazard environments can be greatly reduced by launching to an elliptical orbit such as GTO as the initial starting orbit of the low-thrust transfer. Figure 2 shows the effect of starting apogee on the duration of hazard exposure. The range begins at a stating LEO altitude of 500 km. By starting the low-thrust transfer in GTO with an apogee at GEO, the times below 10,000 and 1,200 km are reduced by more than an order of magnitude from those experienced in a LEO to GEO transfer.

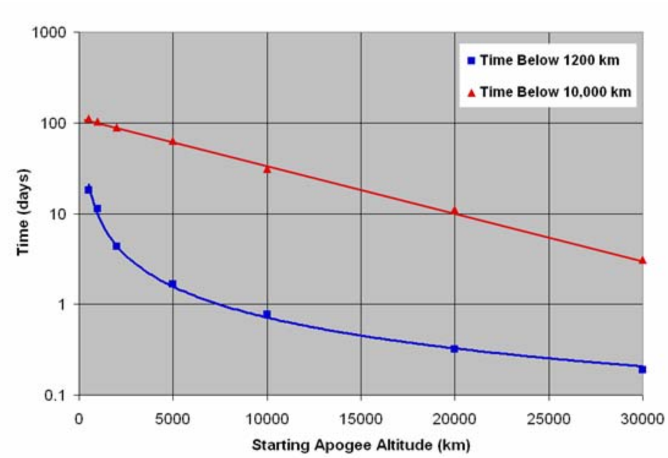

Figure 2. Hazard exposure vs. starting apogee.[4]

Electric propulsion options vary, but based on attainable specific impulse, efficiency and maturity considerations, Hall Effect thrusters (HETS) and Gridded ion engines were selected for evaluations herein. Depending on the emphasis of transfer time or performance, different customers will have a preferred propulsion system. There are several options available today suitable for low-thrust maneuvers and many more are in development or planned for the near-term. As the use of electric propulsion prevails, higher power and more capable systems will be developed. Figure 3 provides an estimate of an integrated propulsion system mass. The integrated propulsion system leverages the payload's on-board power, but the complete propulsion system is independent. The mass value shown is for a single thruster string including the thruster and associated harness, the power processing unit, xenon feed system with a high pressure assembly, low pressure assembly, and tubing; a gimbal, and miscellaneous mass required for system integration. These mass estimates are not necessarily equivalent to what is currently used in flight configurations. For example, a multiple thruster system may only need a single high pressure assembly for the system, or more significantly; a thruster pair (or spare) is often flown with a single PPU. Finally the tank mass must also be accounted for, and can be estimated as a fraction of the required propellant. For a modern composite overwrap pressure vessel, a tank fraction of 4.5% is a reasonable estimate. The propellant mass can be calculated by equation 1.

$$M_p = \left(e^{\frac{\Delta V}{g I_{sp}}} - 1\right) M_{payload} \tag{1}$$

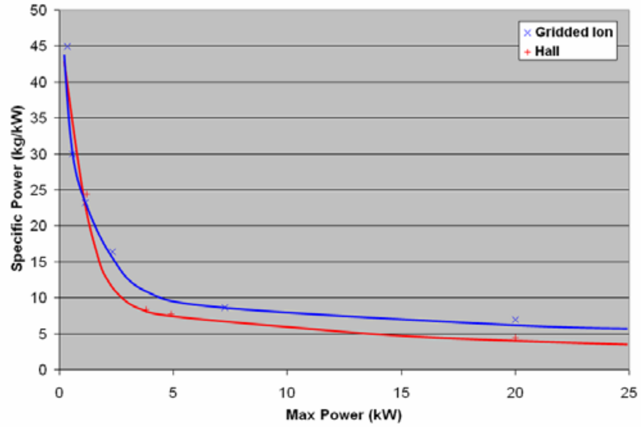

Figure 3. Specific power estimates of electric propulsion systems.

II. Analyses

Even though a significant number of electric propulsion systems are flying successfully, it is still viewed by many to be an expensive and high risk technology. The real challenge to implementing electric propulsion stages is, and will probably always be; the long transfer times. The intent of the results is to provide estimates of the transfer time and ΔV required for GTO to GEO transfers. Analyses were conducted independent from the propulsion system. A large range of parametric trades were conducted to encompass the anticipated performance of existing or near-term propulsion system developments. The baseline system metrics are provided in table 1, and were used unless otherwise stated. The baseline case is a GTO-to-GEO transfer from Kennedy Space Center (KSC) with a 2000s Hall thruster with a thrust to power of 60 mN/kW. Also, unlike figure 1 which represents the specific power at GEO, the specific power used for analyses is based on the GTO start mass.

Table 1. Baseline metrics for optimization.

Starting Perigee Altitude (km)	**185**
Starting Apogee Altitude (km)	**35,786**
Starting Inclination (deg)	**28.5**
Ending Circular Altitude (km)	**35,786**
Ending Inclination (deg)	**0**
Start Mass (kg)	**6600**
Array Power (kW)	**19.8**
Specific Impulse (s)	**2000**
Thrust-to-Power (mN/kW)	**60**
Spacecraft GTO Specific Power (W/kg)	**3**

Another subtlety of using electric propulsion is that more specific impulse is not always better as is common in conventional chemical propulsion systems. For a given input power, an engine can trade thrust for specific impulse as shown in equation 2.

$$FI_{sp} = \frac{2\eta P}{g} \tag{2}$$

While there is a trade between thrust and specific impulse, one should also account for the dependency of efficiency on specific impulse. Figure 4 provides notional thruster curves for efficiency as a function of specific impulse for gridded ion and Hall thrusters. The performance data is from a collection of NASA thrusters of varying development levels, operating power ranges, and expected lifetimes. The solid data points represent Hall thrusters and the open data points represent gridded ion thrusters. The curve fit lines represent generic thruster efficiencies as a function of specific impulse. Hall thrusters are generalized with the curve on the left, with the curve for higher specific impulses representing gridded ion engines. The vertical bars represent the efficiency range considered for this analysis. Clearly there are thrusters that operate above the generalized curves, but all are captured within the range of thrust-to-powers evaluated.

Another consideration when choosing a propulsion system is that increase in specific impulse does not often yield the full gain one might anticipate using Tsiolkovsky's rocket equation. The transfer ΔV is also a function of the spacecraft thrust, or more accurately, acceleration. Using a higher thrust propulsion system can reduce the ΔV requirement by either minimizing losses or by operating with more efficient maneuvers. Examples include gravity losses, raising perigee at apogee, or plane changes at apogee.

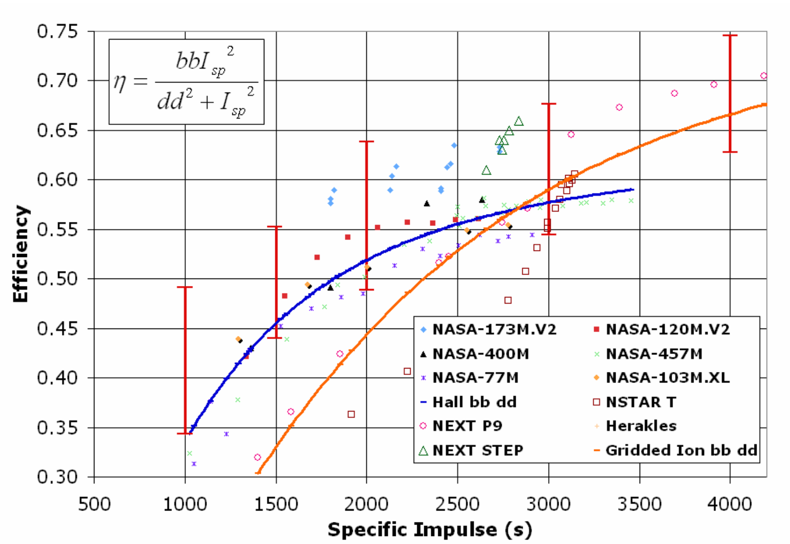

Figure 4. Efficiency versus specific impulse for Hall and gridded ion thrusters.

The transfer time and ΔV for each low-thrust spiral phase has been calculated using SEPSPOT. SEPSPOT is a Glenn Research Center variation of Solar Electric Control Knob Setting Program by Optimal Trajectories (SECKSPOT[6]). The program is a low-thrust trajectory optimizer that calculates the minimum time optimal geocentric trajectory with attitude constraints. The attitude constraint causes power to become a function of thrust and sun direction, and the time optimal thrust direction is a complex function of primer vector direction. The code has capabilities to determine array degradation through an internal radiation model and account for startup delays after emerging from shadow, however; the degradation and delay factors are not current with state-of-the-art technology. Because of the advancement in array technologies and xenon thruster ignition delays, those features of the code have been turned off. A user may choose to add sufficient Beginning of Life (BOL) array margin to account for array degradation, however; GTO-to-GEO degradation should be minimal with a worst case estimate of only 10%. The code also has the ability to determine occultation effects, however; because of the occultation delay dependency on acceleration, launch date and inclination, and difficulty in achieving convergence; this feature has also been turned off. The analyses below do not include occultation delays and should be accounted for as discussed later.

III. Results and Discussion

As the results are interpreted, it is important to note that the transfer time is not dependent on the mass, but rather the vehicle acceleration. The vehicle acceleration is determined by the ratio of thrust-to-mass. Therefore, the results are plotted as a function of the ratio between available spacecraft power and spacecraft initial mass. The baseline trajectory is based on a launch from KSC without occultation or array degradation, and additional plots are provided for correction factors. Figures 5-9 represent the transfer times and ΔV requirements for the time optimal low thrust transfers. The various plots represent the SOA and near-term performance of gridded-ion and Hall thrusters.

As figure 1 shows the average W/kg is above four with an increasing trend. For those preferring shorter transfer time over performance, 1000s specific impulse is possible; through efficiencies currently only allow approximately 70 mN/kW with corresponding transfer times just under 100 days. Because the thruster can operate above the knee in the efficiency curve at 2000s, the performance gain due to the higher specific impulse is possible without a significant increase in transfer times. On the other hand, the performance increase from 2000s to 4000s specific impulse comes at a significant transfer time penalty. Also, the ΔV increase from higher thruster to higher specific impulse is approximately 250 m/s, which partially diminishes the benefit of operate at higher specific impulse.

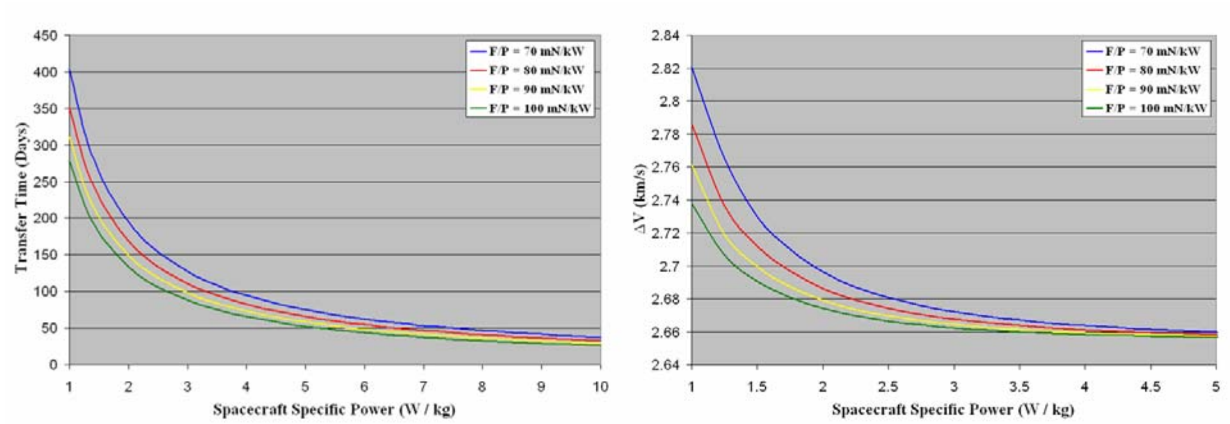

Figure 5. KSC transfer time (left) and ΔV (right) at 1000s.

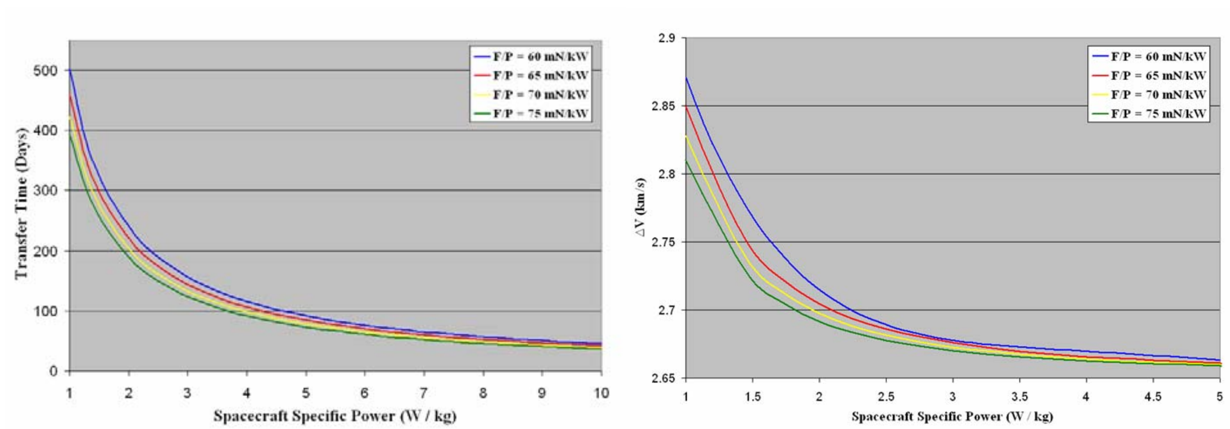

Figure 6. KSC transfer time (left) and ΔV (right) at 1500s.

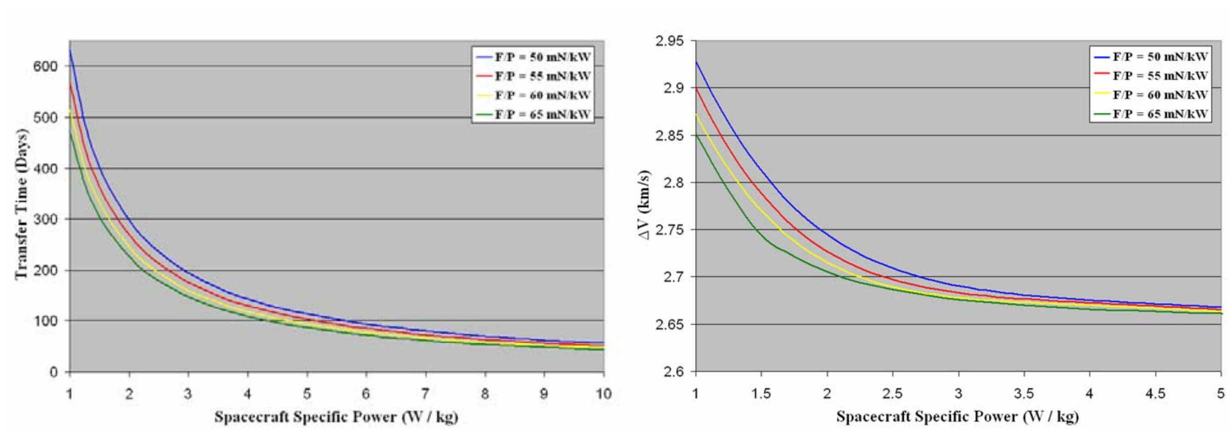

Figure 7. KSC transfer time (left) and ΔV (right) at 2000s.

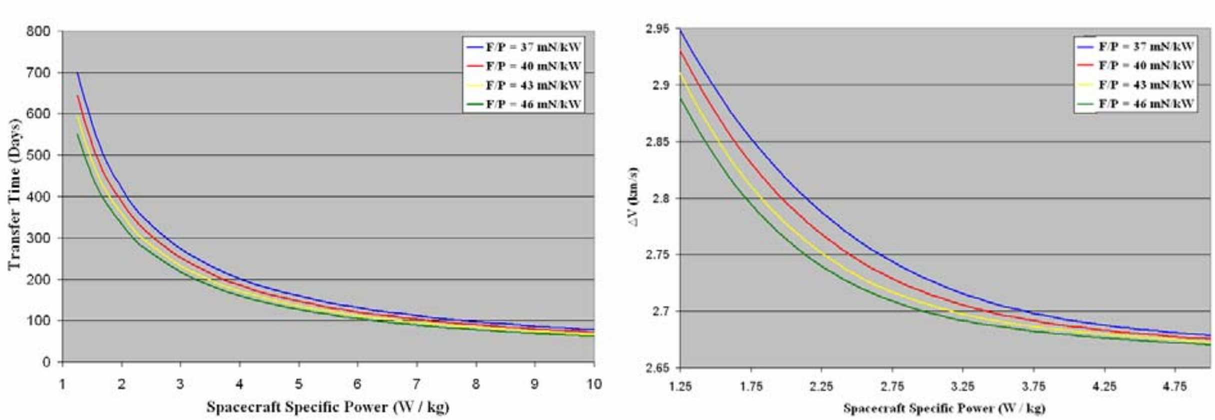

Figure 8. KSC transfer time (left) and ΔV (right) at 3000s.

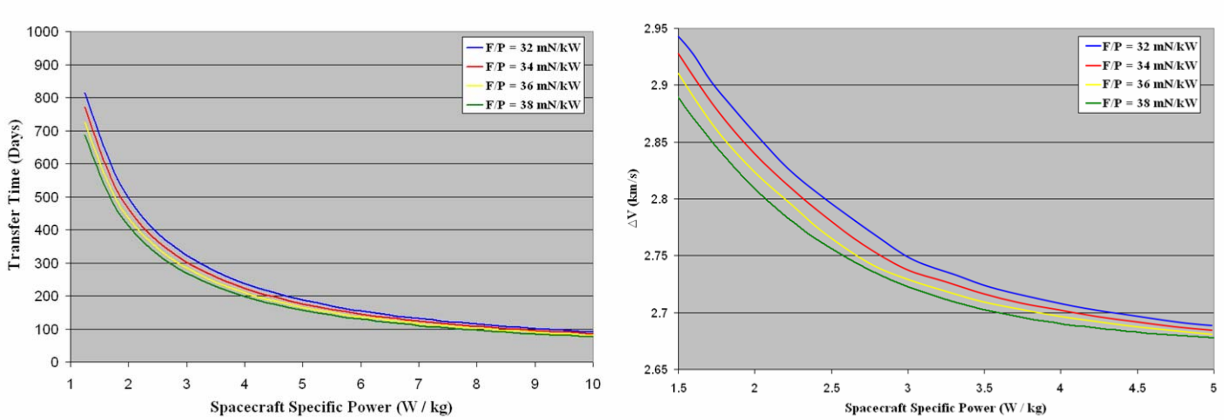

Figure 9. KSC transfer time (left) and ΔV (right) at 4000s.

The launch inclination also plays a major role in the determination of transfer time and ΔV requirements. The inclination change penalty for using low-thrust propulsion is not as significant as chemical propulsion, but can still increase the total transfer time and ΔV significantly. Figure 10 shows the percent correction factor for various launch inclinations. The plane change penalty from low inclinations to equatorial is minimal, but the penalty increases rapidly above 10^{o}.

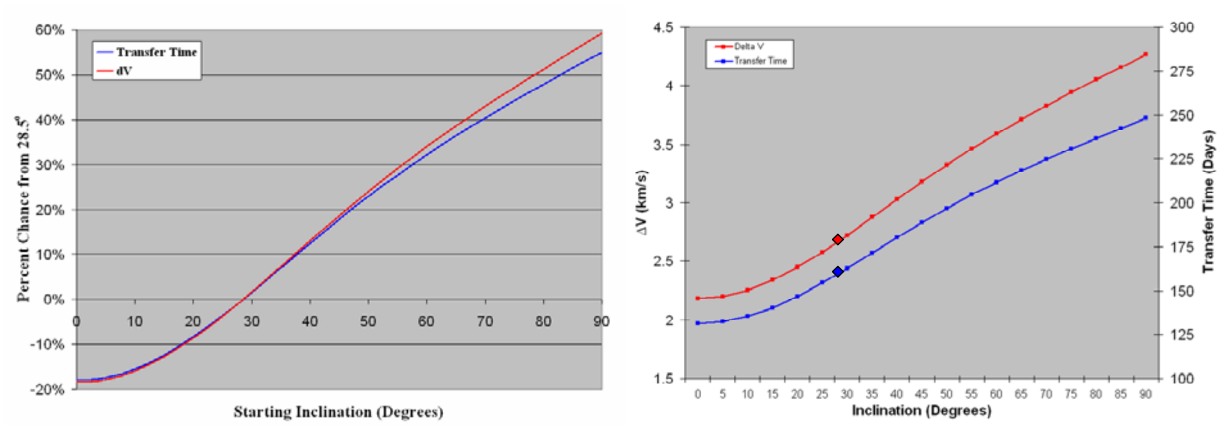

Figure 10. Inclination effects on transfer times and ΔV requirement.

While the figures above can be used for estimated low-thrust transfers from any launch site, it should be noted that some launch services have various GTO orbits prior to GEO insertion. For example, launching a spacecraft to GEO from Baikonur is completed in stages. The first GTO-to-GEO EP transfer could begin immediately after the slightly super-synchronous launch with a large plane change penalty, or begin after the following chemical stage places the spacecraft into a higher perigee orbit and performs a significant portion of the plane change. Various launch options listed in Table 2 are illustrated in Figure 11.

Table 2. Starting orbit parameters from various launch sites.

	Baikonur A	Baikonur B	Kennedy SC	Kourou	Equatorial
Starting Altitude (km)	200	4,100	185	560	185
Starting Altitude (km)	35,950	35,786	35,786	37,786	35,786
Starting Inclination (deg)	48.6^o	23.2^o	28.5^o	7^o	0^o

Figure 11 shows the baseline performance from a few of the major launch sites. The only launch site with a more challenging starting orbit from KSC would be the initial Baikonur orbit because of the starting inclination. Of course, the challenging launch requirements also provide the most significant benefit of an electric propulsion transfer by leveraging the high performance EP system the greatest portion of the insertion maneuver.

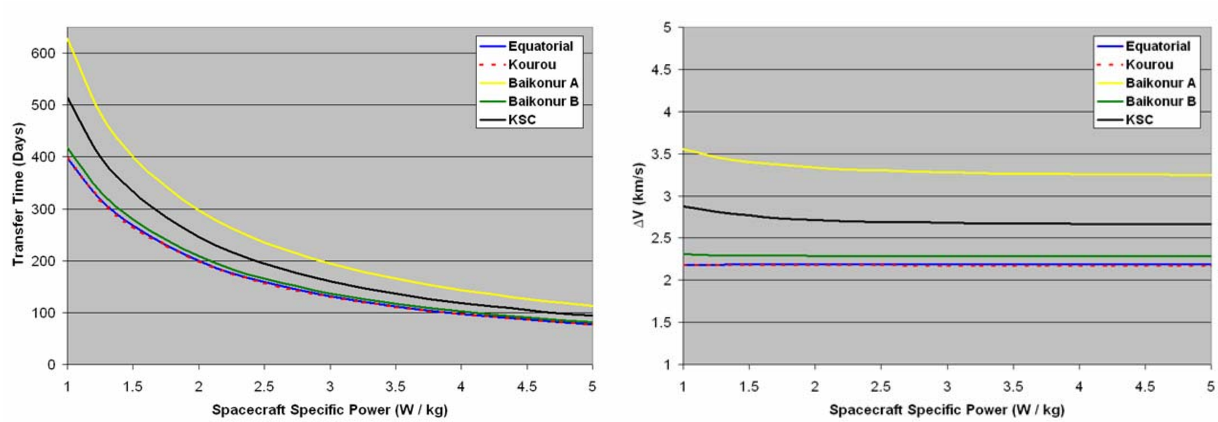

Figure 11. Launch site effect on transfer time and ΔV requirement.

Another influence on transfer times and ΔV requirements is the occultation effects during the transfer. Figure 12 is an illustration of the occultation shadow with the line of nodes aligned with the Sun line, the Sun and spring equinox, and a GTO at 28.5°. Unlike low-thrust transfers starting LEO, GTO-to-GEO occultation penalties are only minimal and sometimes zero. The way to get occultation to zero is to launch when the Sun is leaving equinox and get to GEO before it returns to equinox (in 182 days). Actually, one needs to complete GEO insertion approximately 2 weeks before the Sun returns to equinox, so the trip time needs to be about 160 days or less. Trip times less than 160 days are quite feasible with a 2000s thruster and a spacecraft specific mass of 3 W/kg. The problem with this is that it restricts launches to spring and fall seasons.

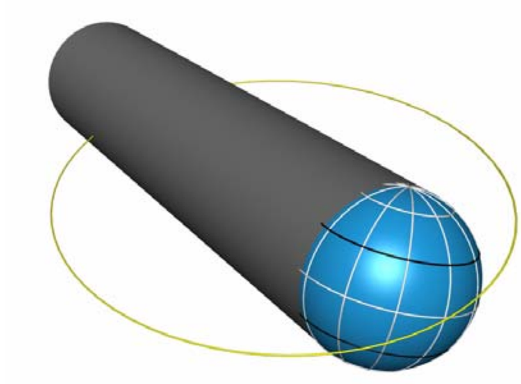

Figure 12. Hazard exposure vs. starting apogee.

Without restrictions on launch days, the penalty can still be minimized. Occultation is affected by departure time (longitude of nodes relative to the Sun-Earth line), departure date (location of the Sun, i.e. solstice, equinox, somewhere in between), and trip time. Figure 13 shows occultation effects for the Sun at both spring equinox and solstice at various longitudes of the node. The worst case is with the GTO starting orbit line of nodes aligned with the Sun line. In that case, occultation occurs at apogee and is maximum duration. This worst case is about 2.4 hours, 21.33% of the orbital period, but is easily avoided. All one must do is launch at a time of day when the initial apogee is sunlit. Launch time of day does not affect GEO apogee longitude with respect to the Earth's surface, but it does affect GEO apogee location relative to the Sun line. The worst case occultation duration is about twice that for normal GEO operations, and could affect spacecraft battery requirements. The worst-case total occultation time as a fraction of total transfer time is about 7%, and a typical value is only about 2% for launches at equinox, and range from 3.5% – 6.2% of total transfer time for the summer solstice launch.

Other occultation effects are possible loss of attitude control, assuming this is provided by the electric propulsion system, and alteration of the transfer trajectory, including slight changes in the transfer delta V. Assuming the spacecraft communications payload is not operating at full power, there should be enough battery power to operate the electric thrusters at a high enough level (about 10% rated power is usually enough) to maintain attitude control. The transfer trajectory is altered because thrusting at a reduced level during portions of an orbit changes orbit parameters. This is predictable and is a minor effect, easily mitigated by commands during the transfer.

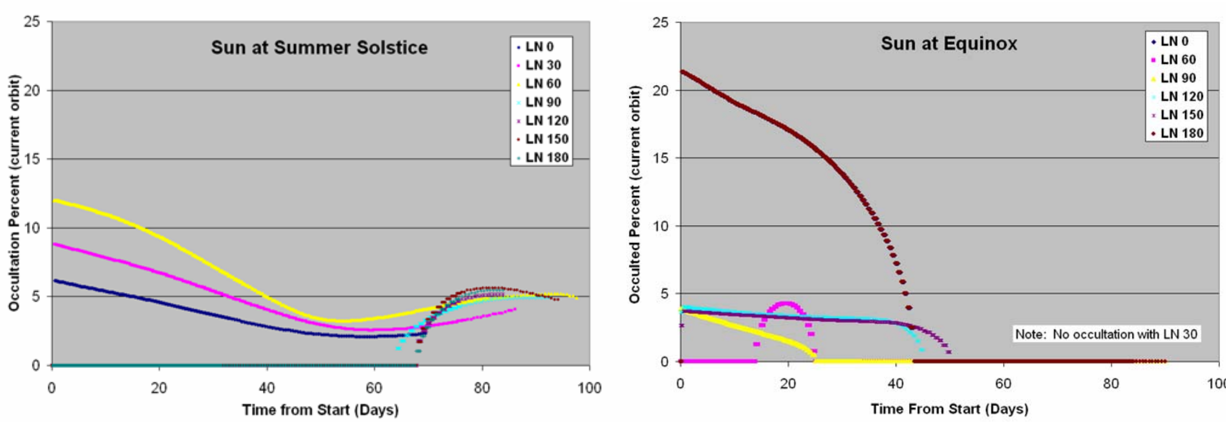

Figure 13. Occultation variations for launch at solstice (left) and equinox (right).

GTO is of course not the only starting orbit that can be used. Because launch vehicles can deliver payloads to highly elliptical orbits more efficiently than high circular orbits, elliptical starting orbits can be used for relatively low to super-synchronous orbits. For those willing to trade trip time for additional performance, a lower starting apogee can be used. For those who prefer to decrease the required transfer time, a super-synchronous starting orbit can be used without a large decrease in performance. Figure 14 shows the trade between transfer time and performance for various starting orbits. The figures below represent the baseline case using the Atlas 551 launch vehicle.[7]

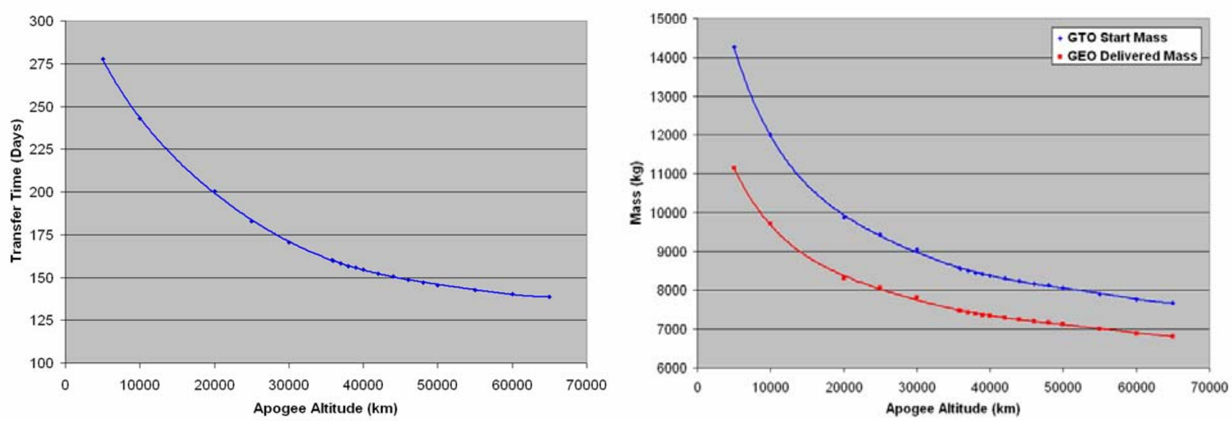

Figure 14. Transfer time (left) and performance (right) versus apogee altitude.

The integrated EP system from GTO-to-GEO clearly shows reasonable transfer times are possible with the trend of increasing power-to-mass of modern spacecraft. Another incremental step closer to high performance EP stages without added risk to the user could be found by the addition of solar array power only for the purpose of shorter trip times. The XM satellites, for example; have 18 kW of on-board power for the spacecraft. Using only the onboard power, the GTO-to-GEO transfer time with associated penalties is less than 100 days from KSC. However, figure 15 shows the reduced transfer times that can be expected with supplemental power. A 50% increase is on-board power can reduce the transfer time to only 60 days. As array costs, mass and stowage volume are reduced, very rapid low-thrust transfers to GEO using supplemental power may become attractive.

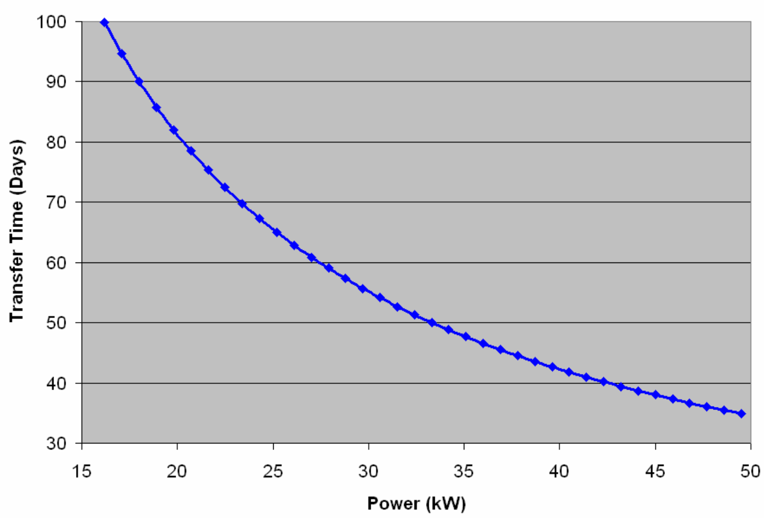

Figure 15. XM satellite GTO-to-GEO transfer with added power.

IV. Example Case and Conclusion

Assume one decides build a launch vehicle company with a 5,000 GTO mass capability, and would like to move launch operations from an inclination of 9° in the south Pacific to South Padre Island with an inclination of 26.5°. In particular, the launch provider has interest in XM radio, and would like to deliver XM radio satellites to GEO at a substantially lower cost. XM satellites have approximately 18 kW of GEO B.O.L. power, and a dry mass of approximately 3000 kg. To do so chemically, the GTO to GEO ΔV would be approximately 1.52 km/s from 9^o and 1.79 km/s from 26.5^o. The chemical engine has a specific impulse of 320s. After calculating the propellant mass by equation 1, the required GTO throw mass would be 4,870 kg and 5,310 kg respectively.

Because the required GTO mass is too high for the current launch vehicle, rather than building a larger vehicle; the addition of an electric propulsion system could be used that operates off the onboard spacecraft power. First a mass penalty of 12 kg to the solar array must be added to ensure the degraded power with the worst case array degradation would still meet the payload power requirements. Next, one can add four 4.5 kW thrusters, 55 – 60 mN / kW, that operate at 2000s specific impulse. From figure 4, the specific mass of each system would be approximately 7.8 kg/kW for a total of 140 kg. Assuming none of the original chemical apogee hardware is removed or modified, the new payload mass is 3,152. The spacecraft specific power is now 16,200 / 3,152 or just under 5.2 W/kg. From figure 7 and correcting for inclination with figure 10, the low thrust ΔVs will be approximately 2.2 km/s and 2.6 km/s with transfer times on the order of 85 and 100 days respectively. From the given ΔVs, the launch vehicle throw capability needs to only be 3,525 and 3,600 kg respectively. For comparison, the specific case for South Padre Island has been optimized to show the required ΔV is 2.6 km/s with a transfer time of 102 days.

Finally, knowing there is considerable margin in mass and without consideration for cost; one could even decide to add supplemental power for a faster transfer. Assuming 6 kg/kW for the array and adding another 12 kW or 10.8 kW of usable power for a total of 27 kW of power for six 4.5 kW thrusters, we have a total added array mass of 84 kg and a new thruster mass of 211 kg. The new payload mass is therefore 3,295 kg with a specific power of 8.2 W/kg. The required GTO throw mass is still only 3,775 kg, and one can be listening to their XM radio in only two months after launch. Again, the specific case optimized for a 3,775 kg GTO start mass provided a GEO delivered mass of 3,300 kg in 64 days.

While the intent of the paper is only to serve as a reference, the primary conclusion is that low-thrust transfer times and ΔV requirements can be accurately estimated using the spacecraft specific power, electric propulsion system performance, and using corrections for launch site inclinations and possible occultation effects for high level analyses and trades.

Acknowledgments

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