HIGH ALTITUDE ORBIT RAISING
WITH ON-BOARD ELECTRIC POWER

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ABSTRACT

Classical LEO/GEO transfers with electric propulsion have been studied in detail, generally with favorable results. For one-way missions, excessive cost of power for short duration transfers and excessive cost of idle capital for long transfer times detract from the gross mass benefits. This dilemma has been resolved in principle by two approaches. When expendable power and electric propulsion systems are used, the strategy is to design the S/C so that one can step down to a smaller launch vehicle, whereupon the launch cost savings are usually more than enough to absorb the cost of both the extra hardware and the maneuver time. This approach generally favors EP devices with lower exhaust velocities. The alternative is to invoke the use of a reusable EP ferry which makes multiple round trips between various LEO's and higher orbits. This strategy permits the cost-effective use of a very large, high-power transfer vehicle, and generally favors the use of EP devices with higher exhaust velocities. Despite the large mass and cost benefits expected through EP orbit raising, there has been considerable reluctance, over several decades, to tackle a task of this magnitude. Perhaps a modest approach would be more feasible early on. The present study deals with a much smaller velocity increment, about 500-2000 m/s. Hardware and power mass penalties are reduced to minimal levels by adapting the power and propulsion systems which are already on board. High altitude orbit raising (HALOR) favors the low end of the EP specific impulse range, in the sense that the payload delivered to GEO tends to be larger, for any given transfer time, when the power/thrust is small. For starters, the study is restricted to orbit raising (OR) only, i.e., no plane change. The EP devices included are the xenon ion thrusters with and without grids, the hydrazine/ammonia thermal arcjet, the hydrogen arcjet, and the hydrogen resistojet. Using the figure of merit of BOL mass in GEO for any given transfer time, the rankings of the EP devices for the OR alone are: (1) H2 resistojet, and (2) H2 arcjet. The SPT gridless ion thruster and the derated XIPS-30 are about tied for a poor third, with the NH3 arcjet and the XIPS-25 close behind. When these results are factored in with the benefits available from other on-board propulsion tasks, the ranking is less obvious. Perhaps even a hybrid EP system will prove to be the best arrangement.
1.0 INTRODUCTION

Classical orbit raising via electric propulsion (EP), in the range of velocity increments (\(u\)) from about 4000-12000 m/s, has been studied in detail. The results to date may be roughly summarized as follows, using a Canaveral launch to 500 km circular LEO:

To compete successfully with high energy chemical propulsion (\(v = 4400\) m/s, \(u (\text{delta V}) = 4190\) m/s), electric propulsion exhaust velocities in excess of about 8000 m/s must be employed to counter the larger \(u\) for EP of about 5860 m/s and the extra mass of the EPS. At the other end of the range of practical exhaust velocities, anything in excess of about 30,000 m/s has too high a power/thrust ratio to allow the completion of the mission with tolerable power levels and transfer times.

Within this range, EP devices with lower propellant exhaust velocities and high efficiency yield substantial payload bonuses in fairly short transfer times. EP devices with high exhaust velocities deliver even greater payloads, but with the penalty of either excessive power or considerably longer trip times (1). A typical illustration of this is shown in Figure 1. In the range of 8000 - 9000 m/s exhaust velocity, thermal devices, both nuclear and solar, will outperform EP devices for one way trips (2), (3), owing to the much lower specific mass of the power supply and PCU. One exception to this may be when very high efficiency, light weight solar arrays become available.

When the orbit raising is accomplished with a dedicated power and propulsion system, and the benefits are measured in terms of launch cost savings, it is critical to specify whether there is a step down to a smaller launch vehicle. If there is no change in the launch vehicle (i.e., just off-loading or payload sharing), the power required for reasonably short trip times is usually much greater than the satellite operating power, and the cost of the excess power is prohibitive. Conversely, for long trip times the cost of idle capital is excessive. This dilemma has been resolved in principle by recourse to a reusable EP ferry which makes multiple round trips between various low Earth orbits (LEO's) and higher orbits. The concept has been explored with favorable results for both solar- (4) and nuclear-powered (5) transfer vehicles. For values of \(u\) ranging up to 12000 m/s for the round trip, the multiple re-use of the EP transfer vehicle allows the use of very large power levels in a cost-effective manner, and the studies therefore show a distinct advantage for EP devices at the higher end of the Isp spectrum. When the launch cost reduction is secured by accomplishing the mission with a smaller launch vehicle, then the launch cost reduction is usually large enough to absorb the cost of a dedicated power and propulsion system, as shown in (6). This latter strategy favors the EP devices at the lower end of the Isp range.
No matter what the details of the approach, it is clear that a substantial OR maneuver results in large mass benefits and consequent cost reduction. Despite this, the full LEO/GEO OR maneuver being a massive undertaking, there has been considerable reluctance to tackling a task of this size, given the past and current budget restraints. Perhaps it is time to look at a more modest approach.

2.0 HIGH ALTITUDE ORBIT RAISING (HALOR).

It is well to remember that chemical and electric propulsion should always be treated as complementary partners. In past OR studies the EP strategy has usually been to preempt as much of the velocity increment as possible, i.e., the full LEO/GEO trip with maximum plane change. This of course results in the highest mass benefits, but at the cost of mounting a major project, and the inclusion of many complications and problems, such as shadowing, drag, solar cell damage, excessive trip time, etc. HALOR goes to the opposite extreme, that is, settles for a modest delta V and smaller benefits, but with much less risk of failure, and without the huge investment and many of the problems associated with the full LEO/GEO maneuver. In addition, since the HALOR approach can be expanded gradually to something like the full OR mission, as the experimental data base grows, it provides a way of choosing the ideal CP to EP handover point, with all the tradeoff factors taken into consideration.

The present study deals with a small range of u, about 500-2000 m/s, starting from appropriately high altitude circular orbits. Two INTELSAT supported studies have already been carried out along these lines, one with Ford Aerospace (now LORAL) and the second with MATRA-MARCONI. Both of these studies were limited in scope to either the XIPS-13 or the HAJ, assumed to be already on board for NSSK, and both assumed a low value for on-board power (3 kW). We will expand the study to include the Russian SPT-100 thruster, the H2 arcjet, and the H2 resistojet. We will also assume that a small portion of the waste heat from the EP systems will be available for minimum satellite temperature maintenance, thereby releasing additional electric power for propulsion (total: 3.2 kW); this is done to provide a better power match with the candidate thruster systems, not for the sake of the slight increase in power. For starters, we choose to limit the study to orbit raising alone, i.e., no plane change, and to use only the onboard power. Since the power is indeed already on board, and costs nothing to use, nuclear- and solar-thermal methods cannot compete with EP for this maneuver. The EP systems which are already assumed to be on board will be modified where necessary to handle the available power. This severely power-limited approach favors the EP devices which have low power to thrust ratios, i.e., those which operate in the lower Isp range. Countering this is the increased mass benefit which is secured by applying the higher Isp devices to NSSK. It may be that the best approach overall is a hybrid EP system, for
example, the H2RJ for orbit raising and the ion thruster for NSSK; or the H2RJ for OR and the hydrazine AJ for NSSK.

2.1 MANEUVERS WITH CHEMICAL PROPULSION

The Ariane 44L has been chosen as the exemplary launch vehicle, but others could have been used to give substantially the same results. The launch sequence is assumed to be simply:

1. Launch to 200 km circular orbit, followed by erection of a suitable elliptical orbit with a chosen apogee short of geosynchronous (cryogenic propellant, ex. vel. = 4400).

2. Circularization and plane change (7 degrees) to the equatorial plane at apogee altitude (ex. vel. = 3020).

3. Handover to the EPS for OR to GEO.

The mass injection capabilities of Ariane 44L to various elliptical orbits are given in Table 1 (7), with corresponding circular orbit masses which are calculated by simple orbital mechanics equations. The equation for circularization with plane change is:

\[ u^2 = Vr^2 + Vap^2 - 2 Vr Vap \cos \theta \]  

\[ (Vr)^2 = \frac{H}{r} \]  

\[ (Vap)^2 = \frac{2H}{r_{ap}^2} - \frac{H}{a} \]  

<table>
<thead>
<tr>
<th>Apogee Rad, km</th>
<th>TO Mass w/o adap kg</th>
<th>Vr km/s</th>
<th>Vap km/s</th>
<th>u cps km/s</th>
<th>Mr kg</th>
</tr>
</thead>
<tbody>
<tr>
<td>16378</td>
<td>6186</td>
<td>4.933</td>
<td>3.735</td>
<td>1.308</td>
<td>4009</td>
</tr>
<tr>
<td>21378</td>
<td>5497</td>
<td>4.318</td>
<td>2.962</td>
<td>1.425</td>
<td>3428</td>
</tr>
<tr>
<td>26378</td>
<td>5013</td>
<td>3.887</td>
<td>2.456</td>
<td>1.480</td>
<td>3070</td>
</tr>
<tr>
<td>31378</td>
<td>4656</td>
<td>3.564</td>
<td>2.098</td>
<td>1.504</td>
<td>2828</td>
</tr>
<tr>
<td>36378</td>
<td>4399</td>
<td>3.310</td>
<td>1.832</td>
<td>1.508</td>
<td>2669</td>
</tr>
<tr>
<td>GEO: 42164</td>
<td>4140</td>
<td>3.075</td>
<td>1.598</td>
<td>1.502</td>
<td>2517</td>
</tr>
</tbody>
</table>

n.b.: apparently there is little or no penalty in u for circularization and plane change above 21000 km.

2.2 ELECTRIC PROPULSION MANEUVER

The electric propulsion portion of the orbit raising is a straightforward spiral expansion from the handoff altitude to GEO. The velocity increments are merely the differences in
circular orbital velocity between the initial and final altitudes, given by:

\[(V_r)^2 = \frac{\mu}{r}; \quad \mu = 3.986 \times 10^5 \text{ km}^3/\text{s}^2\] (4)

For the initial orbits given in Table 1, the velocity increments \((u)\) for the EP maneuver are, in km/s:

<table>
<thead>
<tr>
<th>Ro:</th>
<th>16378</th>
<th>21378</th>
<th>25378</th>
<th>30378</th>
<th>35378</th>
<th>42164</th>
</tr>
</thead>
<tbody>
<tr>
<td>(u: )</td>
<td>1.858</td>
<td>1.243</td>
<td>0.812</td>
<td>0.489</td>
<td>0.235</td>
<td>0.0</td>
</tr>
</tbody>
</table>

If the identical EPS's were used for both OR and NSSK, we could assume complete use of the power available, and calculate the BOL masses straightaway, subtracting the propellant tank mass fraction owing to the OR only. We will go through this exercise to generate an understanding of the tradeoffs among the candidate EPS's, and later make a more precise estimate of mass and cost savings for selected potential combinations of EPS's for the OR and NSSK missions combined.

2.3 EPS CHARACTERISTICS

For the OR maneuver, about 3.2 kW of power is assumed to be available from the S/C, which is modelled along INTELSAT 7 lines. The various electric thrusters were assumed to operate at 1600 W to match this power level, and operating characteristics were calculated and assigned to each EPS. The small XIPS is included for NSSK only. Tables 2 and 3 summarize the input values.

<table>
<thead>
<tr>
<th>EPS Device</th>
<th>Input Power</th>
<th>Thrust mN</th>
<th>(m) (\text{mg/s})</th>
<th>(v) (\text{m/s})</th>
<th>(P/T) (\text{W/mN})</th>
<th>Effic</th>
</tr>
</thead>
<tbody>
<tr>
<td>XIPS-13</td>
<td>500</td>
<td>18</td>
<td>0.7</td>
<td>25300</td>
<td>27.8</td>
<td>0.45</td>
</tr>
<tr>
<td>XIPS-25</td>
<td>1600</td>
<td>70</td>
<td>2.5</td>
<td>27500</td>
<td>22.8</td>
<td>0.61</td>
</tr>
<tr>
<td>XIPS-30</td>
<td>1600</td>
<td>80</td>
<td>3.5</td>
<td>22800</td>
<td>20.0</td>
<td>0.57</td>
</tr>
<tr>
<td>SPT-100</td>
<td>1600</td>
<td>86</td>
<td>5.2</td>
<td>16600</td>
<td>18.6</td>
<td>0.44</td>
</tr>
<tr>
<td>H2AJ</td>
<td>1600</td>
<td>176</td>
<td>20.0</td>
<td>9000</td>
<td>9.1</td>
<td>0.48</td>
</tr>
<tr>
<td>H2RJ</td>
<td>1600</td>
<td>300</td>
<td>37.0</td>
<td>8200</td>
<td>5.3</td>
<td>0.76</td>
</tr>
<tr>
<td>NH3AJ</td>
<td>1600</td>
<td>160</td>
<td>23.0</td>
<td>7000</td>
<td>10.0</td>
<td>0.35</td>
</tr>
</tbody>
</table>

In Table 2, note the unusual combination of high efficiency and low P/T ratio of the H2RJ. The importance of this will be seen later. In the H2AJ this advantage persists to some extent. If both thrusters had the same efficiency, the higher exhaust velocity of the H2AJ by itself would generate a P/T of only 5.8, so most of the difference relative to the RJ must be ascribed to the lower efficiency of the AJ. This cannot be ascribed to the low degree of ionization of the AJ plume, but probably mainly to the variable exhaust velocity of the H2AJ. Thus, the RJ could
probably be improved by a moderate increase in the propellant
temperature, and the AJ by more thorough mixing prior to
expansion.

Table 3. Component Masses for Candidate EPS's

<table>
<thead>
<tr>
<th>Device</th>
<th>Thruster</th>
<th>PCU</th>
<th>GIM</th>
<th>PMU</th>
<th>STRU</th>
<th>HARN</th>
<th>MISC</th>
</tr>
</thead>
<tbody>
<tr>
<td>XIPS-13</td>
<td>5.0</td>
<td>6.8</td>
<td>3.0</td>
<td>3.4</td>
<td>7.0</td>
<td>1.1</td>
<td>20.0</td>
</tr>
<tr>
<td>XIPS-25</td>
<td>10.5</td>
<td>11.0</td>
<td>3.3</td>
<td>3.4</td>
<td>10.0</td>
<td>2.0</td>
<td>22.0</td>
</tr>
<tr>
<td>XIPS-30</td>
<td>9.5</td>
<td>12.0</td>
<td>3.3</td>
<td>7.9</td>
<td>18.0</td>
<td>2.2</td>
<td>23.0</td>
</tr>
<tr>
<td>SPT-100</td>
<td>1.5</td>
<td>11.0</td>
<td>3.3</td>
<td>4.0</td>
<td>9.0</td>
<td>2.5</td>
<td>21.0</td>
</tr>
<tr>
<td>H2AJ</td>
<td>1.5</td>
<td>5.0</td>
<td>3.0</td>
<td>---</td>
<td>9.0</td>
<td>2.5</td>
<td>21.0</td>
</tr>
<tr>
<td>H2RJ</td>
<td>1.5</td>
<td>4.0</td>
<td>3.0</td>
<td>---</td>
<td>9.0</td>
<td>2.5</td>
<td>21.0</td>
</tr>
<tr>
<td>NH3AJ</td>
<td>1.5</td>
<td>5.0</td>
<td>2.2</td>
<td>4.0</td>
<td>9.0</td>
<td>2.5</td>
<td>21.0</td>
</tr>
</tbody>
</table>

In Table 3, the XIPS-13 and -25 thruster masses include part
of the structure; hence the apparent discrepancies in the columns
listing thruster and structure masses. For the hydrogen based
devices, values for the mass of the complete propellant storage
and feed system are available, and will be described later.

The uncorrected BOL mass in GEO is given by:

\[ M_{bol} = Mr \left[ \exp \left( -\frac{u}{v} \right) - \frac{Mt}{Mp} \left( 1 - \exp \left( -\frac{u}{v} \right) \right) \right] \]  \hspace{2cm} (5)

neglecting any extra EPS mass components except the tank fraction
assigned to the OR portion of the mission. The transfer time in
30 day months to GEO is:

\[ T = \frac{Mr \left( 1 - \exp(-\frac{u}{v}) \right)}{\left( 2.59 \times 10^6 \times F/v \right)} \]  \hspace{2cm} (6)

Propellant tank fractions (Mt/Mp) for xenon and hydrogen
have recently been reported. Beattie, Robson, and Williams (8)
report a value of 0.12 for xenon when stored as a compressed
supercritical gas. This value is for the tank alone. The other
parts of the PMU are covered in Table 3. Miller and Cady (9)
give a value of 0.15 for about 1500 kg of liquid hydrogen, which
includes tank, insulation, supports, plumbing, and PMU. Since
our requirement for liquid hydrogen is less than half of this, we
will assume a slightly higher value of 0.17 for the entire H2
PMU. The tank fraction assigned in the case of ammonia is 0.09.

2.4 RESULTS

Uncorrected BOL masses and transit times for the various
EPS's are given below in tabular form, and plotted in Figure 2.
The clear superiority of the H2RJ for this mission is evident,
despite the low exhaust velocity. The H2AJ comes in a distant
second, despite its higher exhaust velocity, owing to its lower
level of efficiency. The other EPS's are bunched too closely to
assign further rankings without a more detailed look at the combined OR and NSSK mission, examples of which will be described later. For these latter examples, BOL values, corrected for extra mass assigned to orbit raising, are given in parentheses below.

<table>
<thead>
<tr>
<th></th>
<th>XIPS-25</th>
<th>XIPS-30</th>
<th>SPT-100</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Mbol</td>
<td>T. mo</td>
<td>Mbol</td>
</tr>
<tr>
<td>3717</td>
<td>19.8</td>
<td></td>
<td>3657</td>
</tr>
<tr>
<td>3259</td>
<td>11.4</td>
<td></td>
<td>3224</td>
</tr>
<tr>
<td>2970</td>
<td>6.7</td>
<td></td>
<td>2950(2834)</td>
</tr>
<tr>
<td>2771</td>
<td>3.9</td>
<td></td>
<td>2761(2647)</td>
</tr>
<tr>
<td>2642</td>
<td>1.8</td>
<td></td>
<td>2638(2525)</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th></th>
<th>H2 AJ</th>
<th>H2 RJ</th>
<th>NH3 AJ</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Mbol</td>
<td>T. mo</td>
<td>Mbol</td>
</tr>
<tr>
<td>3136</td>
<td>7.4</td>
<td></td>
<td>3059(2979)</td>
</tr>
<tr>
<td>2910</td>
<td>4.4</td>
<td></td>
<td>2864(2784)</td>
</tr>
<tr>
<td>2760</td>
<td>2.6</td>
<td></td>
<td>2732(2652)</td>
</tr>
<tr>
<td>2653</td>
<td>1.5</td>
<td></td>
<td>2636(2556)</td>
</tr>
<tr>
<td>2593</td>
<td>0.7</td>
<td></td>
<td>2581(2501)</td>
</tr>
</tbody>
</table>

3.0 HALOR COMBINED WITH NSSK

The results given above are somewhat academic. The real question is how to do both limited orbit raising and NSSK most effectively with an electric propulsion system(s). Out of the many possible approaches, we have chosen to illustrate the problem using the following examples:

1. The H2RJ for HALOR and the hydrazine AJ for NSSK, with the addition of four H2RJ thrusters (configuration 1).
2. The SPT-100 for both tasks, assuming the addition of two thrusters for added redundancy (configuration 2).
3. The XIPS-30 (de-rated) for both tasks, with no additional hardware (configuration 3).
4. The H2RJ for HALOR using a four thruster system, and a four thruster XIPS-13 system for NSSK (configuration 4).

The configurations are sketched in Figure 3.
3.1 CONFIGURATION 1

Configuration 1 assumes the hydrazine arcjets (HAJ) to be mounted at the same angle as on TELSTAR or INTELSAT 8, canted at an angle of 18 degrees away from the pitch axis. The four H2RJ's are mounted on the anti-earth face of the S/C and parallel with the yaw axis. OR will be accomplished by rotating the S/C 90 degrees around the pitch axis, so that the H2RJ's come parallel to the roll axis. Since the NSSK arrangement has not been changed in any way in this example, the benefits of HALOR can be evaluated independently. The correction to Figure 2 in this case amounts to subtracting from the BOL mass, the entire mass of a 4-thruster H2RJ system, i.e., 4 thrusters, PCU's, and gimbals, and the masses of the structure, harness, and miscellaneous, as given in Table 3. To this sum, 66.5 kg, it is customary to add a 20% contingency factor, which brings the total to 80 kg.

3.2 CONFIGURATION 2

Configuration 2 shows six gimbaled SPT-100's mounted on the anti-earth face of the S/C. Two thrusters are parallel to the yaw axis, and will presumably accomplish the HALOR task and then back up the NSSK function. The other two pairs of thrusters can back up the OR task and perform the NSSK. A minimum of one north pointing and one south pointing thruster must be operational at EOL. This configuration sacrifices some thrust efficiency in the NSSK mode, by being canted at a large angle, but this is probably advisable in any event, to avoid S/C contamination. In this configuration the combined OR and NSSK tasks may be handled as a single mission. To evaluate OR alone, the correction to Figure 2 is to subtract from the BOL mass: (a) a 22% xenon mass penalty for NSSK, owing to the higher cant angle of the thrusters, and (b) the mass of a 2-thruster SPT-100 thruster system, that is, 2 thrusters, PCU's, and gimbals, and about half of the masses listed in the remaining columns of Table 3. The hardware mass amounts to 60 kg, including the 20% contingency. The propellant mass penalty is of course a function of the BOL mass, but can be closely approximated by the (3 month OR) value of 26 kg. The total mass penalty chargeable to HALOR is therefore 86 kg.

3.3 CONFIGURATION 3

Configuration 3 uses the same scheme as configuration 2, but with only 4 thrusters, since the XIPS-30 thruster has been shown to have a very high life expectancy even when operated in the normal mode, and we are using the de-rated mode in this example. For the XIPS-30, 4-thruster system, the increase in mass to do OR is the difference between it and the 4-thruster XIPS-13 system which would be used for NSSK alone. Here the propellant mass penalty owing to the cant angle is about 28% (cos 25/cos 45), and this must be increased by an additional 11% because of the lower exhaust velocity of the XIPS-30 relative to the XIPS-13. The
total mass penalty assigned to OR with the XIPS-30 is the sum of 32 kg of extra xenon, and 82 kg of extra hardware, or 114 kg.

3.3 CONFIGURATION 4

Configuration 4 is an attempt to blend the high NSSK mass benefits of the ion thruster with the rapid OR capabilities of the H2RJ. It is geometrically the same as configuration 1, except that the XIPS-13 ion thrusters are mounted at a cant angle of 25 degrees instead of the 18 degrees for the HAJ's. Therefore the mass penalty for doing OR is the same as in configuration 1, namely, the mass of the 4-thruster RJ system, or 80 kg.

4.0 RESULTS

Results will be given under three headings: (a) BOL mass vs transfer time; (b) reduced launch mass requirement; and (c) net cost benefits stemming from reduced launch requirements.

4.1 BOL vs TRANSFER TIME

Figure 4 shows the corrected BOL masses (i.e., OR hardware subtracted) for the four designs described above, next to the uncorrected results from Figure 2 for easy comparison. The superiority of the H2RJ for the OR portion of the OR/NSSK mission is even more evident here, relative to doing the combined mission with the ion thrusters (without and with grids). The increase in the BOL mass, secured by subtracting the all-chemical BOL mass (2517 kg) from the EP values, is tabulated below over the range of interest in transfer time:

<table>
<thead>
<tr>
<th>T. mo.</th>
<th>H2RJ</th>
<th>SPT-100</th>
<th>XIPS-30</th>
</tr>
</thead>
<tbody>
<tr>
<td>2</td>
<td>198</td>
<td>63</td>
<td>43</td>
</tr>
<tr>
<td>3</td>
<td>323</td>
<td>133</td>
<td>113</td>
</tr>
<tr>
<td>4</td>
<td>433</td>
<td>198</td>
<td>183</td>
</tr>
<tr>
<td>6</td>
<td>608</td>
<td>333</td>
<td>323</td>
</tr>
</tbody>
</table>

Thus, for these reasonable transfer times, the increase in BOL mass is 2-4 times greater when the RJ is used, as compared with the ion thrusters.

It is generally conceded that an increase in BOL mass, provided it can be converted to payload or increased S/C life, is worth much more than mere launch cost savings. However, because of the many launch vehicle options (and associated cost options), it is difficult to evaluate the net financial benefit of higher BOL mass except in specific cases. For convenience, in making general comparisons, it has become one of the accepted practices
to convert mass benefits to launch cost savings, and thus to secure a conservative estimate of financial benefits. We will carry out this exercise for the fully-penalized OR portions of the OR/NSSK maneuver, for each of the four configurations in Figure 3.

4.2 LAUNCH MASS REDUCTION

Assuming the required BOL mass (for EP OR) is the sum of (a) the spent EP hardware used solely for OR and (b) the BOL mass for all chemical attainment of GEO, the required mass in the several sub-GEO TO's used in this study can be back calculated and compared with the mass required in GTO for an all-chemical transfer. The use of this technique results in smaller TO masses than those discussed previously, hence the trip times will also be reduced. The results are shown in Figure 5 in the form of launch mass savings vs. trip time. The mass savings stemming from the use of the H2RJ in configurations 1 and 4 are impressive, ranging up to over 1000 kg for a 4 month transfer. Mass benefits of the SPT-100 (configuration 2) and the derated XIPS-30 (configuration 3) are also substantial: above 500 kg for a transfer time of about 5 months.

4.3 NET COST BENEFITS

The launch mass reduction can be converted to cost benefits by applying a simple launch cost per kilogram factor. This is not of course applicable in all cases, especially when the mass savings permit the use of a smaller launch vehicle. However, the use of such a linear relationship provides an easy, conservative yardstick. Factors of $30,000 to $50,000 per kilogram to GTO have been used by various studies. We will adopt the middle course and settle on $40,000 per kilogram. Countering this is the cost of the EP hardware itself, and the cost of idle capital during the transfer. The latter can be estimated as about 1% per month of orbit raising, of the entire capital invested in the S/C including launch cost. An investment of $200 M will be assumed here. The hardware cost is more difficult to estimate. For the H2RJ, a relatively simple device, we will assume a "routine production" cost of about $1.5 M for a complete four thruster system. For the SPT-100 and the XIPS-30, the hardware is more complex, but we need count only the excess cost owing to expanding the NSSK task to include HALOR. For the SPT-100, only two thrusters need be added; for the XIPS-30, only the increased cost of hardware based on 30 cm dia thrusters relative to that of 13 cm thrusters need be considered, since a four thruster system can certainly perform both tasks. For want of any solid figures, we will assume the same added cost as in the case of the RJ.
\text{Cost Benefit} = M \times \$40K - \$1.5M - \$2M \times T \text{ (mo).} \quad (7)

Results are given in Figure 6. From this we can expect substantial net savings from HALOR. For the H2RJ, \$1M to \$30M for transfer times of 1-4 months; For the SPT and gridded ion thrusters, \$1M to \$25M for transfer times of 3-9 months.

4.4 COMPARISON WITH FULL LEO/GEO MANEUVER

Comparison of HALOR with the full LEO/GEO maneuver gives the surprising result that, for short trip times, HALOR actually results in a higher net BOL mass than does the larger mission. Figure 7 shows the comparison for the H2RJ, Figure 8 does the same for the SPT-100. These figures are simply the superposition of the pertinent parts of Figures 1 and 4. For the H2RJ, the HALOR mission outperforms the LEO/GEO for transfer times up to slightly more than 3 months. In the case of the SPT-100, the crossover is not reached until about 4.4 months. The result of course is owing to the large mass for the propulsive power which is required for short LEO/GEO trip times. In most cases, this mass cannot be counted as effective BOL mass.

5.0 CONCLUSIONS AND RECOMMENDATIONS

High altitude orbit raising offers an attractive alternate path to full LEO/GEO transfers. The environment is benign compared with most of the LEO/GEO transfer, and solar cell degradation, drag, and shadowing are reduced to negligible proportions. A HALOR effort would probably cost about two orders of magnitude less than a full LEO/GEO task.

Although all the EP devices studied yield appreciable mass and cost gains relative to straight chemical propulsion, the H2RJ outperforms the others by a considerable margin. The H2AJ is also noticeably better than the electrostatic devices. HALOR actually outperforms the full LEO/GEO maneuver for short trip times (3-4 months), in terms of net BOL mass at GEO.

It is the author's opinion that fear of the potential loss of the entire S/C is the main obstacle to the use of EP orbit raising. This is almost certainly true in the commercial sphere, and may be of equal weight in scientific, military, and other government ventures. By invoking HALOR, substantial cost benefits appear to be available with minimum risk: in the case of the H2RJ, only two months of operation, a mere 1500 hours, will generate net cost benefits in the neighborhood of \$10M. As confidence in the reliability of this and other EP devices increases with experience, the horizons could be gradually expanded to longer trip times and greater benefits. Keep in mind that these estimates are very conservative, based as they are on launch cost savings without stepping down to a smaller launch vehicle. Several studies cited here have shown that such
stepping down may involve savings nearly an order of magnitude greater than those outlined here. Also, once the OR hardware is in place, it can be used for repositioning and graveyard maneuvers, and to back up the CPS used for EWSK. While these maneuvers are small, they add something to the mass savings discussed here. It may be that in certain cases the inclusion of these minor maneuvers may enable a step down in launch vehicle size.

The performance figures given here for the H2RJ are based on work performed in the mid-60's. A fresh look at this device seems in order, first to confirm the figures which Page and his associates generated (10), then to demonstrate firmly the lifetime required for orbit raising, and finally to explore whether technical advances, made over the last three decades, can be applied to this simple yet remarkable device.

REFERENCES


FIGURE 1. ELECTRIC PROPULSION FROM LEO TO GEO.
Figure 4: BOL mass vs. trip time for OR and NSSK configurations.

Figure 2: Uncorrected BOL mass vs. trip time for HALOR.
CONFIGURATION 1: OR WITH H2RJ's; NSSK WITH HAJ's.

CONFIGURATION 2: BOTH OR AND NSSK WITH SPT-100's.

CONFIGURATION 3: BOTH OR AND NSSK WITH XIPS-30's.

CONFIGURATION 4: OR WITH H2RJ's; NSSK WITH XIPS-13's.

FIGURE 3. SELECTED PROPULSION GEOMETRIES FOR ORBIT RAISING AND NSSK.
Figure 5. Launch Mass Reduction for HALOR.

Figure 6. Net Cost Benefits for HALOR, OR/NSK Combos.
FIGURE 7. COMPARISON OF HALOR WITH LEO/GEO FOR H2RJ

FIGURE 8. COMPARISON OF HALOR WITH LEO/GEO FOR SPT-100