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Electric Propulsion for Low Earth Orbit Constellations

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Hall Effect electric propulsion was evaluated for orbit insertion, satellite repositioning, orbit maintenance and de-orbit applications for a sample low earth orbit satellite constellation. Since the low masses of these satellites enable multiple spacecraft per launch, the ability to add spacecraft to a given launch was used as a figure of merit. When compared to chemical propulsion, the Hall thruster system can add additional spacecraft per launch using planned payload power levels. One satellite can be added to the assumed four satellite baseline chemical launch without additional mission times. Two or three satellites may be added by providing part of the orbit insertion with the Hall system. In these cases orbit insertion times were found to be 35 and 62 days. Depending on the electric propulsion scenario, the resulting launch vehicle savings is nearly two, three or four Delta 7920 launch vehicles out of the chemical baseline scenario's eight Delta 7920 launch vehicles.

INTRODUCTION

Many new, low earth orbit (LEO) constellations are being planned or put into service.[1] The lower altitudes of these satellites necessitate many more satellites as opposed to the only three or four geostationary satellites, required for global coverage. Use of electric propulsion for these LEO constellations can greatly reduce propulsion system wet mass. This benefit can be translated into longer lifetimes, larger payload masses, or reduced launch masses. This final benefit allows for either using smaller launch vehicles or launching more satellites per launch vehicle. With the many satellites required for LEO constellations this last scenario can allow a substantial reduction in the amount of launch vehicles required to place the constellation into service.

Hall thrusters have been used on LEO spacecraft since their introduction on the Russian Meteor series of low earth orbit weather satellites.[2] Many proposed LEO satellite systems have relatively high power payloads,[3] which are not in use during satellite delivery, repositioning and disposal and could be effectively used by an electric propulsion system to increase payload mass or reduce launch mass.

A previous study dealt with electric propulsion specifically for LEO *communication* satellites.[3] In the study described in this paper an assessment of the benefits of Hall propulsion systems for a "generic" LEO constellation are made. The performance advantages were determined in terms of increased number of satellites per launch vehicle. The sample mission uses available information on launch vehicles and a sample satellite constellation to create the generic scenario.[2,4,5]

MISSION ANALYSIS, OPTIONS AND ASSUMPTIONS

Several mission tools were used in these analyses to provide low thrust trajectory, atmospheric drag, earth oblateness and shadow modeling. The numerical optimization program Solar Electric Propulsion Steering Program for Optimal Trajectory (SEPSPOT) was used for determining optimal solar electric propulsion starting orbits and optimal steering for constant and shaded thrusting orbits.[6] The numerical orbit integration program Systems Evaluation of Orbit Raising (SEOR), was used to test the use of circumferential steering.[7] The routine, Thrusting Orbiter with Atmospheric Drag (TOAD) was used to assess the impact of atmospheric drag on the transfer time and velocity change (ΔV) required for the low thrust transfer.[8] All chemical systems were assumed to burn impulsively. Repositioning operations were simulated using a scheme developed elsewhere.[9]

Constant, Circumferential Thrusting

In operation LEO satellite systems have active, relatively high power payloads which require power in shade and sunlight.[3] Because the payload is usually not in use during satellite delivery, repositioning and disposal, the power could be made available to the propulsion system. Thus, in this study, the electric propulsion (EP) systems described are assumed to operate from the solar arrays during the sunlit portions of the trajectory and from the batteries in the shadow portion. This use of payload battery power for electric propulsion has precedent with North-South stationkeeping using arcjet thrusters on geostationary spacecraft. It is assumed that the additional cycling and different charging patterns will have minimal impact on the multi-year power systems; a short electric propulsion orbit insertion and de-orbit adds only a few extra months to many years of cycling.

One benefit of using the payload's power system in light and shade is the avoidance of non-thrusting periods during shadow. This should allow for simplified, circumferential steering. It has been shown with SEPSPOT that the required in-plane steering angle for another sample LEO spacecraft without shadowing is 0° or simply circumferential (i.e., perpendicular to the radius direction in the plane of the orbit).[4] The shaded optimal steering is more complex and varies depending on shadow conditions throughout the trajectory.

Other power/orbit/steering scenarios are possible. For instance, using all the available, beginning-of-life (BOL) solar array power, a higher power (but heavier) electric thruster system could be used but only during sunlit portions of the orbit. Such a trajectory would require more complex steering as mentioned above. In addition, the BOL power would not be available at the end-of-life and thus would require a throttleable thruster system. Another possibility would be to use shorter electric propulsion burns and start in an elliptical orbit; the electric propulsion system imitates a chemical thruster. This method, while reducing ΔV, would probably require a longer trip time, as shown by Pollard and Janson.[10] These options will be considered in further analyses.

SYSTEM ASSUMPTIONS

Two propulsion systems were assumed in the analysis performed for this study. The candidate systems were meant to be representative of state-of-the-art and to show the benefit of electric propulsion. The state-of-art chemical system assumed a 5 kg dry mass (less tanks), an 8% tankage, and an I_{sp} of 235 seconds.[2] The candidate electric propulsion system was a 1.2 kW xenon Hall thruster with parameters shown in Table I. This candidate Hall thruster technology is representative of the SPT-100[11], the T-100[12], and the D-55.[13] More information concerning each, including availability, can be found in the referenced texts.

Mission Phases

A generic surveillance system of three planes of 8 satellites each along with 8 spares at an altitude of 1850 km and 90° inclination was chosen as the sample LEO constellation.[2] Based on the reported satellite mass the sample baseline satellite is extrapolated to be approximately 608 kg at launch with a non-propulsion mass of 440 kg. The payload power is assumed to be 1.2 kW. A hydrazine chemical propulsion system is baselined for this sample mission. The Delta 7920[tm] which is assumed to deliver four chemical satellites to the operating orbit, was used in this analysis. A 10 year lifetime is assumed including orbit maintenance and an end-of-life de-orbit.[14] De-orbit for the chemical system is assumed to consist of changing only the orbit perigee to 185 km and allowing atmospheric drag lower the apogee and then de-orbit the spacecraft. To simplify steering, the electric propulsion systems are assumed to spiral down to a 300 km circular orbit and allow drag to de-orbit the spacecraft. Analyses have shown that total disposal times for both the chemical and electrical propulsion are roughly equivalent for the sample system.

The spacecraft are also assumed to require a multiple, quick reposition capability of one 45° orbit spacing change in 3 days for each year of service. This capability is roughly equivalent to a total 450° in-plane reposition at a rate of 15°/day reposition. Moves smaller than 45° may take longer than 15°/day rate. Also, a greater reposition can be made with the same fuel but at a slower rate. Repositions are achieved by maneuvering up or down to a 'coast' orbit where the difference in satellite speed allows a differential drift. At the appropriate time the spacecraft is returned to its original orbit. While use of chemical propulsion for repositioning requires practically all the drift to be made at the coast orbit, the electric propulsion system takes longer to transfer to the coast orbit and allows for some drift during transit. A complete discussion of repositioning can be found in Free[15] and Pollard.[16]

In summary, three mission tasks are assumed for the constellation.

— Ten Orbit Repositions: 45 degree in-plane orbit maneuver in three days ~17m/s/reposition (chemical), ~24 m/s/reposition (electric)

- Orbit Maintenance: ~5 m/s/year (includes margin) + 7 m/s for injection errors
- Deorbit:
    - Chemical Case: lower *perigee* from 1850 km to 185 km
    - Hall Cases: Lower orbit from 1850 km circular to 300 km circular

For some of the electric propulsion scenarios the launch vehicle is allowed to leave the spacecraft off at lower orbits to allow the electric propulsion system to complete the orbit insertion. The lower starting orbit is chosen to allow additional satellites to be launched.

- Orbit Insertion
    - Electric only: raise orbit from lower circular parking orbit to the 1850 km operating orbit

Mission task / propulsion system combinations are also shown in Table II.

Results

Baseline Chemical Scenario

The baseline system was assumed to use a hydrazine monopropellant chemical system (225 s I_{sp}, 8% tankage fraction) for the repositioning, orbit maintenance and the de-orbit. The Delta 7920 was assumed to deliver four, 608 kg satellites to the required 1850 km orbit. In all cases the separated Delta dispensing adapter, margin and penalty for the larger, 10 ft. fairing was assumed to be 160 kg.[5] The yearly 45°, 3 day reposition requires a ΔV of approximately 17 m/s. Orbit maintenance is assumed to be 5 m/s/year. After the 10-year lifetime the satellite must be de-orbited. A disposal orbit perigee of 185 km was assumed which would produce a de-orbit time of roughly three months. The velocity change needed to lower the orbit perigee to 185 km is 403 m/s. Thus, the total ΔV required was 628 m/s. The chemical hydrazine system mass required to perform these maneuvers, assuming a 608 kg initial mass, was 168 kg. Thus, the non-propulsive spacecraft mass required for performing the mission was found to be 440 kg. The spacecraft mass breakouts and the number of spacecraft per launch vehicle for the chemical and electrical propulsion options are shown in Figures 1 and 2, respectively.

Electric Propulsion Scenarios

The approximately 440 kg non-propulsive mass found in the Baseline Chemical Scenario was also assumed for the electric propulsion scenarios. The chemical propulsion system was replaced by an electric propulsion (EP) system. A 1.2 kW Hall propulsion system was considered based on past work (see Table I).[4] Because the payload power is assumed to be 1.2 kW in sunlight and shadow, the EP system was assumed to run off the solar arrays in sunlit portions of the trajectory and the batteries in the shadow portion. This use of payload battery power for electric propulsion was described in the mission analysis section. It was assumed that the additional cycling and different charging pattern will have minimal impact on the assumed 10 year system.

Five Satellites per Launcher

In this scenario the Hall thruster system directly replaced the chemical system. The spacecraft are launched directly to the 1850 km orbit by the Delta launch vehicle. Repositions would be the same 45° in 3 day rate as the chemical system also. While the ΔV to perform the reposition is larger than the chemical reposition ΔV, due to the relative low thrust of the Hall system, the much larger Isp of the Hall system more than offsets the penalty. The de-orbit altitude was changed to 300 km circular to allow for simplified constant thrusting from the Hall system. Once the spacecraft reaches the 300 km orbit, drag could be counted on to complete the de-orbit. Preliminary simulations have shown that the Hall system de-orbit time, a combination of the powered spiral down and the coasting deorbit, would be roughly three months - the same as the chemical baseline. Adding in the orbit maintenance to the repositioning and deorbit, the ΔV budget comes to 1058 m/s. The wet Hall system mass to perform the mission is only 54 kg. Thus the launch mass to provide the 440 kg non-propulsive spacecraft mass is 494 kg. With a launch capability of 2470 kg to the 1850 km, 90° inclination orbit the Delta 7920 should be able to launch five spacecraft. No assessment of volume constraints have been made, but it seems reasonable that an additional satellite could fit in the 10 ft. fairing.

Six Satellites per Launcher

Instead of using the Delta launcher to place the satellites directly into the final orbit like the chemical baseline mission and the five satellite electric propulsion scenario, this option begins from a lower circular orbit. At this starting orbit the Delta can place more mass. The EP system was then tasked with raising the spacecraft to the final 1850 km circular orbit and then maintaining, repositioning and de-orbiting the spacecraft. The starting circular orbit was chosen so that six spacecraft could be launched on the Delta launcher.

By using a Hall system the required EP circular starting orbit was 880 km with a trip time of 35 days. De-orbit time was still around three months total. The total mission ΔV was approximately 1500 m/s. Spacecraft launch masses were 510 kg with a Hall wet propulsion mass of 70 kg. With a launch capability of 3060 kg to the 880 km, 90° inclination orbit the Delta 7920 should be able to launch six spacecraft.

Seven Satellites per Launcher

The higher starting orbits could be lowered even further to 400 km (any lower may encounter excessive drag) and a higher I_{sp} Hall system used to allow seven spacecraft to be launched. The Hall system is assumed to be modified to a 1850 second I_{sp}, 55% efficient system. The I_{sp} is increased to minimize the required fuel mass at the expense of trip time. Starting from the lower 400 km starting orbit would also contribute to a longer insertion time (62 days). The ΔV budget for the orbit insertion, repositioning, maintenance, and de-orbit comes to 1800 m/s. This budget is higher due to the larger orbit insertion (714 m/s). It is also higher due to a higher reposition ΔV (27 m/s) due to the lower thrust of the higher I_{sp} Hall system. The Hall system wet mass for this option is 68 kg. The launch mass required for the 440 kg non-propulsive mass is 508 kg. Thus seven spacecraft would total 3556 kg, within the Delta 7920 capability for the 400 km, 90° starting orbit.

Volume Considerations

Packaging additional satellites into the Delta 7420 fairing was not considered in this analysis due to lack of packaging and dispenser information.

Launches Saved

For the assumed system, a total constellation of 32 satellites including 8 spares must be launched to provide complete service. Assuming all the satellites were to be launched on Delta 7920s, eight launch vehicles would be required: 32 satellites / 4 per launch = 8 Deltas. With electric propulsion adding one satellite per launch almost two Delta launch vehicles could be saved: 32 satellites / 5 per launch = 6 Deltas plus two satellites. These two satellites could be spares and perhaps piggy backed on another launch or launched on a smaller launch vehicle when needed. With electric propulsion adding two satellites per launch almost three Delta launch vehicles could be saved: 32 satellites / 6 per launch = 5 Deltas plus two satellites. Again the two satellites should be cheaper to launch. Finally, with electric propulsion adding three satellites per launch the required number of Delta launch vehicles can be almost halved: 32 satellites / 7 per launch = 4 Deltas plus four satellites. Again, these four satellites can be considered spares and could be launched on smaller launch vehicles.

CONCLUSIONS

It was shown that the mass of up to three satellites can be added to multiple LEO constellation spacecraft launches by using electric propulsion for orbit insertion, maintenance, repositioning, and de-orbit. The result is almost halving the number of launch vehicles required, from eight to four and a half. If only one satellite is added per launch practically no extra time is required for orbit insertion of other mission operations. To add two or three extra satellites, orbit insertion times of 35 and 62 days would be needed, respectively. A simple circumferential steering method was assumed which relies on the payload's solar array and battery power and eliminates the more complex steering required when shading of the solar arrays must be considered. Assumed reposition rates were the same for the chemical and Hall systems: 45° in 3 days. Deorbit times were also the same for the chemical and Hall systems- roughly three months.

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Table I—Propulsion Systems for LEO Constellation

Propulsion System Parameters	Xenon Hall Thruster	Modified Xenon Hall Thruster
Desired PPU Input Power Level	1.2 kW	1.2 kW
Isp	1500 s	*1850 s*
Overall Efficiency (PPU & Thruster)	0.45	*0.51*
Tankage	10%	10%
Masses:		
Thruster	5 kg	5 kg
Gimbals	34 % of Thruster	34 % of Thruster
Support	31% of Gimbals & Thrusters	31% of Gimbals & Thrusters
Controller	0.55 kg/Thruster	0.55 kg/Thruster
Total Thruster + Gimbal + Support + Controller	9.3 kg/thruster	9.3 kg/thruster
Feed System	1.5 kg/kWe	1.5 kg/kWe
PPU	4.7 kg/kWe	4.7 kg/kWe
Cabling	0.4 kg/kWe	0.4 kg/kWe
Thermal Sys. (92% PPU)	31 kg/kWt-disp.	31 kg/kWt-disp.
Total PPU + Feed + Cabling + Thermal	9.1 kg/kWe	9.1 kg/kWe

Table II—Mission phases and propulsion options for LEO spacecraft

Mission Phase	**Baseline: Monoprop 4 Sats/Launch**	**EP 5 Sats/Launch Scenario**	**EP 6 Sats/ Launch Scenario**	**EP 7 Sats/ Launch Scenario**
•Orbit Insertion				
Raise orbit from 880 km to 1850 km			Hall (1500 s Isp)	
-Raise orbit from 400 km to 1850 km				Hall (1850 s Isp)
•Repositions 45° in orbit in 3 days (10 total)	Monoprop (225 s Isp)	Hall (1500 s Isp)	Hall (1500 s Isp)	Hall (1850 s Isp)
•Orbit maintenance 5 m/s/y	Monoprop (225 s Isp)	Hall (1500 s Isp)	Hall (1500 s Isp)	Hall (1850 s Isp)
•Deorbit				
-Lower perigee from 1850 km to 185 km	Monoprop (225 s Isp)			
-Lower orbit from 1850 km to 300 km		Hall (1500 s Isp)	Hall (1500 s Isp)	Hall (1850 s Isp)

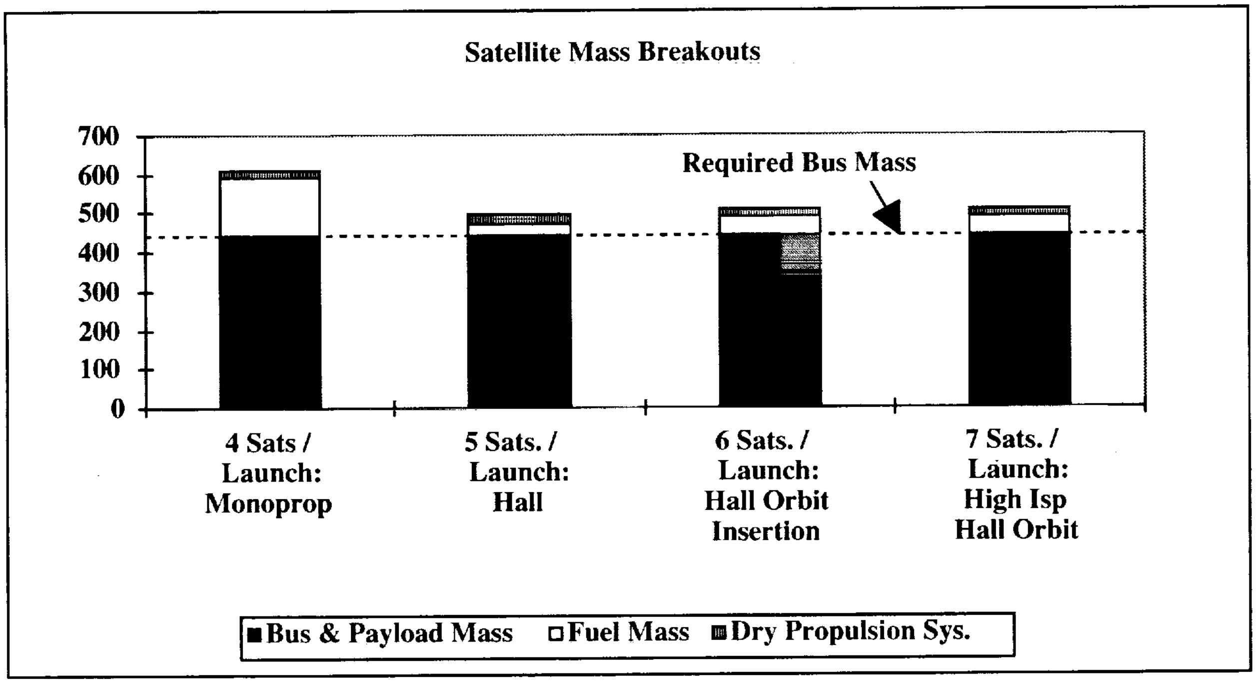

Figure 1—Satellite Mass Breakouts

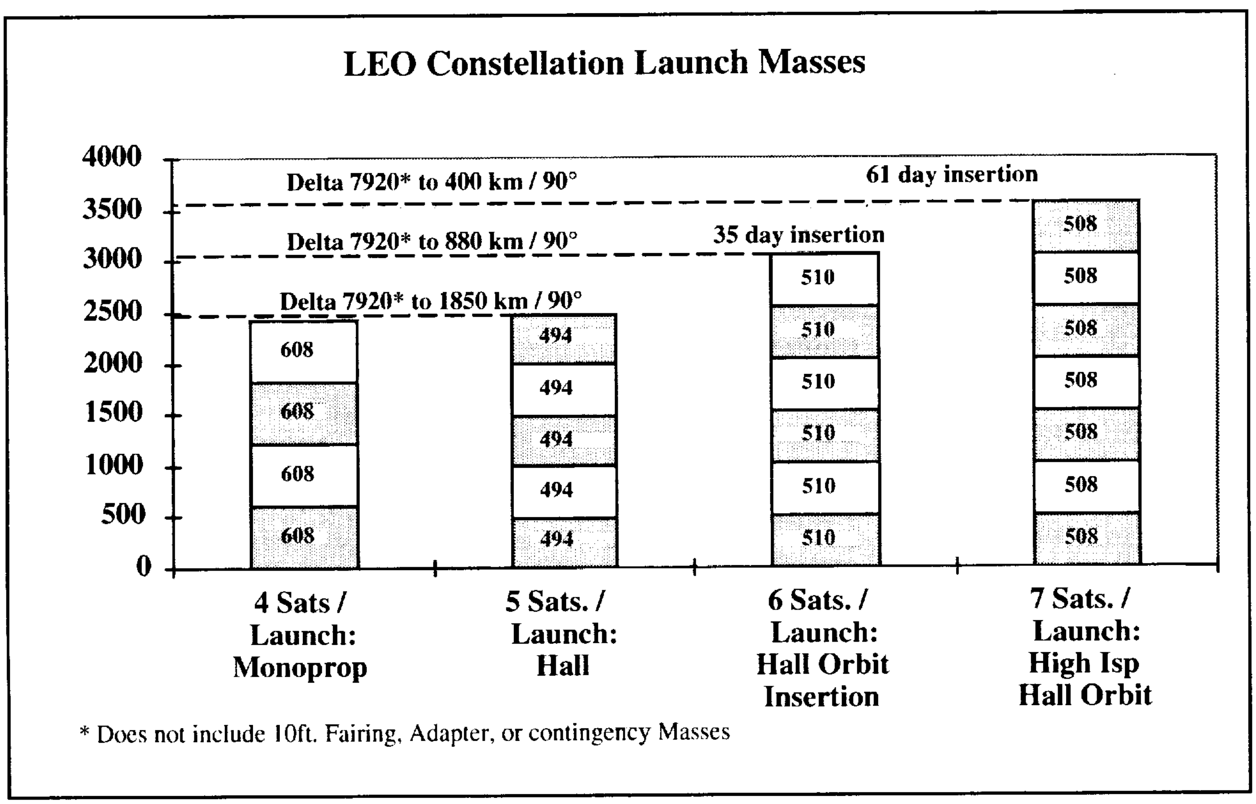

Figure 2—Launch Masses for Each LEO Constellation Propulsion Scenario

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Hall Effect electric propulsion was evaluated for orbit insertion, satellite repositioning, orbit maintenance and de-orbit applications for a sample low earth orbit satellite constellation. Since the low masses of these satellites enable multiple spacecraft per launch, the ability to add spacecraft to a given launch was used as a figure of merit. When compared to chemical propulsion, the Hall thruster system can add additional spacecraft per launch using planned payload power levels. One satellite can be added to the assumed four satellite baseline chemical launch without additional mission times. Two or three satellites may be added by providing part of the orbit insertion with the Hall system. In these cases orbit insertion times were found to be 35 and 62 days. Depending on the electric propulsion scenario, the resulting launch vehicle savings is nearly two, three or four Delta 7920 launch vehicles out of the chemical baseline scenario's eight Delta 7920 launch vehicles.

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