APPLICATION OF DC ARCJET FOR LOW EARTH ORBIT SPACECRAFT

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Abstract

Large orbit change capabilities are required for low earth orbit spacecraft in the Big LEO systems. The total delta-V amounts to several hundred m/s, summarizing the requirements to reach and maintain mission orbit and to deorbit itself at the end of mission. Application of DC arcjets to those LEO(Low Earth Orbit) spacecraft will lead to weight reduction of propulsion system or to improvement of mission capabilities.

A case study was made for typical LEO spacecraft in order to clarify requirements for the propulsion system. If 1000 kg for launch mass, 2000 W for solar array generation power, and 1400 km for mission orbital altitude are assumed for the spacecraft system characteristics, and 100 mN/kW for thrust power ratio and 500 s for specific impulse are assumed as a target performance for the DC arcjet, a cluster of 500 W class arcjets was turned out to be preferable. The development targets for the arcjet were turned out to be 900 hours for operational life time and 1700 restart times. The required period was 55 days for the initial ascent to mission orbit, 9 days for orbital phase change for constellation maintenance, and 54 days for deorbit operation. Required hydrazine propellant amounts to 110 kg for the case. Throttling or pulse modulation capability is highly desirable for this application.

Introduction

So-called Big LEO systems such as Iridium, Globalstar, SkyBridge, Teledesic and so on are now in operation or under construction. Those systems consist of tens or hundreds of satellites in low earth orbits. Large orbit change capability is required for those satellites, considering launcher capability, mission orbit maintenance, and deorbit operation for orbital debris mitigation. If some kind of effective electric propulsion is applied to those satellites, their system capabilities or costs will be much improved, and it will be a good chance to widely enlarge application of electric propulsion.

Objectives of this paper are to clarify performance requirements, system configuration, and operation scenario for the DC arcjet thruster system in those applications. The reason why the DC arcjet is selected is that it has relatively large thrust-to-power ratio and simple geometrical and electrical configurations. Large thrust-to-power ratio leads to less electrical power requirement or to shorter time requirement for the operation. Simplicity leads to lower recurring cost for mass production.

Typical Characteristics of LEO Spacecraft

Designed or planned characteristics of LEO spacecraft gathered from the web sites[1,2] are summarized in Table 1. For a case study, a set of characteristics were selected as shown in Table 2. The characteristics are not simple averages, but some capability enhancement in the future was taken into account.

Table 1. Characteristics of Typical LEO Spacecraft.

Spacecraft Name	Iridium	Globalstar	Teledesic	SkyBridge
Initial Mass (kg)	700	450	771	1,250
Electrical Power (W)	1,400	1,000	—	—
Orbital Altitude (km)	780	1,410	695-705	1,469
Number of Spacecraft	66+6	48+8	840+84	80

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Table 2. Spacecraft Characteristics for Case Study.

Characteristics	Parameters
Initial Mass (kg)	1,000
Initial Orbital Altitude (km)	900
Operational Orbital Altitude (km)	1,400
Solar Cell Genarating Power (kW)	2,500

Orbit Change Requirements

Orbit change requirements for LEO spacecraft are as follows;

1) transfer from initial orbit inserted by launcher to mission orbit,
2) orbit corrections for constellation maintenance,
3) transfer from storage orbit for spare spacecraft to replace malfunctioned spacecraft, and
4) deorbit or reorbit at the end of mission life for space debris mitigation.

Requirement 1) depends on launcher capability. If direct insertion to mission operational orbit is applicable, spacecraft's own orbit change capability is not required. Some of the Globalstar satellites, however, raised their orbital altitude by themselves from 900 km to 1400 km for the launch by Soyuz/Ikar. Also SkyBridge spacecraft has a capability to raise its orbit altitude from 940 km to 1469 km. The spacecraft's own orbital change capability will increase its launch opportunities and be an advantageous feature. For low earth circular orbit, altitude change capability of 500 km is approximately equivalent to delta-V of 200 m/s.

Requirement 2) is negligible in comparison to the other three requirements. Since residual air drag is low at high altitude over 800 km, there is no significant orbital disturbance.

Requirement 3) is determined by tolerance for time interval from loss of an operating spacecraft to replacement by a spare. Although quick system recovery is preferable, available electric power for electric propulsion limits magnitude of thrust and the recovery time. Required resources were estimated by simulation for 180 deg orbital phase change with typical DC arcjet performance.

Requirement 4) is to remove life expired spacecraft from useful orbit in order to mitigate space debris production. According to NASA guideline[3], LEO spacecraft at its mission end should be transferred to those orbits in which the spacecraft reenters the atmosphere within 25 years. Otherwise, it should be transferred to less useful altitude beyond 2500 km. This requirement is the severest of the four requirements and will become inevitable for those large scale activities in LEO. If the altitude of mission operational orbit is around 1400 km, lowering the orbit for earlier reentry is more acceptable than raising the orbit over 2500 km.

Simulation Analysis

Assumptions.

Orbital propagations with low thrusts were calculated with 4th order Runge-Kutta method.

Assumptions for spacecraft characteristics are shown in Table 2. Available electrical power averaged over the orbit is assumed to be 500 W, and peak power of 2000 kW is assumed to be available within a energy balance over the orbit. Though batteries can provide propulsion system with larger peak power over 2000 W, so-called solar array lockup limits the peak power during sun-lit. If electrical load exceeds generated power by solar cells, batteries will discharge even in sun-lit portion of the orbit, and then energy balance of the system will be lost.

As a target performance for the DC arcjet thruster system, thrust power ratio was assumed to be 100 mN/kW, including power processor efficiency, and specific impulse was to be 500 s.

Initial orbit raising.

Spiral orbit raising with low continuous thrust from 900 km altitude circular orbit to 1400 km was simulated. Since available electrical power is assumed to be 500 W and since thrust power ratio of 100 mN/kW is assumed for propulsion system performance, the continuous thrust used in calculation was 0.05 N. Though peak power of 2000 W can produce 0.2 N, appropriate duty control for averaging over the orbit will lead to almost same result as continuous thrust of 0.05 N.

Results of the simulation are shown in Table 3. "Number of restarts" shows minimum restart times of the thruster for the duty controlled peak power (2000 W, 0.2N) operation and coincides with the number of orbital circulations during the orbit raising operation.

Table 3. Results of Simulation Analysis.

Maneuvers	Accumulated Thruster Operation Time (hours)	Number of Thruster Restart Times	Required Period for the Maneuver (days)	Propellant Consumption (kg)
Initial Orbit Raising	340	770	55	50
180 deg Phase Change	220	130	9	9
Deorbit within 25 years	340	770	54	50
total	900	1,670	118	109

Phase change maneuver.

If a spare spacecraft is stored in the same orbital plane and the same altitude as the operational spacecraft, 180 deg phase change is the worst case maneuver requirement. Phase change is achieved by altitude maneuver. Advancing phase change can be achieved by spiral altitude lowering first, and then by spiral raising back to the original altitude. On the other hand, retrograde phase change can be achieved by spiral orbit raising first, and then by spiral lowering back to the original. The quickest maneuver for 180 deg phase change is achieved by two power flight paths with maximum thrust and switching directions at the mid point (90 deg). The maneuvering scenario for phase change is shown in Fig. 1.

The simulation results are shown in Table 3. "Number of restarts" shows minimum restart times of the thruster for the duty controlled peak power (2000 W, 0.2N) operation and coincides with the number of orbital circulations during the phase change operation.

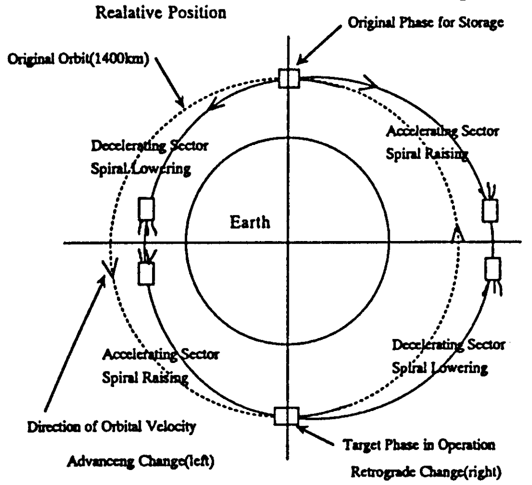

Fig. 1 Schematic Illustration of Maneuver for 180 deg Phase Change

Deorbit maneuver.

As shown in Fig. 2, orbital life is limited by air drag in LEO[3]. It is more effective to lower perigee altitude than to spiral lowering both perigee and apogee for the same delta-V. If the start point is a circular orbit of 1400 km altitude, and if area-to-mass ratio of the spacecraft is 0.02 m^2/kg, perigee lowering to 450 km is required for the reentry within 25 years.

For perigee lowering, propellant will be used effectively, if power flight path is limited in the vicinity of apogee as shown in Fig. 3. If electrical power for the thruster system is constant, operation period will increase as the power flight path is shortened. Available electrical power, however, will increase as the power flight path is shortened, since the averaged power is constant and the duty ratio decreases.

Since the powered flight path has finite length and since the impulse is not applied just at the apogee point for such low thrust propulsion system as electric propulsion, apogee is also lowered. Then the required perigee altitude for the early reentry becomes higher as the apogee altitude is lowered.

The results of the simulation, in which variations of available peak power and variation of perigee altitude are taken into account, are shown in Table 4. Within the peak power limit of 2000 W, power flight path of apogee plus or minus 45 deg gives the least propellant consumption and the shortest elapse time for the operation.

Table 3 includes the requirements for this most efficient case. Since the thruster system is powered on and off once per every circulation of the orbit, number of thruster restarts coincides to the number of orbital circulation during the orbit change operation. Duration of each power flight was approximately 26 min. Quick start and stabilization within a minute are required for this maneuver.

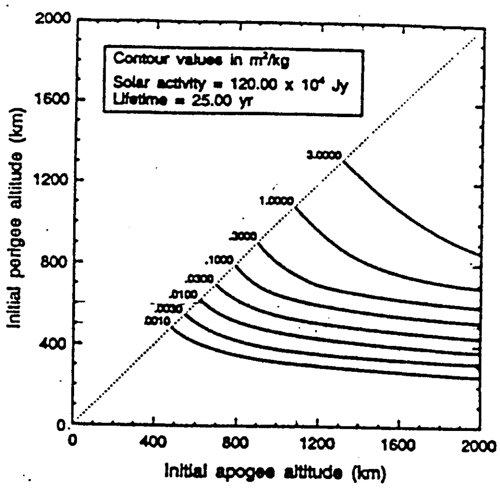

Fig. 2 Contours of Constant Area to Mass Ratio for 25 Year Orbit Lifetimes for LEO Orbits[3].

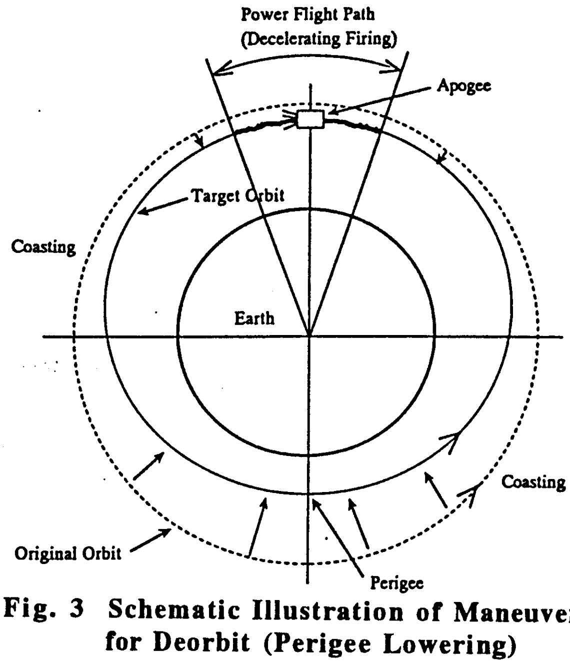

Fig. 3 Schematic Illustration of Maneuver for Deorbit (Perigee Lowering)

Table 4. Summary of Deorbit Simulation

Power Flight Arc (deg)	Electrical Power for Propulsion (W)	Thrust (N)	Mass Flow Rate (mg/s)	Target Apogee Altitude (km)	Target Perigee Altitude (km)	Accumulate Firing Time (hours)	Number of Firing Restarts	Priod for Operation (days)	Propellant Consumption (kg)
±5	18,000	1.8	367	1,399	450	36	725	50	47.3
±15	6,000	0.6	122.3	1,394	450	110	730	51	47.6
±30	3,000	0.3	61.2	1,375	450	220	750	52	48.4
±45	2,000	0.2	40.8	1,340	450	340	770	54	49.8
±60	1,500	0.15	30.6	1,300	460	465	800	56	51.2
±90	1,000	0.1	20.4	1,180	480	760	880	61	55.8
±120	750	0.075	15.3	1,020	510	1130	1000	69	62.0
±150	600	0.06	12.2	824	560	1570	1130	78	69.1
±180	500	0.05	10.2	670	670	1940	continuous	81	71.1

Discussions

Advantages of DC arcjet application.

If hydrazine mono-propellant thruster system is employed to the same orbit change maneuver as discussed in the former section, its propellant consumption amounts to 270 kg, since its specific impulse is approximately 200 s. More than 100 kg (10%) mass reduction can be expected even if the additional mass of the arcjet system is taken into account. The mass reduction leads to handling and launch cost reduction or to increase in payload resources.

Long elapse time for operation is a demerit of low thrust system. Approximately 4 months required for orbit raising and deorbit maneuver by the DC arcjet system. The operational cost, however, will be much reduced, since some kind of autonomous control for the spacecraft must be introduced in the system. The autonomous control system will be inevitable to operate numerous satellites in the constellation system. For phase change operation, high thrust system needs time of almost the same order as the low thrust system, since coasting time is required between two altitude change maneuvers. The coasting time will be several days, unless extreme resource allocation for the operation is accepted.

Thruster system configuration.

To apply DC arcjet system to the spacecraft will need additional design and manufacturing costs, unless the conventional thruster system is eliminated from the spacecraft. Although significant weight reduction is expected, it has become less advantageous, since launch cost has become lowered in these days, and since advanced micro electronics reduces payload requirements. In order to overcome additional cost, applicability to attitude control is highly desired for such advanced propulsion system as DC arcjet system. Throttling or pulse modulation capability is required for the attitude control application.

Since the assumed averaged available power is 500 W, the minimum configuration of the thruster system consists of a single 500 W class thruster. Although a 2000 W class thruster with 25% duty control is equivalent to a 500 W class thruster with 100 % duty, power processor for the 2000 W thruster will requires more resources than that for the 500 W thruster.

Single thruster propulsion system requires auxiliary reaction control system, since misalignment of the primary thruster or location uncertainty of center of mass of the spacecraft may cause attitude disturbance. If correction for this operational disturbance is made only by conventional chemical thrusters, effective specific impulse will be lowered by accounting propellant for attitude control. Then advantage of high specific impulse of the electric propulsion will be lowered, too. A gimbal mechanism for a single thruster or throttling and/or modulation capability for a cluster of thrusters is required for effective application of the electric propulsion. A cluster system of four thrusters aligned in parallel can generate control torques around 2 axes other than roll axis, if their thrust magnitudes can be modulated or throttled as quickly as the attitude control system requires.

Therefore, even if the available peak power is 2000 W, a cluster of 500 W class arcjets is preferred to a single 2000 W class arcjet, provided that throttling or modulation capability is applicable.

Conclusion

Advantage of application of DC arcjet system to a typical LEO spacecraft was confirmed by a case study. Design target for DC arcjet propulsion system was defined as follows;

1) 500 W class arcjet is preferable to multi-kiro W class arcjet,
2) thrust-to-power ratio shall be better than 100 mN/kW,
3) accumulated operational time shall be longer than 900 hours,
4) total number of restarts shall be more than 1700,
5) capability to apply to both long continuous operation and to quick stabilization for short operation within 30 min. and,
6) throttling or modulation capability is highly desirable.

Acknowledgement

The authors would like to express their thanks to Mr. Yukio Shimizu of the Institute of Space and Astronautical Science and to Dr. Hiroshi Suzuki of Ishikawajima-harima Heavy Industries Co., Ltd. for fruitful discussions on development target for DC arcjet propulsion system.

References

[1] http://www.ee.surray.ac.uk/Personal/L.Wood/constellations /tables/tables.html

[2] http://www.skybridgesatellite.com/system/cont_41.htm

[3] R.C. Reynolds and J.P. Loftus, "A Hand Book to Support the NASA Policy to Limit Orbital Debris Generation", IIA.3.3-93-700, 44th Congress of the International Astronautical Federation, Graz, Austria, 1993.