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BENEFITS OF LOW-POWER ELECTROTHERMAL PROPULSION

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ABSTRACT

Mission analyses were completed to show the benefits of low-power electrothermal propulsion systems for three classes of LEO smallsat missions. Three different electrothermal systems were considered: (1) a 40 W ammonia resistojet system, (2) a 600 W hydrazine arcjet system, and (3) a 300 W ammonia arcjet. The benefits of using two 40 W ammonia resistojet systems were analyzed for three months of drag makeup of a Shuttle-launched 100 kg spacecraft in a 297 km orbit. The two 40 W resistojets decreased the propulsion system wet mass by 50% when compared to state-of-art hydrazine monopropellant thrusters. The 600 W arcjet system was used for a 300 km sunsynchronous drag makeup mission of a 1000 kg satellite and was found to decrease the wet propulsion system mass by 30%. Finally, the 300 W arcjet system was used on a 200 kg Earth-orbiting spacecraft for both orbit transfer from 300 to 400 km, two years of drag make-up, and a final orbit raise to 700 km. The arcjet system was determined to halve the propulsion system wet mass required for that scenario as compared tohydrazine monopropellant thrusters.

INTRODUCTION

The use of high specific impulse on-board propulsion systems can be mission enhancing/enabling from both a cost and payload benefit standpoint. Advanced on-board systems can offer the same payload on a smaller launch vehicle class, increase payload for the same mission by off-loading fuel, and/or increase on-orbit life while maintaining the same propulsion system wet mass. With the ever increasing need for higher performance, GEO comsats have led the way in acceptance of advanced on-orbit systems. Electric propulsion is now a serious option for a wide range of missions due to the large on-board propulsion impacts.[1]

Hydrazine resistojet technology was first used to replace hydrazine monopropellant engines increasing the specific impulse from 225 s to 300 s.[2] This has provided a significant saving of stationkeeping fuel and a nearly equivalent savings in apogee engine fuel required to place the stationkeeping propellant in GEO. Increases in hydrazine resistojet performance beyond 300 s Isp is limited by fundamental heater materials properties. The maximum enthalpy, and hence the specific impulse, is of the propellant is directly tied to the maximum heater temperature. The arcjet overcame this limitation and was the next revolutionary technology to be accepted operationally.[3,4]

Arcjets use an electric arc to increase the bulk temperature of the gas to temperatures exceeding the melting temperature of the nozzle. This heating scheme permits specific impulse levels in the range of 400 s to 650 s for hydrazine and ammonia, significantly above those attainable by resistojets. With the increase in performance comes an increase in system complexity. Resistojets run directly from the spacecraft bus with power conditioning limited to operation of a heater. Arcjets require a power processing unit to condition the power from the bus to meet arcjet ignition and steady-state operation conditions. For many satellites, especially large GEO comsats, the increased complexity and cost is offset by propellant savings.[5]

The niche for electrothermal systems is rapidly moving away from large, power-rich spacecraft to small, power-limited ones, especially in low and mid-Earth orbits (LEO/MEO). Many small spacecraft require insertion, on-orbit control, and/or deorbit propulsion functions which at present are performed either with monopropellant hydrazine (225 s Isp) or high pressure cold gas nitrogen (60-70 s Isp). Higher performance electrothermal systems offer a cost effective, higher specific impulse alternative to enhance or enable a range of missions. When compared to other classes of electric propulsion devices, electrothermal systems provide significantly greater thrust, allowing lower orbit maintenance. Although the specific impulse is lower than electrostatic systems, the dry system mass of electrothermal systems is significantly lower as is the system complexity and cost. This paper demonstrates the projected benefits of three low-power electrothermal thrusters for three Earth orbit satellites with on-board power levels of 100 to 600 W, typical of many planned smallsat applications. First, two 40 W-class ammonia resistojets were used for drag makeup and to extend the life of 100 kg-class Shuttle-launched Get-Away-Special Canister payloads in 300 km orbits. Next, a 600 W hydrazine arcjet system was used for orbit maintenance of a 1000 kg satellite in a 300 km orbit. Finally, a 300 W ammonia arcjet was used to replace a chemical system on a Clementine-class LEO spacecraft and performs orbit raising of the 200 kg spacecraft from 300 km to 400 km, followed by two years of orbit maintenance, and then a final orbit raise to 700 km to extend mission life.

MISSION MODELING METHODOLOGY AND ASSUMPTIONS

For the mission analysis performed for this study, a simple iterative routine was used which calculates the amount of circular orbit altitude change. The routine assumes constant drag force versus thrusting force over a circular orbit. Shading is included where applicable and reduces the constant thrust over the orbit by the shadow fraction. The atmospheric density is assumed not to vary temporally.

In practice, impulsive devices, such as SOA monopropellant thrusters, may employ several perigee and apogee burns to achieve the higher orbit with the thrusters pointed in the circumferential direction at the apogee and perigee, respectively. The continuous thrust electrothermal thrusters need to be pointed continuously in the circumferential direction shown in Figure 1 (normal to the radius vector and in the orbit plane). Power profiles were not coupled to the S/C orientation in these analyses. Consequently, for the continuous thrust devices, the spacecraft is assumed to be Earth-pointing to allow for circumferential thrusting.

All three mission examples deal with LEO spacecraft whose orbit altitude will be significantly influenced by atmospheric drag. For the first two examples, drag makeup is the primary mission. In the last, orbit raising is included. For the electrothermal devices circular thrusting orbits are assumed. For the state-of-art monopropellant thrusters Hohmann impulsive transfers are assumed.

The atmospheric density model used in these analyses is based on either the F10.7 = 150 or 250×10^{-22} W/(m^2 Hz) index to represent an all-time average and a solar maximum atmospheric density, respectively.[6] Of the last five solar maximum years only one had a monthly mean radio flux at F10.7 cm over 250×10^{-22} W/(m^2 Hz) and that only shortly peaked at 290×10^{-22} W/(m^2 Hz). The other four solar maximum years had peaks below 250×10^{-22} W/(m^2 Hz). The F10.7 = 150×10^{-22} W/(m^2 Hz) atmospheric density is roughly average for all the years of maximums and minimums. The density model is based on the F10.7 index atmosphere calculated with the DENS code. For example, the F10.7 = 250×10^{-22} W/(m^2 Hz) atmosphere predicts a density of 5.2e-11 kg/m^3 for a 300 km circular orbit.

MISSION I: DRAG MAKEUP FOR A 100 kg LEO SMALLSAT

The spacecraft assumed for this mission is the NASA Spartan-LITE, a 100 kg LEO smallsat. The Spartan LITE spacecraft is deployed from the Shuttle Get-Away-Special Canister at 297 km and requires a three month life even during solar maximum years. The spacecraft is cylindrical in shape with approximate dimensions 46 cm in diameter and 96 cm long. It also has eight 66 cm by 18 cm solar arrays which deploy at one end like flower petals. Assumed orbit average available power for the propulsion system is 100 W. The spacecraft attitude can be sun-pointing, stellar-pointing, or Earth-pointing during operations depending on user needs. The required operating altitude is also user specific.

The mission assumed is to keep the Spartan-LITE spacecraft at the 297 km shuttle drop-off orbit for three months of life before re-entry into the atmosphere. While raising the spacecraft to a higher orbit would ensure a three month lifetime (this might decrease the required propulsion system wet mass), it is assumed here that the payload requires an ~300 km orbit for operations. An operational orbit band with lower and upper allowed altitudes of 297 km and 300 km was assumed to be representative and was used for this study. Thus, the spacecraft would drift from the upper altitude due to drag and then perform a maneuver to raise the orbit. This orbit band depends on many factors including the how accurately the spacecraft's altitude is known and the level of spacecraft autonomy. The payload is assumed to be continuously Earth-pointing, and mission analysis assumptions include no shading with an average 80 W power.

The drag force on the satellite is directly attributable to the Spartan-LITE cross-sectional area a deployed. The minimum cross-sectional area is assumed to be the cylinder axis perpendicular to the velocity vector with an area of 0.44 m^2 due to the Earth pointing. The coefficient of drag is also assumed to be 2.2, a typical value.

A 300 s Isp ammonia resistojet system and a state-of-art 225 s hydrazine monopropellant propulsion system were compared for this mission. Because of the power limitations on the spacecraft, electrostatic systems were not studied. Also, a cursory look at low-power electromagnetic pulsed plasma thrusters showed the thrust was too low to overcome the drag for this mission and further analysis was not continued. For the electrothermal system, two 40 W resistojets are assumed to use 80 W of the available Spartan-LITE power. Ammonia was chosen as the propellant to eliminate hydrazine from the spacecraft. Very little technology development work has been done in the past several decades in this area. During the 1960's NASA expended significant resources toward the development of 5-20 W resistojets with flight-representative systems being demonstrated.[7] It is projected that if a small ammonia resistojet system is required in the near future, advances in materials technology, low Reynold's number flow modeling, and valve component miniaturization would lead to thrusters weighing less than 50 g, processing up to 40 W of power and achieving specific impulse levels on the order of 300 s. Recent studies into the technology including Direct Simulation Monte Carlo analyses indicate the performance estimates to be reasonable. Figure 2 shows a 20 W ammonia resistojet from the 1960's manufactured by AVCO under an early NASA program next to an advanced MOOG cold gas thruster which incorporates some of the technologies needed for resistojet miniaturization.

Both the SOA hydrazine monopropellant and ammonia resistojet propulsion system are able to maintain the ~300 km operating orbit for three months or more. The thrust force to drag force ratio of the two 40 W ammonia resistojets for the Spartan-LITE during solar maximum was over 25. (The hydrazine monopropellant had 100 times that ratio.) The results shown in Table I demonstrate the ability for both of the propulsion systems to perform the ~300 km mission with either the solar maximum 250×10^{-22} W/(m^2 Hz) index or solar average 150×10^{-22} W/(m^2 Hz) index. For either atmospheric density assumption all the ammonia resistojets are able to keep the Spartan-LITE at the ~300 km operating orbit with a duty cycle of only 2-4%. Thus 80W would need to be dedicated to the resistojets only 2-4% of the time. The use of the two ammonia resistojets instead of a SOA monopropellant thruster would reduce the wet propulsion system mass by approximately 50%. The significant benefits of the electrothermal system shown are for an assumed mission duration of three months, longer mission life requirements would increase the benefits. The lower density of the ammonia fuel would require about 30% more fuel volume, but would result in a lower cost due to the simpler ground handling.

MISSION II: DRAG MAKEUP FOR A 1000 kg LEO SPACECRAFT

For Mission II, 600 W hydrazine arcjet technology was compared to a baseline 225 s Isp hydrazine monopropellant system used for orbit maintenance for the proposed NASA ORACLE spacecraft. The mission deploys a Differential Absorption Lidar (DIAL) instrument based on NASA Langley Research Center's LASE and is currently in the planning stages. Spacecraft initial mass is estimated to be 1000 kg with a projected area of 5 m2. A 6am/6pm, 300 km sun synchronous orbit with a 2 year lifetime is desired. The required 600 W solar array is assumed to fly edge on with respect to the velocity vector, to minimize drag. Instrument contamination is an important consideration and minimizing hydrazine throughput, and hence exhaust products, is advantageous.

Mission representative lower and upper allowed altitudes were assumed to be 300 km and 310 km. This orbit band depends on many factors including the how accurately the spacecraft's altitude is known and the level of spacecraft autonomy. Mission analysis assumptions include no shading and an average 600 W of power for propulsion. The lower thrust devices need to burn continuously to complete the orbit change in the minimum time. To achieve this the thrusters need to always point in the circumferential direction which is possible.

The ORACLE mission was analyzed for both an average and solar maximum atmosphere. Both cases were evaluated assuming that the power was available from the baseline spacecraft. They were then also analyzed for the case in which an extra 600 W array, which was charged to the propulsion system, was added. Both cases assume that either the spacecraft power is available, but provisions for adding an extra 600 W array are included. A quick investigation of electrostatic systems determined that the thrust levels were marginal for mission. Ion thrusters only have slightly more thrust than the worst case drag and similarly, the Hall thruster has a little over twice the thrust to worse case drag fraction; therefore, these systems may provide unacceptable risk. The low-power electromagnetic pulsed plasma thrusters also do not have enough thrust to overcome the worst case drag. The arcjet has over four times the thrust to worst case drag and should be sufficient to overcome the drag in all cases. The 600 W arcjet system is the most mature of the electrothermal technologies presented in this paper. For the past three years NASA has supported the 600 W-class arcjet system development under the LPATS program.[8] The LPATS program leverages technology developed under previous higher power thruster programs and will appear very similar to those systems as shown in Figure 3. Within the next year that program will demonstrate a 600 W hydrazine arcjet system including thruster, 28 V input power processor, and a regulated feed system over a representative qualification envelope. The system is targeted to provide 450 s of specific impulse for a minimum of 1000 h of operation. Due to the large total impulse requirement for ORACLE-class missions, three thrusters operated in series and connected to one power processor and propellant system were used .

The results of the ORACLE mission analysis is shown in Table II. Because of its low thrust, the arcjet option has a duty cycle of 0.10 for the average atmosphere case and 0.20 for the solar maximum case. If on-board power is available for the electric propulsion system, the arcjet system reduced the wet mass by over 40%. If additional array power must be added on the spacecraft and the mass charged to the propulsion system, the arcjet would decrease the propulsion system wet mass by ~30%. Spacecraft contamination by the propulsion system if often a concern on spacecraft with high quality sensors. Reduction in the hydrazine required to complete the mission by a minimum of 50% through the use of arcjets is directly proportional to the reduction in contamination potential over the SOA monopropellant.

MISSION III: ORBIT RAISING AND DRAG MAKEUP FOR A 200 kg LEO/MEO SMALLSAT

Extended mission lifetimes for low-cost Earth-observing spacecraft are required for monitoring the environment over several cyclces of seasonal changes. Cost constraints necessitate simplicity, including small power systems and low-cost ground handling, enabled by hydrazine removal. The Mission III consists of three phases. Phase 1 is a transfer from a launch altitude of 300 km to the operational orbit of 400 km. In Phase 2 the thrusters are used for drag makeup at ~400 km by using a lift to 410 km then decay to 400 km operational band over a period of two years. Phase 3 requires a transfer from the operational 400 km orbit to a final, relatively drag-free 700 km orbit.

The spacecraft used for this analysis was derived from the Naval Research Laboratory's Clementine spacecraft. The basic spacecraft bus was used including the 360 W solar array, but the large bipropellant propulsion system was replaced with either a monopropellant hydrazine system or an ammonia arcjet. That change allows the spacecraft initial mass to be reduced to approximately 200 kg. Of the 360 W of bus power , 300 W of power is assumed available as needed for the ammonia arcjet thruster during sunlit periods only. The bus dimensions used for the analysis were 1.14 m diameter by 1.88 m length, and average cross-sectional area of 1.6 m2 is assumed during all mission phases. The performance data used for the 300 W ammonia arcjet were taken from demonstrated performance of laboratory systems. Throttleability of the thrust at constant power allowed variation of performance from 360 s Isp@0.30 efficiency to 470 s Isp@0.23 efficiency.[9] A development program similar to that of the NASA LPATS program would be required to bring this technology to flight readiness.

As in the previous missions, the analysis was bounded by using the solar average and solar maximum atmospheres as well as the two arcjet system operating points. Tables III and IV show the results of the calculations for each phase. In general, the arcjet can reduce the wet propulsion system mass by 30% to 50% by using the 360 s or 470 s Isp performance levels, respectively. Due to the throttleability of the device the arcjet can be operated at any point within that which could be useful if very large unexpected drag forces occur; although, for this mission even the lower thrust 470 s Isp performance level has roughly six times the thrust force to the drag force, which should be sufficient. The duty cycles for Phase 2 (drag makeup) range from 2-10% depending on the density model and thruster operating assumptions. Such duty cycles should not encroach to heavily upon the payloads power requirement. In fact, it may be possible to use the batteries to power the arcjet during this mission phase. The orbit transfer times range from 5 to 20 days, again depending on the density model and thruster operating assumptions. These times are small relative to the assumed mission operation time of 2 years.

CONCLUDING REMARKS

Although electrothermal resistojets have been in operation for over a decade and electrothermal arcjets are currently operational, the need for these types of systems is not waning. As GEO communications satellites continue to grow in size and power level, xenon-based electrostatic systems are become more attractive and the economic advantage significant. The niche for electrothermal systems is becoming small, low-power LEO spacecraft.

The benefits of three types of new electrothermal on-board propulsion technologies were shown. For a 100 kg Shuttle-launched GAS-Can spacecraft the propulsion system wet mass can be cut by 50% for a three month orbit maintenance at 297 km or the mission life can be significantly extended via the use of 40 W ammonia resistojets. Similarly, a 600 W hydrazine arcjet system providing the same drag makeup function on a 1000 kg LEO spacecraft decreases the propulsion system wet mass by approximately 30%. Finally in a combined orbit raise from 300 km to 400 km, followed by 2 y of drag makeup and a final orbit raise to 700 km of a 200 kg LEO spacecraft the 300 W ammonia arcjet was found to decrease the propulsion system wet mass by up to 50% over state-of-art hydrazine monopropellant thrusters.

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	solar avg			solar max		
Continuous or Impulsive Thrust	Continuous	none	Impulsive	Continuous	none	Impulsive
Thrust Calculations	NH3 RJ	drag only	SOA MonoProp	NH3 RJ	drag only	SOA MonoProp
Isp	300 sec		225 sec	300 sec		225 sec
Overall Efficiency	0.70		1.00	0.70		1.00
Thruster(s) Power Level	80 W			80 W		
Per Thruster Mass	0.05 kg		0.33 kg	0.05 kg		0.33 kg
Per PPU Mass	0.3 kg/kW			0.3 kg/kW		
Fixed Propellant Sys Mass	0.5 kg		1.82 kg	0.5 kg		1.82 kg
Propellant Tankage Fraction	0.070		0.072	0.070		0.072
Propellant density	0.60 g/cc		1.00 g/cc	0.60 g/cc		1.00 g/cc
# of thrusters	2		1	2		1
Engine Thrust	38.1 mN		4450 mN	38.1 mN		4450 mN
Engine Mass Flow Rate	12.9 mg/s		2020 mg/s	12.9 mg/s		2020 mg/s
Initial Satellite Mass	114 kg	114 kg	114 kg	114 kg	114 kg	114 kg
Drag Calculations						
F10.7cm Radio Flux Index	150	150	150	250	250	250
Cross-Sectional Area	0.44 m^2	0.44 m^2	0.44 m^2	0.44 m^2	0.44 m^2	0.44 m^2
Coefficient of Drag	2.2	2.2	2.2	2.2	2.2	2.2
Starting Altitude	297 km	300 km	297 km	297 km	300 km	297 km
Desired final circ alt	300 km	297 km	300 km	300 km	297 km	300 km
Initial Drag Force	0.81 mN	0.76 mN	0.81 mN	1.6 mN	1.5 mN	1.6 mN
Approx Time to 297 km circ with NO THRUST		3.0 d			1.5 d	
Approx Thrust and Coast Time	0.06 d		1 min	0.06 d		1 min
Duty Cycle	0.02		n/a	0.02		n/a
Total Impulse/ burn(no shade)	207 N-s		198 N-s	207 N-s		198 N-s
Dry propulsion system mass	0.6 kg		2.2 kg	0.6 kg		2.2 kg
Mission Equivalent ΔV/burn	1.8 m/s		1.7 m/s	1.8 m/s		1.7 m/s
3 month wet system mass	2.9 kg		5.1 kg	4.9 kg		7.8 kg

Table I. Spartan-LITE Drag Makeup Mission Assumptions and Results

Continuous or Impulsive Thrust	solar avg Continuous	solar avg none	solar avg Impulsive	solar max Continuous	solar max none	solar max Impulsive
Thrust Calculations	Arcjet	drag only	SOA MonoProp	Arcjet	drag only	SOA MonoProp
Isp	450 s		225 s	450 s		225 s
Overall Efficiency	0.32		1.00	0.32		1.00
Thruster(s) Power Level	570 W			570 W		
Per Thruster Mass	1.00 kg		0.33 kg	1.00 kg		0.33 kg
Per PPU Mass	7.0 kg/kW			7.0 kg/kW		
Fixed Propellant Sys Mass	1.0 kg		1.82 kg	1.0 kg		1.82 kg
Propellant Tankage Fraction	0.070		0.072	0.070		0.072
Propellant density	1.00 g/cc		1.00 g/cc	1.00 g/cc		1.00 g/cc
# of thrusters	3		2	3		2
Engine Thrust	81.8 mN		4450 mN	81.8 mN		4450 mN
Engine Mass Flow Rate	18.5 mg/s		2020 mg/s	18.5 mg/s		2020 mg/s
Initial Satellite Mass	1000 kg	1000 kg	1000 kg	1000 kg	1000 kg	1000 kg
Drag Calculations						
F10.7cm Radio Flux Index	150	150	150	250	250	250
Cross-Sectional Area	5.00 m^2	5.00 m^2	5.00 m^2	5.00 m^2	5.00 m^2	5.00 m^2
Coefficient of Drag	2.2	2.2	2.2	2.2	2.2	2.2
Starting Altitude	300 km	310 km	300 km	300 km	310 km	300 km
Desired final circ alt	310 km	300 km	310 km	310 km	300 km	310 km
InitialDrag Force	8.7 mN	7.0 mN	8.7 mN	17 mN	14 mN	17 mN
Approx Time to 300 km circ with NO THRUST		8.6 d			4.3 d	
Approx Thrust and Coast Time	0.9 d		22 min	1.1 d		22 min
Total Impulse(worst case shade)	481225N-s		438624 N-s	911329 N-s		785496 N-s
Total Mission Equivalent ΔV	510 m/s		489 m/s	1021 m/s		971 m/s
Duty Cycle	0.10		<.01	0.20		<.01
Dry propulsion system mass (less tanks)	8.0 kg		2.5 kg	8.0 kg		2.5 kg
2.0 yr wet system mass	125 kg		216 kg	229 kg		384 kg
Extra Array Option (no duty cycle)						
EP Solar Array Power level	570 W			570 W		
EP Solar Array Mass	30 kg			30 kg		
2.0 yr wet system mass	155 kg			259 kg		

Table II. ORACLE Drag Makeup Mission Assumptions and Results-Solar Maximum Case

	low Isp			high Isp						
Continuous or Impulsive Thrust	Continuous	Continuous	Continuous	Continuous	Continuous	Continuous	none	Impulsive	Impulsive	Impulsive
Thrust Calculations	NH3 Arcjet	NH3 Arcjet	NH3 Arcjet	NH3 Arcjet	NH3 Arcjet	NH3 Arcjet	drag only	SOA MonoProp	SOA MonoProp	SOA MonoProp
Isp	360 sec	360 sec	360 sec	470 sec	470 sec	470 sec		225 sec	225 sec	225 sec
Overall Efficiency	0.30	0.30	0.30	0.23	0.23	0.23		1.00	1.00	1.00
Thruster(s) Power Level	301 W	301 W	301 W	301 W	301 W	301 W				
Per Thruster Mass	0.25 kg	0.25 kg	0.25 kg	0.25 kg	0.25 kg	0.25 kg		0.33 kg	0.33 kg	0.33 kg
Per PPU Mass	3.3 kg/kW	3.3 kg/kW	3.3 kg/kW	3.3 kg/kW	3.3 kg/kW	3.3 kg/kW				
Fixed Propellant Sys Mass	0.5 kg	0.5 kg	0.5 kg	0.5 kg	0.5 kg	0.5 kg		1.82 kg	1.82 kg	1.82 kg
Propellant Tankage Fraction	0.070	0.070	0.070	0.070	0.070	0.070		0.072	0.072	0.072
Propellant density	0.60 g/cc	0.60 g/cc	0.60 g/cc	0.60 g/cc	0.60 g/cc	0.60 g/cc		1.00 g/cc	1.00 g/cc	1.00 g/cc
# of thrusters	1	1	1	1	1	1		1	1	1
Engine Thrust	50.97 mN	50.97 mN	50.97 mN	30.36 mN	30.36 mN	30.36 mN		4450 mN	4450 mN	4450 mN
Engine Mass Flow Rate	14.4 mg/s	14.4 mg/s	14.4 mg/s	6.59 mg/s	6.59 mg/s	6.59 mg/s		2020 mg/s	2020 mg/s	2020mg/s
Initial Satellite Mass	200 kg	196 kg	165 kg	200 kg	197 kg	173 kg	200 kg	200 kg	195 kg	148 kg
Drag Calculations										
F10.7cm Radio Flux Index	250	250	250	250	250	250	250	250	250	250
Cross-Sectional Area	1.6 m^2	1.6 m^2	1.6 m^2	1.6 m^2	1.6 m^2	1.6 m^2	2.74 m^2	1.6 m^2	1.6 m^2	1.6 m^2
Coefficient of Drag	2.2	2.2	2.2	2.2	2.2	2.2	2.2	2.2	2.2	2.2
Starting Altitude	300 km	400 km	400 km	300 km	400 km	400 km	410 km	300 km	400 km	400 km
Desired final circ alt	400 km	410 km	700 km	400 km	410 km	700 km	400.0 km	400 km	410 km	700 km
Initial Drag force	5.4 mN	1.2 mN	1.2 mN	5.4 mN	1.2 mN	1.2 mN	1.8 mN	5.4 mN	1.2 mN	1.2 mN
Approx Time to 400 km circ with NO THRUST							6.8 d			
Approx Thrust and Coast Time	4.8 d	0.5 d	9.7 d	8.6 d	0.8 d	17.4 d		42 min	4 min	88 min
Duty Cycle		0.06			0.10				n/a	
Toal Impulse (per raise)	12600 N-s	1210 N-s	26800 N-s	13500 N-s	1230 N-s	28500 N-s		11300 N-s	1100 N-s	23400 N-s
Mission Equivalent ΔV (per raise)	64 m/s	6 m/s	167 m/s	68 m/s	6 m/s	168 m/s		57 m/s	6 m/s	164 m/s
Dry propul. sysmass (less tanks)	1.8 kg	1.8 kg	1.8 kg	1.8 kg	1.8 kg	1.8 kg		2.2 kg	2.2 kg	2.2 kg
Wet propulsion system mass (per raise)	5.3 kg	2.1 kg	9.3 kg	4.7 kg	2.0 kg	7.9 kg		7.3 kg	2.6 kg	12.8 kg
2.0 yr wet system mass		36 kg			28 kg				35 kg	
	Total Mission Wet Mass	48 kg		Total Mission Wet Mass	38 kg			Total Mission Wet Mass	69 kg	

Table III. Clementine-Derived Spacecraft Drag Makeup Mission Assumptions and Results- Solar Maximum Case

	low Isp			high Isp						
Continuous or Impulsive Thrust	Continuous	Continuous	Continuous	Continuous	Continuous	Continuous	none	Impulsive	Impulsive	Impulsive
Thrust Calculation	NH3 Arcjet	NH3rcjet	NH3 Arcjet	NH3 Arcjet	NH3 Arcjet	NH3 Arcjet	drag only	SOA MonoProp	SOA MonoProp	SOA MonoProp
Isp	360 sec	360 sec	360 sec	470 sec	470 sec	470 sec		225 sec	225 sec	225 sec
Overall Efficiency	0.30	0.30	0.30	0.23	0.23	0.23		1.00	1.00	1.00
Thruster(s) Power Level	301 W	301 W	301 W	301 W	301 W	301 W				
Per Thruster Mass	0.25 kg	0.25 kg	0.25 kg	0.25 kg	0.25 kg	0.25 kg		0.33 kg	0.33 kg	0.33 kg
Per PPU Mass	3.3 kg/kW	3.3 kg/kW	3.3 kg/kW	3.3 kg/kW	3.3 kg/kW	3.3 kg/kW				
Fixed Propellant Sys Mass	0.5 kg	0.5 kg	0.5 kg	0.5 kg	0.5 kg	0.5 kg		1.82 kg	1.82 kg	1.82 kg
Propellant Tankage Fraction	0.070	0.070	0.070	0.070	0.070	0.070		0.072	0.072	0.072
Propellant density	0.60 g/cc	0.60 g/cc	0.60 g/cc	0.60 g/cc	0.60 g/cc	0.60 g/cc		1.00 g/cc	1.00 g/cc	1.00 g/cc
# of thrusters	1	1	1	1	1	1		1	1	1
Engine Thrust	51.0 mN	51.0 mN	51.0 mN	30.4 mN	30.4 mN	30.4 mN		4450 mN	4450 mN	4450 mN
Engine Mass Flow Rate	14.4 mg/s	14.4 mg/s	14.4 mg/s	6.59 mg/s	6.59 mg/s	6.59 mg/s		2020 mg/s	2020 mg/s	2020 mg/s
Initial Satellite Mass	200 kg	197 kg	184 kg	200 kg	197 kg	187 kg	200 kg	200 kg	195 kg	176 kg
Drag Calculation										
F10.7cm Radio Flux Index	150	150	150	150	150	150	150	150	150	150
Cross-Sectional Area	1.6 m^2	1.6 m^2	1.6 m^2	1.6 m^2	1.6 m^2	1.6 m^2	2.74 m^2	1.6 m^2	1.6 m^2	1.6 m^2
Coefficient of Drag	2.2	2.2	2.2	2.2	2.2	2.2	2.2	2.2	2.2	2.2
Starting Altitude	300 km	400 km	400 km	300 km	400 km	400 km	410 km	300 km	400 km	400 km
Desired final circ alt	400 km	410 km	700 km	400 km	410 km	700 km	400 km	400 km	410 km	700 km
Initial Drag Force	2.78 mN	0.45 mN	0.45 mN	2.78 mN	0.45 mN	0.45 mN	0.66 mN	2.78 mN	0.45 mN	0.45 mN
Approx Time to 400 km circ with NO THRUST							18.3 d			
Approx Thrust and Coast Time	4.5 d	0.5 d	10.8 d	7.8 d	0.8 d	18.6 d		42 min	4 min	104 min
Duty Cycle		0.02			0.04				n/a	
Total Impulse(per raise)	12000 N-s	1210 N-s	29700 N-s	12200 N-s	1230 N-s	30400 N-s		11300 N-s	1100 N-s	27900 N-s
Mission Equivalent ΔV (per raise)	60 m/s	6 m/s	166 m/s	62 m/s	6 m/s	165 m/s		57 m/s	6 m/s	164 m/s
Dry propul. sysmass (less tanks)	1.8 kg	1.8 kg	1.8 kg	1.8 kg	1.8 kg	1.8 kg		2.2 kg	2.2 kg	2.2 kg
Wet propulsion system mass (per raise)	5.1 kg	2.1 kg	10.2 kg	4.4 kg	2.0 kg	8.4 kg		7.3 kg	2.6 kg	14.8 kg
2.0 yr wet system mass		16 kg			12 kg				19 kg	
	Total Mission Wet Mass	28 kg		Total Mission Wet Mass	22 kg			Total Mission Wet Mass	41 kg	

Table IV. Clementine-Derived Spacecraft Drag Makeup Mission Assumptions and Results- Solar Average Case

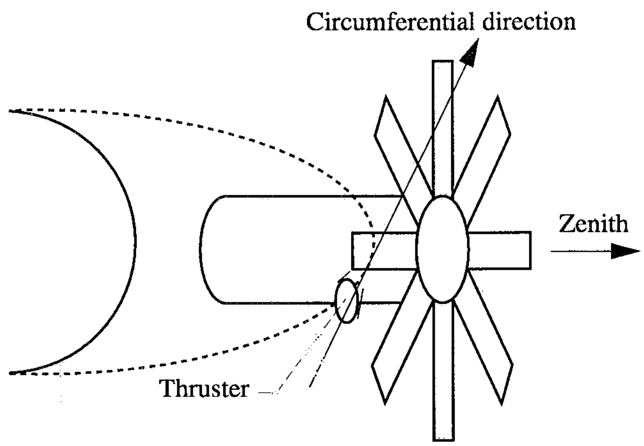

Figure 1. Orbital configuration.

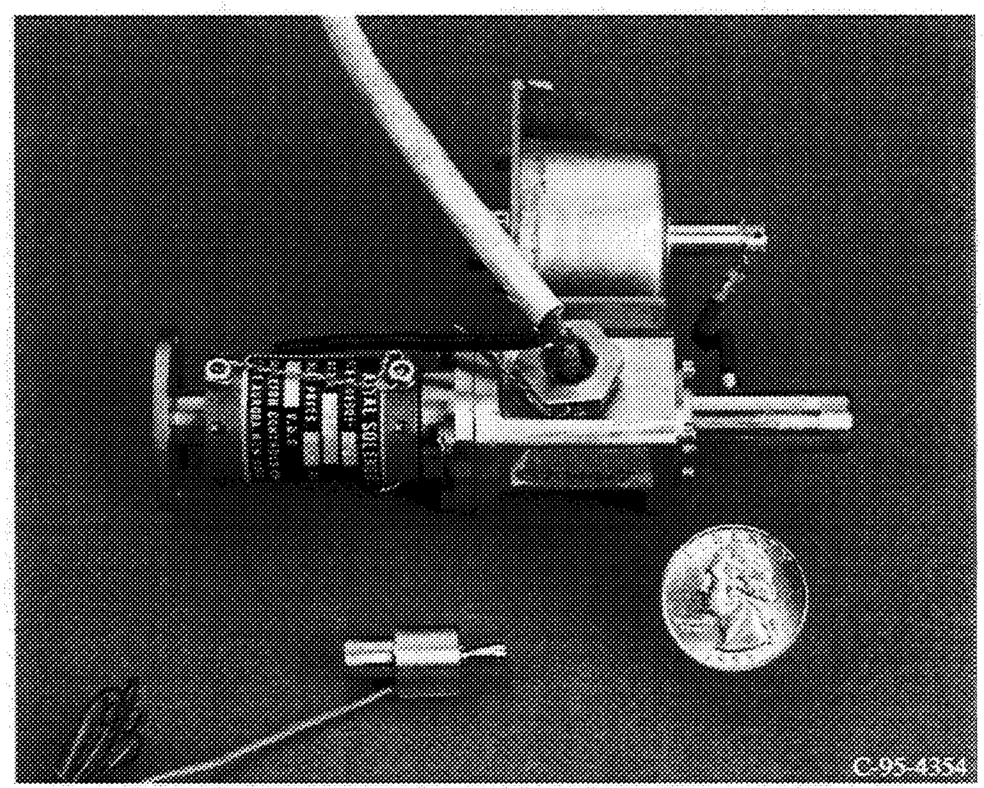

Figure 2. AVCO 20-W class ammonia resistojet (circa 1965) with advanced MOOG cold gas thruster.

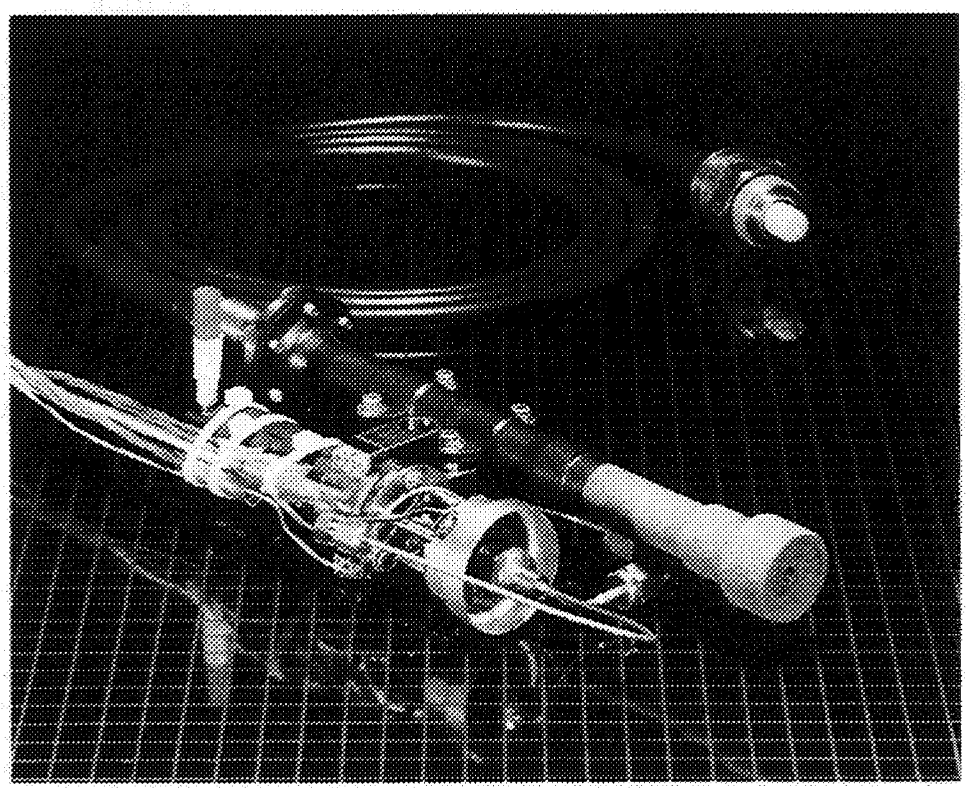

Figure 3. Olin Aerospace Co. 2 kW-class hydrazine arcjet.

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13. ABSTRACT ***(Maximum 200 words)***

Mission analyses were completed to show the benefits of low-power electrothermal propulsion systems for three classes of LEO smallsat missions. Three different electrothermal systems were considered: (1) a 40 W ammonia resistojet system, (2) a 600 W hydrazine arcjet system, and (3) a 300 W ammonia arcjet. The benefits of using two 40 W ammonia resistojet systems were analyzed for three months of drag makeup of a Shuttlemass by 50% when compared to state-of-art hydrazine monopropellant thrusters. The 600 W arcjet system was used for a 300 km sunsynchronous drag makeup mission of a 1000 kg satellite and was found to decease the wet propulsion system mass by 30%. Finally, the 300 W arcjet system was used on a 200 kg Earth-orbiting spacecraft for both orbit transfer from 300 to 400 km, two years of drag make-up, and a final orbit raise to 700 km. The arcjet system was determined to halve the propulsion system wet mass required for that scenario as compared to hydrazine monopropellant thrusters.

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