Section 7: Implications of Maneuvering for Satellite Mass

The previous section showed that the laws of orbital mechanics determine the amount of velocity change DV required for a satellite to carry out various types of maneuvers (see Table 6.1). This section shows that the mass of propellant a satellite needs to change its speed by DV increases rapidly with DV. Placing a heavy object in orbit is technologically more difficult than placing a light object. Moreover, the cost of putting a satellite into orbit increases roughly in proportion to its overall mass. These launch factors place practical limits on the amount of propellant a satellite can carry and thus on the amount of maneuvering it can carry out.

The relationship between maneuvering and propellant mass has important implications for space missions such as the proposed military space plane. The space plane is envisioned as a vehicle that, after being launched into orbit, maneuvers to accomplish a variety of tasks. These might include placing satellites or ground-attack weapons in orbit or rendezvousing with satellites to inspect, repair, or refuel them. However, limits on the mass of propellant that can be launched with such a vehicle places strict limits on how much maneuvering the vehicle could do.

Similarly, the amount of maneuvering a reconnaissance satellite could do for either offensive or defensive purposes is limited by the amount of propellant it carries. Section 9 discusses some of these consequences.

Section 6 showed that the ΔV required for a particular maneuver is determined by physics. However, the propellant mass required to provide that ΔV depends on how efficiently the thruster can use propellant to bring about a velocity change, which depends on the thruster technology.[1] Most of the calculations in this paper assume conventional thrusters using chemical propellant, for reasons explained below.[2] Other thruster technologies are also discussed, along with their applications and limitations.

SATELLITE MASS

The relationship between ΔV, the mass of propellant M_p needed to impart ΔV to this satellite, and the satellite mass M_s (which does not include M_p)[3] is given by the so-called *rocket equation*, which the Appendix to Section 7 discusses in detail.

1. The thruster efficiency is sometimes expressed in terms of its *specific impulse*. It can be expressed equivalently as the exhaust velocity V_e of the particles ejected from the thruster, which is the term used in this paper.

2. We use a value of V_e = 3 km/s for conventional thrusters.

3. The mass M_s of the satellite may include propellant for purposes such as stationkeeping and maneuvers other than the one being considered here.

Table 7.1 lists several values of ΔV and the corresponding values of M_p/M_s, assuming conventional propulsion technology. (The equations used to calculate these values are given in the Appendix to Section 7.) For example, to carry out a maneuver requiring a ΔV of 2 km/s, the propellant mass M_p required for this maneuver is 0.9 times that of the satellite itself, that is, the propellant nearly doubles the total mass that must be placed in orbit. In other words, a satellite with a mass of one ton (excluding the propellant for this maneuver) would need to carry 0.9 tons of propellant to provide the ΔV for this maneuver.

This mass penalty increases rapidly as ΔV increases, as Figure 7.1 and Table 7.1 show. To carry out a maneuver requiring a ΔV of 5 km/s (or several maneuvers that added up to 5 km/s), a one-ton satellite would need to carry 4.3 tons of propellant to conduct this maneuver. For a maneuver (or set of maneuvers) requiring a ΔV of 10 km/s, that same satellite would need to carry 27 tons of propellant.

Figure 7.1. This figure shows the ratio of propellant mass to satellite mass M_p/M_s required to produce a given velocity change (ΔV) assuming conventional propulsion (V_e = 3 km/s). M_s is the mass of the satellite excluding the propellant mass M_p.

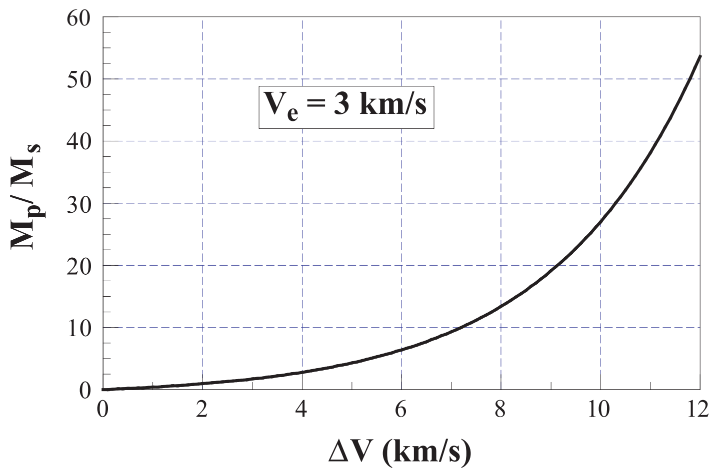

Table 7.1. Selected values of the ratio of propellant mass to satellite mass M_p/M_s required to produce a given velocity change (ΔV) assuming conventional propulsion (V_e = 3 km/s).

ΔV (km/s)	M_p/M_s
1	0.4
2	0.9
3	1.7
4	2.8
5	4.3
8	13.4
10	27.0
12	53.6

Note that the propellant mass needed to deliver a given ΔV and carry out a given maneuver depends on the total satellite mass at the time the maneuver takes place. Therefore, if a satellite is carrying a large amount of propellant for multiple maneuvers, the initial maneuvers require proportionately more propellant since the mass that must be accelerated is that of the satellite plus the remaining propellant.

Table 7.2 gives the ratio of propellant mass to satellite mass required by the maneuvers considered in Section 6 (assuming conventional thruster technology on the satellite). These numbers were calculated using the rocket equation (see the Appendix to Section 7).

Table 7.2. This table gives the ratio of propellant mass to satellite mass M_p/M_s required for the space activities listed in Table 6.1, assuming conventional propulsion (V_e = 3 km/s). M_s is the mass of the satellite excluding the propellant mass M_p.

Type of Satellite Maneuver	Required ΔV (km/s)	M_p/M_s
Changing orbital altitude within LEO (from 400 to 1,000 km)	0.3	0.1
Stationkeeping in GEO over 10 years	0.5–1	0.2–0.4
De-orbiting from LEO to Earth	0.5–2	0.2–1
Changing inclination of orbital plane in GEO		
by $\Delta\theta$ = 30°	2	1
by $\Delta\theta$ = 90°	4	3
Changing orbital altitude from LEO to GEO (from 400 to 36,000 km)	4	3
Changing inclination of orbital plane in LEO		
by $\Delta\theta$ = 30°	4	3
by $\Delta\theta$ = 90°	11	38

THE POTENTIAL IMPACT OF NEW TECHNOLOGIES

How much propellant the satellite needs to produce the necessary ΔV depends on the thruster technology. The basic physics of rocket thrusters of all types is the same: a power source accelerates the propellant material to high speed and ejects it in a specific direction as exhaust, which propels the satellite in the direction opposite to that of the exhaust. The amount of thrust produced by this process depends on the speed of the particles in the exhaust (the *exhaust velocity*) and the amount of mass the thruster can eject every second (the *mass flow rate*).

The mass ratios given in the tables above are for conventional thruster technologies fueled by chemical propellants. These are by far the most prevalent and will remain so for many applications.

New technologies that use propellant more efficiently are being developed, including electric arcjet thrusters and electric ion thrusters. However, these systems produce much less thrust than conventional thrusters. While these thrusters require less propellant mass to produce a given ΔV, the thruster must operate for a much longer time to produce that ΔV. Ion thrusters, for example, have exhaust velocities 10 to 20 times higher than chemical thrusters, but currently their mass flow rates are many thousands of times smaller. As a result, these engines produce thrust levels thousands of times less than conventional thrusters, which would require them to operate thousands of times longer than a conventional engine to bring about the same ΔV (see below).

Such thrusters can be practical for an application such as stationkeeping, which does not need to occur rapidly. But low-thrust engines are not appropriate for missions that require a rapid response, such as ballistic missile defense and other military missions. For such applications, chemical thrusters remain the only practical choice for the foreseeable future.

For the longer term, NASA is considering various types of nuclear propulsion for its spacecraft. While such engines may produce higher thrust, they are unlikely to be used near the Earth for the applications of interest in this report.

Below we provide more information on conventional thrusters and several new thruster technologies.[4]

Conventional Thrusters

In a conventional thruster, the power source is a chemical reaction, which heats the propellant to high temperatures. A nozzle directs the hot gases so that they provide thrust efficiently. Conventional chemical thrusters have moderate values of exhaust velocity (up to 3 or 4 km/s), but can have large mass flow rates that give rise to large thrust forces. For example, conventional thrusters used on satellites produce thrusts of several hundred or even several thousand Newtons (N).[5,6]

Electric Arcjet Thrusters

This technology provides a way of improving the efficiency above that of conventional thrusters and increasing the exhaust velocity to above 5 km/s. These systems use an arcjet to superheat the propellant before it is burned, which increases the efficiency of the process. However, as with ion thrusters, the thrusts that can currently be produced by these systems are small—less than a Newton. Such thrusters were first used on satellites for stationkeeping in

4. For an overview of electric propulsion technologies, see Alec D. Gallimore, "The Past and Future of Rocket Engine Technologies," http://www.fathom.com/course/21701743/index.html, accessed January 20, 2005.

5. A Newton is a unit of force, with dimensions of $kg\text{-}m/s^2$.

6. For examples of thrusters developed for this use, see the EADS Space Transportation website http://cs.space.eads.net/sp/, accessed January 20, 2005. An example of a large upper stage engine is the Fregat upper stage of the Soyuz vehicle, which has a thrust of 19.4 kN; see Starsem, "Soyuz User's Manual," April 2001, http://www.starsem.com/services/images/soyuz_users_manual_190401.pdf, accessed January 20, 2005.

1993. Compared with a conventional thruster with an exhaust velocity of 3 km/s, a thruster with an exhaust velocity of 5 km/s could reduce the amount of stationkeeping propellant required[7] by 40%.

Ion Thrusters

The main alternative to conventional propulsion now being developed is electric ion thrusters. A number of varieties are under development, but all work on the principle of creating charged ions that are accelerated to high speed by an electric field. This method can produce high exhaust velocities—values 10 to 20 or more times those provided by conventional thrusters have been achieved. However, the mass flow rate is many thousands of times smaller than that produced by conventional thrusters, and their thrust levels are still typically less than a Newton. An ion thruster was first used on a commercial satellite in 1997.[8]

An example of the ion engine is the Xenon-Ion Propulsion System (XIPS) engines used on Boeing 702 communications satellites for stationkeeping and changing altitude.[9] It has an exhaust velocity of 35 km/s, which is more than ten times as great as conventional thrusters. However, the mass flow is only 5×10^{-6} kg/s, providing a maximum thrust of 0.165 N.

Each Boeing satellite reportedly operates a set of four of these engines for 30 minutes per day for stationkeeping, using 5 kg of propellant per year, or a total of 50 to 75 kg over the 10- to 15-year lifespan of the satellite.[10] A conventional thruster with a V_e of 3 km/s on a satellite of the same mass would require 10 to 15 times as much propellant mass to provide the same ΔV.

However, ion engines require high power: the XIPS engine uses 4.5 kW of power. This power is supplied by the solar panels of the Boeing 702 satellites, which deliver 10 to 15 kW of power.

The High Power Electric Propulsion (HiPEP) engine currently being developed by NASA is also an ion engine.[11] Because this engine is intended to operate at up to 50 kW and NASA plans to use it in spacecraft that operate far from the Sun, it is being developed as part of the nuclear electric propulsion (NEP) programs under Project Prometheus.

The HiPEP engine has demonstrated high exhaust velocities, but as with other ion engines, produces low thrust. In an initial test in November 2003, the HiPEP engine demonstrated exhaust velocities from 60 to 80 km/s and

7. Nelson.

8. Nelson.

9. Technical information on the XIPS thruster is available on the Boeing website at http://www.boeing.com/defense-space/space/bss/factsheets/xips/xips.html, accessed January 20, 2005 and http://www.boeing.com/ids/edd/ep.html, accessed January 20, 2005.

10. "Boeing 702 Fleet," http://www.boeing.com/defense-space/space/bss/factsheets/702/702fleet.html, accessed January 20, 2005.

11. NASA Glenn Research Center, "High Power Electric Propulsion Program (HiPEP)," http://www.grc.nasa.gov/WWW/ion/present/hipep.htm, accessed January 20, 2005.

operated at power levels up to 12 kW.[12] A February 2004 test apparently operated at 34 kW, with an exhaust velocity of 95 km/s. The thrust generated in this test was 0.6 N.[13]

To illustrate the difference in time required for the same maneuver using different thrusters, consider a maneuver that requires a ΔV of 1 km/s. A conventional thruster (with a thrust of 1,000 N and exhaust velocity of 3 km/s) would need to operate for about 4 minutes to execute this maneuver. The current generation XIPS engine would need to operate for 2 weeks, and the advanced HiPEP engine for 4 days (see the Appendix to Section 7 for details).[14]

Nuclear Propulsion

NASA is working, under Project Prometheus, on several projects related to nuclear power in space and is considering several others.[15] A primary goal is to develop electric power sources as alternatives to solar power for spacecraft operating far from the Sun. One project is developing generators containing radioisotopes to produce relatively low levels of electric power (hundreds of watts) for spacecraft. Generators of this type use plutonium-238, and versions have been flown in previous NASA spacecraft. A second project under consideration would develop a uranium-fueled nuclear reactor for use in space that would produce electric power for electric propulsion and other systems on the spacecraft. Current discussions call for a reactor capable of producing 100 kW of electricity.[16]

A longer-term focus is the nuclear thermal propulsion program, which would use the heat from the nuclear reactor to heat a propellant and create high thrust.

12. NASA, "NASA Successfully Tests Ion Engine," Press Release 03-377, November 20, 2003, http://www.nasa.gov/home/hqnews/2003/nov/HQ_03377_ion_engine.html, accessed January 20, 2005.

13. Alan Newhouse, "Project Prometheus: Program Overview," briefing slides, April 29, 2004, http://www.spacecongress.org/2004/Other/Newhouse.pdf, accessed January 20, 2005.

14. This example assumes a satellite mass, without propellant, of 200 kg.

15. Project Prometheus, http://spacescience.nasa.gov/missions/prometheus.htm, accessed March 1, 2004.

16. Ben Iannotta, "Jupiter Moon Probe Goes Nuclear," *Aerospace America*, March 2004, http://www.aiaa.org/aerospace/Article.cfm?issuetocid=474&ArchiveIssueID=50, accessed January 20, 2005, and Newhouse, "Project Prometheus."

Section 7 Appendix: The Rocket Equation

One of the most basic and important equations of orbital dynamics is the rocket equation, which relates the mass of propellant required to impart a given ΔV to a satellite of given mass, for a given thruster technology. The equation is a direct consequence of conservation of momentum (see below for derivation). The equation can be written in several useful forms:

$$\Delta V = V_e \ln\left(\frac{M_i}{M_f}\right) \tag{7.1}$$

$$\frac{M_i}{M_f} = e^{\Delta V/V_e} \tag{7.2}$$

where M_i and M_f are the initial and final mass of the satellite before and after the thruster operates, and V_e is the exhaust velocity of the rocket motor providing the thrust, or the average speed at which the mass in the exhaust is ejected from the thruster. In the process of bringing about this maneuver, the thruster burns a mass $M_p = M_i - M_f$ of propellant. The final mass includes the mass of the unfueled satellite and of any additional propellant it carries. From Equation 7.2, the mass of propellant M_p required for a maneuver of ΔV for a satellite can be written in terms of the initial mass M_i, as

$$M_p = M_i(1 - e^{-\Delta V/V_e}) \tag{7.3}$$

or in terms of the final mass M_f, as

$$M_p = M_f(e^{\Delta V/V_e} - 1) \tag{7.4}$$

The exhaust velocity is often expressed in terms of specific impulse, I_{sp}:

$$V_e = g_0 I_{sp} \tag{7.5}$$

where g_0 is the acceleration of gravity at the Earth's surface (9.81 m/s^2). The thrust T of a rocket engine is given by

$$T = V_e \frac{dM}{dt} \tag{7.6}$$

where dM/dt is the mass flow rate of propellant.

Equation 7.6 shows that even if V_e is large, the thrust will be small if the mass flow is small. This is the situation discussed for ion thrusters in the text.

DERIVATION OF THE ROCKET EQUATION

The rocket equation is a simple consequence of conservation of momentum. If a body of mass M ejects a particle of mass dM at a speed V_e, the body's speed increases in the opposite direction by an amount dV. Momentum conservation requires

$$MdV = -V_e dM \tag{7.7}$$

Repeating this process increases the body's speed and decreases its mass. Solving for dV and integrating gives

$$\int_{V_i}^{V_f} dV = V_f - V_i \equiv \Delta V = -V_e \int_{M_i}^{M_f} \frac{dM}{M} = -V_e \left(\ln M_f - \ln M_i\right) \tag{7.8}$$

or

$$\Delta V = V_e \ln\left(\frac{M_i}{M_f}\right) \tag{7.9}$$

where V_i and V_f are the initial and final velocities, and M_i and M_f are the initial and final masses.

THE TIME REQUIRED FOR A MANEUVER

Since the rocket equation includes the exhaust velocity of the engine used to create the required ΔV, the propellant mass required to deliver a given ΔV can be reduced by technologies that increase V_e relative to conventional thrusters. However, if the engine produces a small thrust, the time to conduct such a maneuver can be long.

The time required to complete a given maneuver can be found by dividing the mass of propellant needed to provide the ΔV by the mass flow rate of the thruster. Using Equations 7.4 and 7.6 this can be written as

$$\Delta t = \frac{M_p}{dM/dt} = \frac{M_f\,(e^{\Delta V/V_e} - 1)}{T/V_e} \tag{7.10}$$