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Rocket Science

Technology Trends in Propulsion

by Robert A. Nelson

A satellite is launched into space on a rocket, and once there it is inserted into the operational orbit and is maintained in that orbit by means of thrusters onboard the satellite itself. This article will summarize the fundamental principles of rocket propulsion and describe the main features of the propulsion systems used on both launch vehicles and satellites.

The law of physics on which rocket propulsion is based is called the principle of momentum. According to this principle, the time rate of change of the total momentum of a system of particles is equal to the net external force. The momentum is defined as the product of mass and velocity. If the net external force is zero, then the principle of momentum becomes the principle of conservation of momentum and the total momentum of the system is constant. To balance the momentum conveyed by the exhaust, the rocket must generate a momentum of equal magnitude but in the opposite direction and thus it accelerates forward.

The system of particles may be defined as the sum of all the particles initially within the rocket at a particular instant. As propellant is consumed, the exhaust products are expelled at a high velocity. The center of mass of the total system, subsequently consisting of the particles remaining in the rocket and the particles in the exhaust, follows a trajectory determined by the external forces, such as gravity, that is the same as if the original particles remained together as a single entity. In deep space, where gravity may be neglected, the center of mass remains at rest.

ROCKET THRUST

The configuration of a chemical rocket engine consists of the combustion chamber, where the chemical reaction takes place, and the nozzle, where the gases expand to create the exhaust. An important characteristic of the rocket nozzle is the existence of a throat. The velocity of the gases at the throat is equal to the local velocity of sound and beyond the throat the gas velocity is supersonic. Thus the combustion of the gases within the rocket is independent of the surrounding environment and a change in external atmospheric pressure cannot propagate upstream.

The thrust of the rocket is given by the theoretical equation

$$F = \lambda \dot{m} v_e + (p_e - p_a) A_e$$

This equation consists of two terms. The first term, called the momentum thrust, is equal to the product of the propellant mass flow rate $\dot{m}$ and the exhaust velocity v_e with a correction factor λ for nonaxial flow due to nozzle divergence angle. The second term is called the pressure thrust. It is equal to the difference in pressures p_e and p_a of the exhaust velocity and the ambient atmosphere, respectively, acting over the area A_e of the exit plane of the rocket nozzle. The combined effect of both terms is incorporated into the effective exhaust velocity c. Thus the thrust is also written

$$F = \dot{m} c$$

where an average value of c is used, since it is not strictly constant.

The exhaust exit pressure is determined by the expansion ratio given by

$$\varepsilon = A_e / A_t$$

which is the ratio of the area of the nozzle exit plane A_e and the area of the throat A_t. As the expansion ratio ε increases, the exhaust exit pressure p_e decreases.

The thrust is maximum when the exit pressure of the exhaust is equal to the ambient pressure of the surrounding environment, that is, when $p_e = p_a$. This condition is known as optimum expansion and is achieved by proper selection of the expansion ratio. Although optimum expansion makes the contribution of the pressure thrust zero, it results in a higher value of exhaust velocity v_e such that the increase in momentum thrust exceeds the reduction in pressure thrust.

A conical nozzle is easy to manufacture and simple to analyze. If the apex angle is 2α, the correction factor for nonaxial flow is

$$\lambda = \tfrac{1}{2}(1 + \cos\alpha)$$

The apex angle must be small to keep the loss within acceptable limits. A typical design would be $\alpha = 15°$, for which $\lambda = 0.9830$. This represents a loss of 1.7 percent. However, conical nozzles are excessively long for large expansion ratios and suffer additional losses caused by flow separation. A bell-shaped nozzle is therefore superior because it promotes expansion while reducing length.

ROCKET PROPULSION PARAMETERS

The specific impulse I_{sp} of a rocket is the parameter that determines the overall effectiveness of the rocket nozzle and propellant. It is defined as the ratio of the thrust and the propellant weight flow rate, or

$$I_{sp} = F / \dot{m} g = c / g$$

where g is a conventional value for the acceleration of gravity (9.80665 m/s^2 exactly). Specific impulse is expressed in seconds.

Although gravity has nothing whatever to do with the rocket propulsion chemistry, it has entered into the definition of specific impulse because in past engineering practice mass was expressed in terms of the corresponding weight on the surface of the earth. By inspection of the equation, it can be seen that the specific impulse I_{sp} is physically equivalent to the effective exhaust velocity c, but is rescaled numerically and has a different unit because of division by g. Some manufacturers now express specific impulse in newton seconds per kilogram, which is the same as effective exhaust velocity in meters per second.

Two other important parameters are the thrust coefficient C_F and the characteristic exhaust velocity c^*. The thrust coefficient is defined as

$$C_F = F / A_t p_c = \dot{m} c / A_t p_c$$

where F is the thrust, A_t is the throat area, and p_c is the chamber pressure. This parameter is the figure of merit of the nozzle design. The characteristic exhaust velocity is defined as

$$c^* = A_t \; p_c \; / \; \dot{m} \;\; = c \; / \; C_F$$

This parameter is the figure of merit of the propellant. Thus the specific impulse may be written

$$I_{sp} \;= C_F \;\; c^* / \; g$$

which shows that the specific impulse is the figure of merit of the nozzle design and propellant as a whole, since it depends on both C_F and c^*. However, in practice the specific impulse is usually regarded as a measure of the efficiency of the propellant alone.

LAUNCH VEHICLE PROPULSION SYSTEMS

In the first stage of a launch vehicle, the exit pressure of the exhaust is equal to the sea level atmospheric pressure 101.325 kPa (14.7 psia) for optimum expansion. As the altitude of the rocket increases along its trajectory, the surrounding atmospheric pressure decreases and the thrust increases because of the increase in pressure thrust. However, at the higher altitude the thrust is less than it would be for optimum expansion at that altitude. The exhaust pressure is then greater than the external pressure and the nozzle is said to be underexpanded. The gas expansion continues downstream and manifests itself by creating diamond-shaped shock waves that can often be observed in the exhaust plume.

The second stage of the launch vehicle is designed for optimum expansion at the altitude where it becomes operational. Because the atmospheric pressure is less than at sea level, the exit pressure of the exhaust must be less and thus the expansion ratio must be greater. Consequently, the second stage nozzle exit diameter is larger than the first stage nozzle exit diameter.

For example, the first stage of a Delta II 7925 launch vehicle has an expansion ratio of 12. The propellant is liquid oxygen and RP-1 (a kerosene-like hydrocarbon) in a mixture ratio (O/F) of 2.25 at a chamber pressure of 4800 kPa (700 psia) with a sea level specific impulse of 255 seconds. The second stage has a nozzle expansion ratio of 65 and burns nitrogen tetroxide and Aerozene 50 (a mixture of hydrazine and unsymmetrical dimethyl hydrazine) in a mixture ratio of 1.90 at a chamber pressure of 5700 kPa (830 psia), which yields a vacuum specific impulse of 320 seconds.

In space, the surrounding atmospheric pressure is zero. In principle, the expansion ratio would have to be infinite to reduce the exit pressure to zero. Thus optimum expansion is impossible, but it can be approximated by a very large nozzle diameter, such as can be seen on the main engines of the space shuttle with ε = 77.5. There is ultimately a tradeoff between increasing the size of the nozzle exit for improved performance and reducing the mass of the rocket engine.

In a chemical rocket, the exhaust velocity, and hence the specific impulse, increases as the combustion temperature increases and the molar mass of the exhaust products decreases. Thus liquid oxygen and liquid hydrogen are nearly ideal chemical rocket propellants because they burn energetically at high temperature (about 3200 K) and produce nontoxic exhaust products consisting of gaseous hydrogen and water vapor with a small effective molar mass (about 11 kg/kmol). The vacuum specific impulse is about 450 seconds. These propellants are used on the space shuttle, the Atlas Centaur upper stage, the Ariane-4 third stage, the Ariane-5 core stage, the H-2 first and second stages, and the Long March CZ-3 third stage.

SPACECRAFT PROPULSION SYSTEMS

The spacecraft has its own propulsion system that is used for orbit insertion, stationkeeping, momentum wheel desaturation, and attitude control. The propellant required to perform a maneuver with a specified velocity increment Δv is given by the "rocket equation"

$$\Delta m \;=\; m_0 \,[\, 1 - \exp(- \,\Delta v \,/\, I_{sp}\, g) \,]$$

where m_0 is the initial spacecraft mass. This equation implies that a reduction in velocity increment or an increase in specific impulse translates into a reduction in propellant.

In the case of a geostationary satellite, the spacecraft must perform a critical maneuver at the apogee of the transfer orbit at the synchronous altitude of 35,786 km to simultaneously remove the inclination and circularize the orbit. The transfer orbit has a perigee altitude of about 200 km and an inclination roughly equal to the latitude of the launch site. To minimize the required velocity increment, it is thus advantageous to have the launch site as close to the equator as possible.

For example, in a Delta or Atlas launch from Cape Canaveral the transfer orbit is inclined at 28.5° and the velocity increment at apogee is 1831 m/s; for an Ariane launch from Kourou the inclination is 7° and the velocity increment is 1502 m/s; while for a Zenit flight from the Sea Launch platform on the equator the velocity increment is 1478 m/s. By the rocket equation, assuming a specific impulse of 300 seconds, the fraction of the separated mass consumed by the propellant for the apogee maneuver is 46 percent from Cape Canaveral, 40 percent from Kourou, and 39 percent from the equator. As a rule of thumb, the mass of a geostationary satellite at beginning of life is on the order of one half its mass when separated from the launch vehicle.

Before performing the apogee maneuver, the spacecraft must be reoriented in the transfer orbit to face in the proper direction for the thrust. This task is sometimes performed by the launch vehicle at spacecraft separation or else must be carried out in a separate maneuver by the spacecraft itself. In a launch from Cape Canaveral, the angle through which the satellite must be reoriented is about 132°.

Once on station, the spacecraft must frequently perform a variety of stationkeeping maneuvers over its mission life to compensate for orbital perturbations. The principal perturbation is the combined gravitational attractions of the sun and moon, which causes the orbital inclination to increase by nearly one degree per year. This perturbation is compensated by a north-south stationkeeping maneuver approximately once every two weeks so as to keep the satellite within 0.05° of the equatorial plane. The average annual velocity increment is about 50 m/s, which represents 95 percent of the total stationkeeping fuel budget. Also, the slightly elliptical shape of the earth's equator causes a longitudinal drift, which is compensated by east-west stationkeeping maneuvers about once a week, with an annual velocity increment of less than 2 m/s, to keep the satellite within 0.05° of its assigned longitude.

In addition, solar radiation pressure caused by the transfer of momentum carried by light and infrared radiation from the sun in the form of electromagnetic waves both flattens the orbit and disturbs the orientation of the satellite. The orbit is compensated by an eccentricity control maneuver that can sometimes be combined with east-west stationkeeping. The orientation of the satellite is maintained by momentum wheels supplemented by magnetic torquers and thrusters. However, the wheels must occasionally be restored to their nominal rates of rotation by means of a momentum wheel desaturation maneuver in which a thruster is fired to offset the change in angular momentum.

Geostationary spacecraft typical of those built during the 1980s have solid propellant rocket motors for the apogee maneuver and liquid hydrazine thrusters for stationkeeping and attitude control. The apogee kick motor uses a mixture of HTPB fuel and ammonium perchlorate oxidizer with a specific impulse of about 285 seconds. The hydrazine stationkeeping thrusters operate by catalytic decomposition and have an initial specific impulse of about 220 seconds. They are fed by the pressure of an inert gas, such as helium, in the propellant tanks. As propellant is consumed, the gas expands and the pressure decreases, causing the flow rate and the specific impulse to decrease over the mission life. The performance of the hydrazine is enhanced in an electrothermal hydrazine thruster (EHT), which produces a hot gas mixture at about 1000 °C with a lower molar mass and higher enthalpy and results in a higher specific impulse of between 290 and 300 seconds.

For example, the Ford Aerospace (now Space Systems/Loral) INTELSAT V satellite has a Thiokol AKM that produces an average thrust of 56 kN (12,500 lbf) and burns to depletion in approximately 45 seconds. On-orbit operations are carried out by an array of four 0.44 N (0.1 lbf) thrusters for roll control, ten 2.0 N (0.45 lbf) thrusters for pitch and yaw control and E/W stationkeeping, and two 22.2 N (5.0 lbf) thrusters for repositioning and reorientation. Four 0.3 N (0.07 lbf) EHTs are used for N/S stationkeeping. The nominal mass of the spacecraft at beginning of life (BOL) is 1005 kg and the dry mass at end of life (EOL) is 830 kg. The difference of 175 kg represents the mass of the propellant for a design life of 7 years.

Satellites launched in the late 1980s and 1990s typically have an integrated propulsion system that use a bipropellant combination of monomethyl hydrazine as fuel and nitrogen tetroxide as oxidizer. The specific impulse is about 300 seconds and fuel margin not used for the apogee maneuver can be applied to stationkeeping. Also, since the apogee engine is restartable, it can be used for perigee velocity augmentation and supersynchronous transfer orbit scenarios that optimize the combined propulsion capabilities of the launch vehicle and the spacecraft.

For example, the INTELSAT VII satellite, built by Space Systems/Loral, has a Marquardt 490 N apogee thruster and an array of twelve 22 N stationkeeping thrusters manufactured by Atlantic Research Corporation with a 150:1 columbium nozzle expansion ratio and a specific impulse of 235 seconds. For an Ariane launch the separated mass in GTO is 3610 kg, the mass at BOL is 2100 kg, and the mass at EOL is 1450 kg. The mission life is approximately 17 years.

The Hughes HS-601 satellite has a similar thruster configuration. The mass is approximately 2970 kg at launch, 1680 kg at BOL, and 1300 kg for a nominal 14 year mission.

An interesting problem is the estimation of fuel remaining on the spacecraft at any given time during the mission life. This information is used to predict the satellite end of life. There are no “fuel gauges” so the fuel mass must be determined indirectly. There are three principal methods. The first is called the “gas law” method, which is based on the equation of state of an ideal gas. The pressure and temperature of the inert gas in the propellant tanks is measured by transducers and the volume of the gas is computed knowing precisely the pressure and temperature at launch. The volume of the remaining propellant can thus be deduced and the mass determined from the known density as a function of temperature. Corrections must be applied for the expansion of the tanks and the propellant vapor pressure. The second method is called the “bookkeeping” method. In this method the thruster time for each maneuver is carefully measured and recorded. The propellant consumed is then calculated from mass flow rate expressed in terms of the pressure using an empirical model. The third method is much more sophisticated and is based on the measured dynamics of the spacecraft after a stationkeeping maneuver to determine its total mass. In general, these three independent methods provide redundant information that can be applied to check one another.

NEW TECHNOLOGIES

Several innovative technologies have substantially improved the fuel efficiency of satellite stationkeeping thrusters. The savings in fuel can be used to increase the available payload mass, prolong the mission life, or reduce the mass of the spacecraft.

The first of these developments is the electric rocket arcjet technology. The arcjet system uses an electric arc to superheat hydrazine fuel, which nearly doubles its efficiency. An arcjet thruster has a specific impulse of over 500 seconds. Typical thrust levels are from 0.20 to 0.25 N. The arcjet concept was developed by the NASA Lewis Research Center in Cleveland and thrusters have been manufactured commercially by Primex Technologies, a subsidiary of the Olin Corporation.

AT&T’s Telstar 401 satellite, launched in December 1993 (and subsequently lost in 1997 due to an electrical failure generally attributed to a solar flare) was the first satellite to use arcjets. The stationkeeping propellant requirement was reduced by about 40 percent, which was critical to the selection of the Atlas IIAS launch vehicle. Similar arcjet systems are used on INTELSAT VIII and the Lockheed Martin A2100 series of satellites. INTELSAT VIII, for example, has a dual mode propulsion system incorporating a bipropellant liquid apogee engine that burns hydrazine and oxidizer for orbit insertion and four arcjets that use monopropellant hydrazine in the reaction control subsystem for stationkeeping.

Electrothermal hydrazine thrusters continue to have applications on various geostationary satellites and on some small spacecraft where maneuvering time is critical. For example, EHTs are used on the IRIDIUM satellites built by Lockheed Martin.

The most exciting development has been in the field of ion propulsion. The propellant is xenon gas. Although the thrust is small and on the order of a few millinewtons, the specific impulse is from 2000 to 4000 seconds, which is about ten to twenty times the efficiency of conventional bipropellant stationkeeping thrusters. Also, the lower thrust levels have the virtue of minimizing attitude disturbances during stationkeeping maneuvers.

The xenon ion propulsion system, or XIPS (pronounced "zips"), is a gridded ion thruster developed by Hughes. This system is available on the HS-601 HP (high power) and HS-702 satellite models and allows for a reduction in propellant mass of up to 90 percent for a 12 to 15 year mission life. A typical satellite has four XIPS thrusters, including two primary thrusters and two redundant thrusters.

Xenon atoms, an inert monatomic gas with the highest molar mass (131 kg/kmol), are introduced into a thruster chamber ringed by magnets. Electrons emitted by a cathode knock off electrons from the xenon atoms and form positive xenon ions. The ions are accelerated by a pair of gridded electrodes, one with a high positive voltage and one with a negative voltage, at the far end of the thrust chamber and create more than 3000 tiny beams. The beams are neutralized by a flux of electrons emitted by a device called the neutralizer to prevent the ions from being electrically attracted back to the thruster and to prevent a space charge from building up around the satellite.

The increase in kinetic energy of the ions is equal to the work done by the electric field, so that

$$\tfrac{1}{2} m v^2 = q V$$

where q, m, and v are the charge, mass, and velocity of the ions and V is the accelerating voltage, equal to the algebraic difference between the positive voltage on the positive grid and the negative voltage on the neutralizer. The charge to mass ratio of xenon ions is 7.35×10^5 C/kg.

The HS-601 HP satellite uses 13-centimeter diameter XIPS engines to perform north-south stationkeeping and to assist the spacecraft's gimballed momentum wheel for roll and yaw control. The accelerating voltage is about 750 volts and the ions have a velocity of 33,600 m/s. The specific impulse is 3400 seconds with a mass flow rate of 0.6 mg/s and

18 mN of thrust. Each ion thruster operates for approximately 5 hours per day and uses 500 W from the available 8 kW total spacecraft power.

The HS-702 spacecraft has higher power

25-centimeter thrusters to perform all stationkeeping maneuvers and to complement the four momentum wheels arranged in a tetrahedron configuration for attitude control. The accelerating voltage is 1200 volts, which produces an ion beam with a velocity of 42,500 m/s. The specific impulse is 4300 seconds, the mass flow rate is 4 mg/s, and the thrust is 165 mN. Each HS-702 ion thruster operates for approximately 30 minutes per day and requires 4.5 kW from the 10 to 15 kW solar array. The stationkeeping strategy maintains a tolerance of $\pm$ 0.005° that allows for the collocation of several satellites at a single orbital slot.

The HS-702 satellite has a launch mass of up to 5200 kg and an available payload mass of up to 1200 kg. The spacecraft can carry up to 118 transponders, comprising 94 active amplifiers and 24 spares. A bipropellant propulsion system is used for orbit acquisition, with a fuel capacity of 1750 kg. The XIPS thrusters need only 5 kg of xenon propellant per year, a fraction of the requirement for conventional bipropellant or arcjet systems. The HS-702 also has the option of using XIPS thrusters for orbit raising in transfer orbit to further reduce the required propellant mass budget.

The first commercial satellite to use ion propulsion was PAS-5, which was delivered to the PanAmSat Corporation in August 1997. PAS-5 was the first HS-601 HP model, whose xenon ion propulsion system, together with gallium arsenside solar cells and advanced battery performance, permitted the satellite to accommodate a payload twice as powerful as earlier HS-601 models while maintaining a 15 year orbital life. Four more Hughes satellites with XIPS technology were in orbit by the end of 1998. In addition, Hughes also produced a 30-centimeter xenon ion engine for NASA's Deep Space 1 spacecraft, launched in October 1998.

Another type of ion thruster is the Hall effect ion thruster. The ions are accelerated along the axis of the thruster by crossed electric and magnetic fields. A plasma of electrons in the thrust chamber produces the electric field. A set of coils creates the magnetic field, whose magnitude is the most difficult aspect of the system to adjust. The ions attain a speed of between 15,000 and 20,000 m/s and the specific impulse is about 1800 seconds. This type of thruster has been flown on several Russian spacecraft.

SUMMARY

The demand for ever increasing satellite payloads has motivated the development of propulsion systems with greater efficiency. Typical satellites of fifteen to twenty years ago had solid apogee motors and simple monopropellant hydrazine stationkeeping thrusters. Electrically heated thrusters were designed to increase the hydrazine performance and the principle was further advanced by the innovation of the arcjet thruster. Bipropellant systems are now commonly used for increased performance and versatility.

The future will see a steady transition to ion propulsion. The improvements in fuel efficiency permit the savings in mass to be used for increasing the revenue-generating payloads (with attendant increase in solar arrays, batteries, and thermal control systems to power them), extending the lifetimes in orbit, or reducing the spacecraft mass to permit a more economical launch vehicle.

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