# Ion engines 6

In this chapter we look at a type of non-thermal rocket, the ion engine. This uses electric or magnetic fields to ionize and accelerate a propellant. Although there are many types, we focus here on the so-called gridded ion engine to illustrate the general principles, and to identify how the effective exhaust velocity, thrust, and power depend on characteristics of the engine. We will examine a real interplanetary mission, Dawn, that uses this engine. We finish by looking briefly at some of the other types of ion engine.

## 6.1 Principle

In a chemical thermal rocket, the fuel energy and the propellant mass come from the same source. For example, hydrogen is burned in oxygen to release energy that then propels the product of this reaction, water. This is convenient, but it is not the only way of producing rocket propulsion.

In an ion engine the fuel and propellant are separate. The principle is as follows. A gas is ionized into its constituent ions and electrons. An electric field is then used to accelerate the much more massive ions into a collimated beam out of the back of the rocket (figure 6.1). By conservation of momentum, this accelerates the rocket in the opposite direction, just like in a chemical thermal rocket. The electrical energy required to ionize and accelerate the propellant could come from any source, such as solar panels, a nuclear reactor, or even chemical reactions.

If the potential difference between the anode and the cathode of the accelerating field is $U$, then the kinetic energy of each ion of mass $m$ and charge $q$, which is $(1/2)mv^2$, is equal to the work done by the electric field, $qU$. Thus the velocity of the ions is

$$v = \sqrt{\frac{2qU}{m}}\,. \tag{6.1}$$

When we consider the ions at the cathode leaving the spacecraft, $v$ is the (effective) exhaust velocity of the ion engine. Intuitively, a larger potential, a larger charge, or a smaller ion mass yield a larger velocity.

Several ion engines have been built and used for interplanetary missions. The NSTAR engine (discussed below) uses a potential of around $U = 1250\,\mathrm{V}$. Although equation 6.1 implies we want to use ions with small masses, effective exhaust velocity is not the only parameter of interest. We also want sufficient thrust and, for efficiency reasons, a high thrust-to-power ratio. The latter is given by $2/v_e$ (equation 3.23), which suggests we want massive ions. In practice there is a trade-off.

Noble gases are preferred as the source of ions because they have low first ionization potentials, and because they are relatively massive yet gaseous. With full outer electron shells

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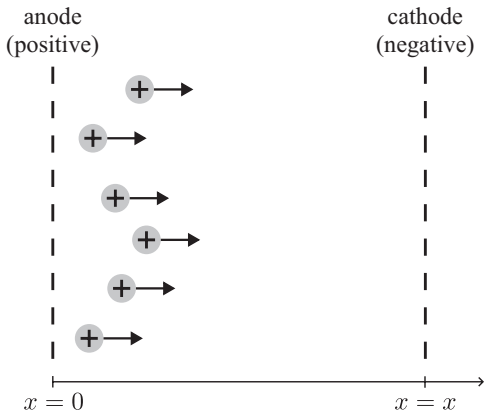



**Fig. 6.1** The basic principle of an ion engine.

in their neutral state, noble gases cannot build molecules, so energy is not wasted during the ionization process exciting internal rotational or vibrational states.

A common choice of gas is xenon (atomic number 54), which has seven stable isotopes with a mean nucleon number in nature of 131.3. When this is singly ionized ($q = e$), equation 6.1 gives $v = 38.3\,\mathrm{km\,s^{-1}}$. This compares to about $4.5\,\mathrm{km\,s^{-1}}$ for liquid hydrogen–oxygen chemical engines. In practice there will be some losses, so the effective exhaust velocity of the engine will be lower than the simple calculation above, but still considerably higher than a chemical thermal rocket.

Let us now compute the mass flow rate so we can calculate the engine's thrust. If there were no ions in the space between the anode and cathode in figure 6.1, then the electric potential would vary linearly from $U$ at the anode to zero at the cathode. But the positive ions being accelerated partially shield the negative cathode from the anode, so the anode experiences a lower potential difference in the downstream direction. Thus the more ions there are between the two grids (anode and cathode), the smaller the potential gradient $dU/dx$. Once the ion density is large enough, this gradient drops to zero. As the magnitude of the force on the ions is $F = qE = -qdU/dx$, $E = |\mathbf{E}|$ being the magnitude of the electric field, this means no more ions can be accelerated. Hence there is a maximum charge density that can be attained between the grids, and this places an upper limit on the mass flow rate.

To solve for this maximum flow rate we use Poisson's equation. This relates the gradient of the electric field to the charge density $nq$, where $n$ is the ion number density,

$$-\nabla \cdot \mathbf{E} = \frac{d^2U}{dx^2} = -\frac{nq}{\epsilon_0}\,. \tag{6.2}$$

If $v$ is the velocity of the ion at some position between the grids, then the current density (amperes per unit cross-sectional area of the flow) is $j = nqv$. Using equation 6.1 we can rewrite the above equation as

$$\frac{d^2U}{dx^2} = -\frac{j}{\epsilon_0}\sqrt{\frac{m}{2q}}\,U^{-1/2} = -CU^{-1/2}, \tag{6.3}$$

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which defines $C$. Multiplying both sides by $dU/dx$ and integrating, we can rewrite this as[1]

$$\int_0^{(dU/dx)^2} d\left(\frac{dU}{dx}\right)^2 = -4C\int_{U^{1/2}}^0 d(U^{1/2}). \tag{6.4}$$

The lower integration limit refers to the maximum charge density at the anode, where the potential is $U$ and the potential gradient is zero. The upper integration limit is at some arbitrary downstream point $x$ where we take the potential to be zero, for example at the cathode. The result of the integration is

$$\left(\frac{dU}{dx}\right)^2 = 4CU^{1/2}. \tag{6.5}$$

Taking the square root of this – and retaining the negative solution because the potential gradient is negative – we get

$$\frac{dU}{dx} = -2C^{1/2}U^{1/4}, \tag{6.6}$$

which we integrate again between $U = U$ at $x = 0$ and $U = 0$ at $x = x$ to get

$$\frac{4}{3}U^{3/4} = 2C^{1/2}x. \tag{6.7}$$

Substituting for $C$ (defined in equation 6.3) we find that the maximum achievable current density is

$$j_{\max} = \frac{4\epsilon_0}{9}\left(\frac{2q}{m}\right)^{1/2}\frac{U^{3/2}}{x^2}. \tag{6.8}$$

$U$ is the potential difference between anode and cathode of separation $x$. The above equation has $U \propto x^{4/3}$. In a vacuum the dependence is $U \propto x$, because the electric field is constant. In the ion engine the presence of the ions increases the gradient of the potential.

The mass flow rate in the ion engine is $\dot{m} = jAm/q$, where $A$ is the cross-sectional area of the grids that make up the engine. The thrust is

$$F = \dot{m}v_e = jA\sqrt{\frac{2mU}{q}} \tag{6.9}$$

using equation 6.1. From equation 6.8, the maximum mass flow rate is

$$\dot{m}_{\max} = \frac{4\epsilon_0}{9}A\left(\frac{2m}{q}\right)^{1/2}\frac{U^{3/2}}{x^2} \tag{6.10}$$

and so the maximum thrust, achieved at the charge density limit, is

$$F_{\max} = \frac{8}{9}\epsilon_0 A\left(\frac{U}{x}\right)^2. \tag{6.11}$$

Although the effective exhaust velocity depends on the mass and charge of the ions, the maximum thrust is independent of both. This thrust depends only on the gradient of the

[1] I have used

$$\frac{1}{2}\frac{d}{dx}\left(\frac{dU}{dx}\right)^2 = \frac{dU}{dx}\frac{d^2U}{dx^2} \qquad \text{and} \qquad 2\frac{d}{dx}(U^{1/2}) = U^{-1/2}\frac{dU}{dx}.$$

Also, I somewhat sloppily do not distinguish between the integration variable and the limits in equation 6.4.

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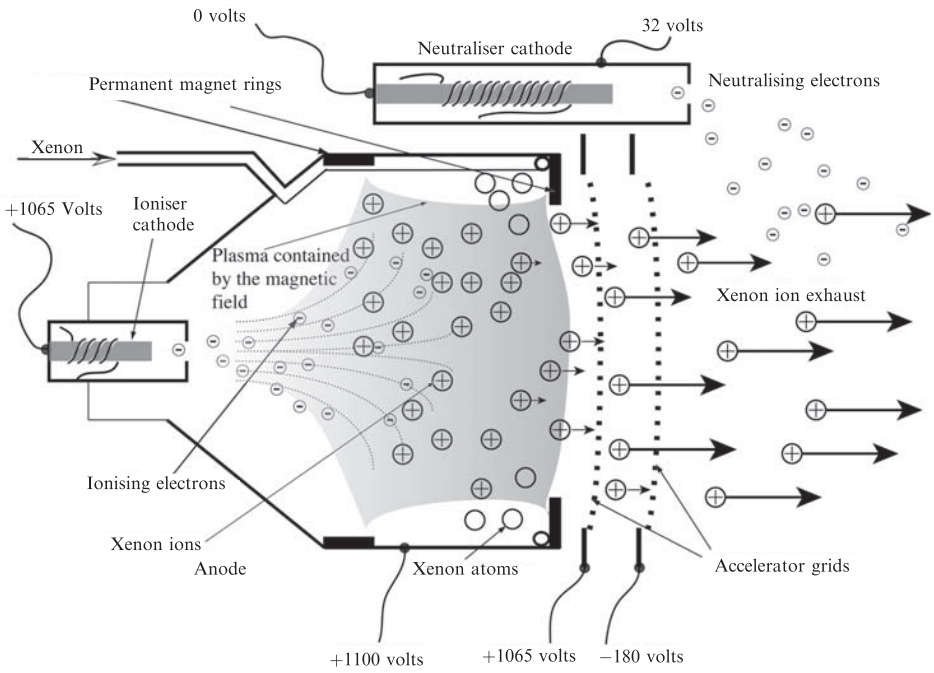



**Fig. 6.2** A schematic of the NSTAR gridded ion engine on Deep Space 1. Credit: Used with permission of Springer Nature BV, from Turner (2009); permission conveyed through Copyright Clearance Center, Inc.

electric potential and the size of the engine. This might seem counter-intuitive, and will be discussed further in section 6.2.

An example of a real ion thruster is shown in figure 6.2. This is the NSTAR ion engine, which was used on the NASA Deep Space 1 and Dawn missions (see section 6.3). The process in the engine is as follows. In the left part of the engine, a hot cathode releases electrons by the process of thermionic emission, the same process that is used in low pressure mercury lamps. These electrons are accelerated to a few tens of electronvolts by an electric field into a region where the xenon atoms are pumped in. These xenon atoms are ionized by collisions with the electrons. To increase the probability that any one atom is ionized, an axial magnetic field is used to make the electrons move on long spiral paths. The ions drift to the right end of the ionization chamber. Here they are accelerated by a large positive potential ($+1065 - -180\,\mathrm{V} = 1245\,\mathrm{V}$) held between the two grids through which ions pass (these are the anode and cathode of the earlier description). The ions escape through the cathode with a large velocity relative to the spacecraft, thus providing the thrust.

This positive charge current leaving the spacecraft would cause the spacecraft to become negatively charged over time, which among other things would result in an attractive force between the spacecraft and the departing ions, which would in turn decelerate the spacecraft. To avoid this, electrons are injected into the ion beam after it has passed through the cathode in order to neutralize the spacecraft.

Unlike a chemical rocket, there is no nozzle, because we do not need to extract internal thermal energy from the ions. In fact we are not even able to, because the ion flow is already ordered. The ion engine is, in some sense, an electromagnetic nozzle.

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With an accelerating potential of $U = 1000\,\mathrm{V}$, a grid separation $x = 1\,\mathrm{mm}$, and an area $A = \pi 0.15^2 = 0.07\,\mathrm{m}^2$, the above expressions give $\dot{m}_{\max} = 1.5 \times 10^{-5}\,\mathrm{kg\,s^{-1}}$ and $F_{\max} = 0.56\,\mathrm{N}$. Mass flow rates and thus thrusts are typically low from ion engines, despite their high effective exhaust velocities. The density of the ion gas is also very low (see exercises), resulting in a very low exit pressure. The engines therefore need to be operated in a very low pressure environment, otherwise the surrounding gas would flow in and prevent it from operating. Ion engines are therefore not suitable for launching a spacecraft from the surface of the Earth. But on account of their large effective exhaust velocities, they can be used over long periods to acquire a significant $\Delta v$ on a low-mass spacecraft.

## 6.2 Power, potential, and propellant

It may seem odd that the maximum thrust of the gridded ion engine is independent of the ion mass (equation 6.10). After all, if we double the mass of the ion, $v_e$ will decrease by a factor of $\sqrt{2}$ (equation 6.1), but wouldn't $\dot{m}$ double and so the thrust ($\dot{m}v_e$) increase by a factor of $\sqrt{2}$ also? With such apparent contradictions we need to careful about what quantities are being kept constant, and what is forced to change when we vary some parameter. In this case, decreasing $v_e$ by a factor of $\sqrt{2}$ lowers the ion flow rate (number per second) by the same factor, and so even if we double the individual ion masses, $\dot{m}$ – mass times rate – increases by a factor of $\sqrt{2}$. This agrees with equation 6.10 (the other terms are fixed). Thus the thrust is unchanged. Note in particular that we kept $U$ constant.

We can get a bit more insight if we consider the power. The energy of an ion is $qU$, and the ion flow rate is $\dot{m}/m$, so the power of the engine output is their product:

$$P = \frac{q}{m} U \dot{m} \,. \tag{6.12}$$

Using the expression for the maximum mass flow rate (equation 6.10) we can write this power at the maximum thrust in terms of the quantities we directly control in the engine:

$$P_{\max} = \frac{4\epsilon_0}{9} A \left(\frac{2q}{m}\right)^{1/2} \frac{U^{5/2}}{x^2} \,. \tag{6.13}$$

Thus if we double the mass of the ion at constant potential – keeping also $A$, $q$, and $x$ constant – the power decreases by a factor of $\sqrt{2}$. If we instead want to keep the power constant while doubling the ion mass, then we have to increase the potential by a factor of $2^{1/5}$. It follows that $v_e$ then decreases by a factor of $2^{2/5}$ (from equation 6.1), $\dot{m}$ increases by a factor of $2^{4/5}$ (from equation 6.12), and so the thrust increases by a factor of $2^{2/5}$.

Causally, the power is not something we fix, but is rather what we get out having fixed the potential and the mass flow rate. This is just like with an electric current (not least because this *is* an electric current): we fix the potential $U$ and current $jA$, and these determine the power, $P = UjA$. The power we need to provide to the engine is of course closely related to $P$, but is higher, due to losses.

Another important parameter for ion engines is the thrust-to-power ratio, $F/P$. For any type of rocket this is $2/v_e$ (equation 3.23), and for an ion engine this equals $\sqrt{2m/qU}$ (equation 6.1). A more powerful engine will be more massive, and the thrust must also

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accelerate that mass. Chemical rockets tend to have initial propellant-to-spacecraft mass ratios of a few tens, so most of the thrust is used to propel the remaining propellant at first. But with ion engines, the mass flow rates are so small that there is relatively little propellant; the initial propellant-to-spacecraft mass ratio is often much less than one. The total mass is then dominated by the mass of the spacecraft including its engine. This makes it more important to achieve a high thrust-to-power ratio, even though this means a low $v_e$. In general there will be an optimal value of $v_e$ that minimizes the engine mass.

During a mission, $m$, $q$, $A$, and $x$ cannot be changed. The only control parameter we have is $U$, and varying this will vary both the effective exhaust velocity and the maximum thrust, because $v_e \propto U^{1/2}$ and $F_{\max} \propto U^2$. There is a limit to how large $U$ can be, because too large a potential gradient ($U/x$) can lead to the acceleration grid being distorted, as the propulsion force acts on the grid. We can of course operate the engines below maximum thrust by lowering the current. We could also install multiple ion engines in parallel, then adjust the thrust by varying the number of engines in operation, which is effectively equivalent to varying $A$.

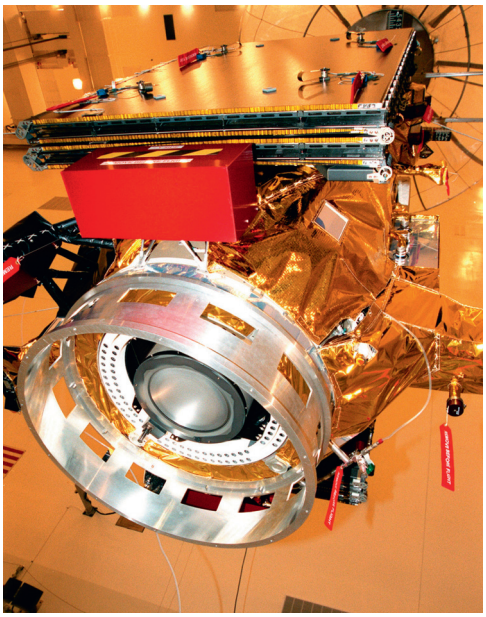
With chemical rockets we sometimes choose a propellant with a low molar mass to raise $v_e$ (following equation 5.10), in order to achieve a larger $\Delta v$ for a given mass ratio, in accordance with the rocket equation. As mass flow rates tend to be large in chemical rockets, the propellant can be exhausted quickly and the required $\Delta v$ achieved quickly (of the order of minutes). In ion engines the mass flow rate is typically so low that the $\Delta v$ takes a long time to achieve (of the order of months). If we lower the ion mass at constant power to increase $v_e$ (following equation 6.1), this lowers the mass flow rate even further, and so it takes even longer to achieve the desired velocity. Depending on the distance to the destination, this may increase the total journey time. Thus depending on the mission profile, maximizing $v_e$ may not be optimal. Indeed, ion engines are not necessarily employed to get somewhere quickly, but rather to provide a continuous, long-term thrust that can provide more flexible trajectories (more on this below).

From the above discussion we see that the choice of propellant, in particular the ion mass, will affect the effective exhaust velocity and thrust, depending on whether we allow the grid potential and/or the power to vary. As we have just seen, we do not necessarily want the highest $v_e$ possible, and so in practice heavier ions such as xenon are often used (mercury and caesium were used in early engines). There are other criteria in the choice of propellant. For example, xenon has a relatively low ionization potential, and so is easier to ionize than the lighter noble gas helium. Xenon is also non-toxic, making it easier to handle (unlike mercury), can be liquefied and stored for long periods (harder with helium), and is less reactive than caesium or mercury, which tended to corrode the grid more rapidly.

## 6.3 An example application: Dawn

The NSTAR engine shown in figure 6.3 has been used on two NASA interplanetary spacecraft, Deep Space 1 launched in 1998 and Dawn launched in 2007, both of which visited minor bodies in the solar system.

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An NSTAR ion engine on the interplanetary mission Deep Space 1. Credit: NASA, cropped by the author. Fig. 6.3

When used on the Dawn spacecraft, NSTAR delivered an effective exhaust velocity of $30\,\mathrm{km\,s^{-1}}$ and a thrust of up to 90 mN, implying $\dot{m} = 3 \times 10^{-6}\,\mathrm{kg\,s^{-1}}$. Dawn had a dry mass of 747 kg plus 425 kg of xenon propellant that enabled it to achieve a total[2] $\Delta v$ of $11.5\,\mathrm{km\,s^{-1}}$. This was used to put it into orbit, first around the asteroid Vesta (in 2011) and then around the asteroid Ceres (in 2015), making it the first spacecraft to orbit two different extraterrestrial bodies. The power for the ion engines and the spacecraft as a whole was provided by $36.4\,\mathrm{m^2}$ of photovoltaics (figure 6.4) providing in excess of 10 kW of power at 1 au, but nine times less at 2.99 au (the aphelion of Ceres). The power of the ion beam when operating at the above thrust and effective exhaust velocity is $P = (1/2)\dot{m}v_e^2 = 1350\,\mathrm{W}$. The input power must be larger than this for reasons we will discuss below.

Using the above data, equation 6.1 suggests that the accelerating potential was 610 V. In practice it must have been larger than this, due to imperfect energy transfer to the ions.

With such a low thrust and an initial mass of 1218 kg, the acceleration from the ion engine is extremely small, initially just $0.09/1218 = 7.4 \times 10^{-5}\,\mathrm{m\,s^{-2}}$. Using equation 3.14 and the data above, we can estimate that in 10 days of continuous thrust, the spacecraft changes its velocity by just $64\,\mathrm{m\,s^{-1}}$ during which it expels 2.6 kg of propellant. (As the mass has hardly changed, we can also estimate the change in velocity as $v = at$ in this case.) The engine clearly has to be used for prolonged periods to achieve a large change in velocity. If run continuously at the above rate, the engine could have run for 4.5 years. It actually operated for a total of 5.9 years, indicating that it was sometimes used at lower values of $\dot{m}$. These long durations mean ion engines cannot be approximated as giving impulsive thrusts, as we assumed when computing orbit changes with the short-burst, high-thrust chemical engines in chapter 4. Thus, assuming that an ion engine is not always thrusting radially away from the

[2] The rocket equation implies $\Delta v = 30 \times \ln[(747+425)/747] = 13.5\,\mathrm{km\,s^{-1}}$. The discrepancy comes in part from the fact that Dawn also had hydrazine thrusters and fuel.

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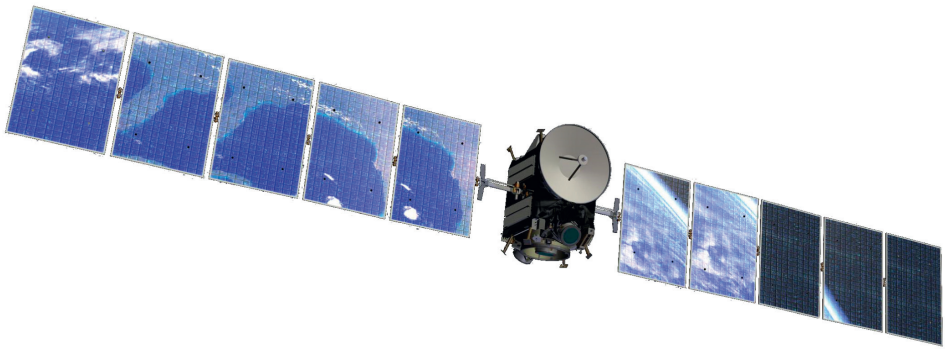


Fig. 6.4 An artist's impression of the Dawn spacecraft. Credit: NASA/JPL.

Sun, it provides a continuous non-radial force, so the spacecraft will not move on a Keplerian orbit. This opens up interesting opportunities for unusual trajectories, something we shall return to when we discuss another means of achieving this with solar sails in chapter 10.

Dawn had three NSTAR thrusters, but only used one at a time. An advantage of ion engines is that not only can the mass flow rate be varied, but also the effective exhaust velocity, by varying the potential between the anode and cathode. In the case of Dawn, the effective exhaust velocity could be varied between 7 and 30 km s$^{-1}$ and the thrust between 19 and 90 mN.

The power we calculated above is the output power of the ion beam. The power that must be provided to achieve this is higher because we first need to ionize the atoms. If the energy required to ionize each atom is $E_i$, then the power required for ionization at the output mass flow rate is $P_i = E_i\dot{m}/m$. The ionization energy of xenon is 12.1 eV, but significantly more energy is required on average. One reason is that electron collisions may not ionize the atom, but instead excite the electron, which then radiates the energy as photons as it decays back to its ground state. In practice of the order of $E_i = 500$ eV is required per ion. Using the above data for Dawn's nominal mass flow rate, and with $m = 131\, m_\mathrm{u}$, this implies $P_i = 1100$ W. This is comparable to the power of the ion beam itself calculated earlier. It would be inefficient if we spent most of the available power to ionize the atoms but then comparatively little to accelerate them. The ratio of the ion beam power (equation 6.12) to $P_i$ is $qU/E_i$. So for high energy efficiency, we want the acceleration potential to be much higher than the potential needed for the ionization.

Additional energy losses include electrons that fail to ionize any atoms and simply pass straight to the anode in the ionization chamber, as well the energy that must be expended to accelerate the neutralizing electrons at the back of the engine.

If we define an energy efficiency as the ratio of the rocket's kinetic energy to the energy in the ion beam, then we arrive at the same expression for efficiency as that for a generic rocket given in equation 3.21 (see exercises). This is arguably not a very useful expression in the present application, however, because it ignores the energy needed to ionize the atoms. Note, though, that thermal engines are limited in their efficiency to convert heat into work by the second law of thermodynamics. Ion engines, in contrast, are not heat engines, and so by making $qU$ much greater than $E_i$ we can in principle achieve high efficiencies. Whether that is important depends on the main purpose of the ion engine.

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## 6.4 Other types of ion engine

There are many variants on gridded ion engine. They vary in how they use electric and/or magnetic fields to ionize and accelerate the plasma, and have a range of performances in terms of effective exhaust velocity and thrust. Here we look very briefly at a few alternatives; many more variants exist.

Field emission electric propulsion is a variant that uses an electric field to extract ions from the surface of a metal. The metal is usually in liquid form; indium or gallium is often used. Once extracted from the metal, the ions are accelerated by an electric field to velocities of order $100\,\mathrm{km\,s^{-1}}$. The thrusts are very low, however, typically in the region of micronewtons or millinewtons, so they are mostly used for attitude control and orbit adjustment.

One of the disadvantages of the gridded ion engine is that the cathode tends to erode over time. A Hall effect thruster overcomes this by using a non-physical cathode. Electrons fired out of the back of the engine are attracted into the chamber towards the anode by an axial electric field. A radial magnetic field puts these electrons on circular paths around the axis. Atoms released near the anode are ionized by these electrons through collisions. As the ions are much more massive than the electrons, they are deflected little by the magnetic field. The axial electric field then accelerates the positive ions to produce thrust. The electrons act both as a virtual cathode and serve to ionize the atoms, with ion generation and acceleration occurring in the same region of the chamber. Some electrons fired from the back of the engine neutralize the exiting ion beam and thus keep the engine neutral overall. Effective exhaust velocities of around $15\,\mathrm{km\,s^{-1}}$ have been achieved with Hall effect thrusters, and like gridded ion engines they produce very small thrusts (typically less than a newton). The Psyche spacecraft, for example, has four Hall effect thrusters operating with xenon, each with $v_e = 18\,\mathrm{km\,s^{-1}}$ and $F = 0.28\,\mathrm{N}$, which corresponds to an ion beam power of 2.5 kW, although they can operate at up to 4.5 kW. Hall effect thrusters are more compact than gridded thrusters, and tend to be the choice of ion engine in newer spacecraft.

Some ion engines use a magnetic field to accelerate the ions via the Lorentz force, rather than an electric field to accelerate them via the Coulomb force. One particular implementation of this concept is the magnetoplasmadynamic (MPD) thruster. This has a cylindrical anode surrounding a central cathode. A large current between them ionizes the gas. The current interacts with a strong magnetic field to accelerate the plasma out of the cylinder via the Lorentz force. This may be able to achieve an effective exhaust velocity as high as $100\,\mathrm{km\,s^{-1}}$ when using hydrogen propellant.

Another proposed variant is the Variable Specific Impulse Magnetoplasma rocket (VASIMR). This uses radio waves to ionize and heat a plasma, which is then accelerated by a magnetic field. A solenoid directs the plasma, behaving as the magnetic equivalent of the convergent–divergent nozzle (see section 8.4 for more details). Effective exhaust velocities of $50\,\mathrm{km\,s^{-1}}$ and powers of 100 kW have been achieved in laboratory tests which, if achieved together, imply a relatively large thrust of 4 N. One advantage over gridded ion engines is that VASIMR has no electrode grids, so does not suffer from problems of their corrosion. Indeed, the plasma is confined electromagnetically, so could be heated to extremely high temperatures of millions of kelvins.

In none of these descriptions have we been concerned with the source of the energy (electricity) to accelerate the ions. For interplanetary spacecraft such as Dawn and Psyche, the

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source has often been solar power collected by photovoltaic cells. To achieve a higher intensity radiation than the Sun, and thus a larger output power for a given collecting area, one could consider using laser energy beamed from the Earth. This could involve similar technology and infrastructure on the Earth as that required for propelling a laser sail, which we look at in chapter 11. The main difference is that whereas for the laser sail we use the energy and momentum of the beamed photons, for the ion engine we would only use the energy.

Another source of energy is a nuclear reactor. A heat engine can be used to convert heat into mechanical motion and then into electricity, as done in a conventional power station. This could potentially provide a large current that could give larger thrusts than have been considered so far for most ion engines. This approach to accelerating ions is often referred to as nuclear electric propulsion.

## 6.5 Summary

A gridded ion engine works by ionizing a propellant and then accelerating it using an electric field. Variants on this use a magnetic field or a combination of electric and magnetic fields to ionize and accelerate the propellant. The propellant depends on the type of engine, but is often a heavy noble gas such as xenon. In contrast to chemical thermal rockets in which the fuel (and oxidizer) is the source of both the energy and the propellant, in an ion engine a separate energy source is used to generate electromagnetic fields that then accelerate an inert propellant. The energy source is often solar electric power, but could be anything in principle.

Ion engines typically provide high exhaust velocities (10–100 $\mathrm{km\,s^{-1}}$) but low thrusts (often below 1 N) on account of their low mass flow rates. The acceleration is therefore low but can be maintained for a long time. The mass ratios (spacecraft plus propellant to spacecraft) used to date are typically low, so the total $\Delta v$ achieved is often less than the effective exhaust velocity.

Ion engines are not, at present, an option for achieving fast interstellar travel. Even if a very large mass ratio (for an ion engine) of 100 could be achieved with $v_e = 100\,\mathrm{km\,s^{-1}}$, this would still only provide $\Delta v = 460\,\mathrm{km\,s^{-1}}$. But because they can be operated continuously for long periods of time (years), ion engines are suited for long-duration deep space missions where multiple manoeuvres or non-Keplerian orbits are required.

## 6.6 Exercises

1. Derive an expression for the number density of the accelerated ions at the exit of an ion engine, and compute this for an engine with $U = 1000\,\mathrm{V}$ and $x = 1\,\mathrm{mm}$ using single ionized molecules. Compare this to the number density of air molecules at the Earth's surface. What does this imply?
2. Derive an expression for the ratio of an ion engine's final kinetic energy to the total energy put into the ion beam. Is this a useful measure of efficiency?

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