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RADIOISOTOPE ELECTRIC PROPULSION FOR ROBOTIC SCIENCE MISSIONS TO NEAR-INTERSTELLAR SPACE

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Abstract

The use of radioisotope electric propulsion for sending small robotic probes on fast science missions several hundred astronomical units (AU) from the Sun is investigated. Such missions would address a large variety of solar, interstellar, galactic and cosmological science themes from unique vantage points at 100 to 600 AU, including parallax distance measurements for the entire Milky Way Galaxy, sampling of the interstellar medium and imaging of cosmological objects at the gravitational lens foci of the Sun ($\geq$ 550 AU). Radioisotope electric propulsion (REP) systems are low-thrust, ion propulsion units based on multi-hundred watt, radioisotope electric generators and ion thrusters. In a previous work, the flight times for rendezvous missions to the outer planets ($<$ 30 AU) using REP were found to be less than fifteen years. However fast prestellar missions to several hundred AU are not possible unless the probe's energy can be substantially increased in the inner Solar System so as to boost the final hyperbolic excess velocity. In this paper an economical hybrid propulsion scheme combining chemical propulsion and gravity assist in the inner Solar System and radioisotope electric propulsion in the outer Solar System is studied which enables fast prestellar missions. Total hyperbolic excess velocities of 15 AU/year and flight times to 550 AU of about 40 years are possible using REP technology that may be available in the next decade.

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Introduction

The use of radioisotope electric propulsion for sending small robotic probes on fast science missions several hundred astronomical units (AU) from the Sun is investigated. Such missions would address a large variety of solar, interstellar, galactic and cosmological science themes from unique vantage points at 100 to 600 AU. Topics of interest include heliopause physics and the origin of anomalous cosmic rays, parallax distance measurements for the entire Milky Way Galaxy which may elucidate the distribution of galactic dark matter, sampling of the interstellar medium, new opportunities for long baseline interferometry, detection of circumstellar dark objects in the Kuiper belt via infrared, gravitational or radio-occultation signatures and the imaging of distant objects at the gravitational lens foci of the Sun ($\geq$ 550 AU or about 3 light-days).

Such missions cannot be carried out with flight times of less than fifty years unless one can economically produce hyberbolic excess velocities of 10 to 20 AU/year (1 AU/year = 4.74 km/s). Chemical propulsion plus gravity assist, as used for the 800-kg Voyager planetary probes of the late 1970's, yields hyperbolic velocities of about 3 AU/year, making the flight time to 550 AU about 180 years. Using more energetic maneuvers based on chemical propulsion and gravity assist to increase velocity is problematic for such massive probes because of the large quantities of propellant needed in low Earth orbit for these missions.

Miniaturization is reducing the size and mass of spacecraft instruments so that a modest payload of 50 to 100 kilograms may soon accomodate several experiments and the associated support systems. Small payloads would encourage more frequent deep-space missions and faster response to changing scientific needs. It has long been appreciated that high specific impulse, electric propulsion allows large payload fractions to be delivered to deep-space at high velocities because of the reduced amount of propellant needed. But electric propulsion is viable only if an adequate space-power source exists.

Radioisotopes have been used successfully for nearly twenty years to supply the heat for thermoelectric generators on various deep-space probes. Radioisotopes have the advantage that their thermal power simply depends on the energy of the decay products and the isotope lifetime. The typical specific mass of a pure radioisotope with a 100 year lifetime is initially about 1 kg/kW of thermal power. Extending the use of radioisotopes to primary electric propulsion of small probes has recently been investigated.[1] Radioisotope electric propulsion (REP) systems are low-thrust, ion propulsion units based on multi-hundred watt radioisotope electric generators and ion thrusters.

The perceived liability of radioisotope electric generators for ion propulsion is their high specific mass. Present radioisotope thermoelectric generators (RTG)[2,3] have a specific mass of about 200 kg/kW although many development efforts[4,5,6,7,8] are underway with the aim of reducing the specific mass of radioisotope electric systems toward 50 kg/kW over the next decade. For electric propulsion systems, higher specific impulse (propellant velocity divided by the acceleration of gravity at the Earth's surface) requires more power and a more massive powerplant (generator plus ion thruster), which offsets the advantage of reduced propellant mass. The achievable velocity change and the optimal propellant velocity for the maximum payload fraction both decrease with increasing specific mass. This results in a longer mission time. However it is known from the elementary rocket equations in field-free space[9] that for a constant thrust, the total thrust time τ and velocity change Δv over a constant distance are only weakly dependent on the powerplant specific mass α, namely $\tau \propto \alpha^{1/3}$ and $\Delta v \propto \alpha^{-1/3}$. This suggests that high specific mass powerplants may be acceptable for certain electric propulsion missions.

Two conclusions from the study in Reference 1 were that heliocentric flight time is indeed a weak function of powerplant specific mass, and that small robotic science missions can probably be initiated with a powerplant specific mass of 100 to 200 kg/kW. As will be discussed in the next section, powerplants constructed from radioisotope electric generators and ion thrusters are likely to have specific masses in this range during the next ten years. Because of the need to avoid excessive launch masses and conserve strategic nuclear fuels, high specific mass powerplants are only practical for relatively low power levels and small payloads.

Although the flight times for rendezvous missions to the outer planets (< 30 AU) using REP alone has been calculated to be less than fifteen years,[1] fast missions beyond 100 AU are not possible unless a probe's energy can be substantially increased in the inner Solar System so as to boost the final hyperbolic excess velocity. In this paper a hybrid propulsion scheme combining chemical propulsion and gravity assist in the inner Solar System and radioisotope electric propulsion in the outer Solar System is studied which enables fast prestellar missions. The method is analogous to rocket staging and is only economical for small probes whose mass including the REP powerplant would not exceed 200 to 400 kg.

REP Technology

Several schemes for radioisotope electric generators are under development (Refs. 4-8) and may lend themselves well to an evolutionary REP program over the next decade. A gradual reduction in the specific mass of radioisotope electric generators from $\alpha_e = 200$ kg/kW to 50 kg/kW can be envisioned. Most of these devices are intended to produce total electric powers of a few hundred watts to a few kilowatts, and their modular designs allow them to be easily scaled up or down. Table 1 summarizes the performance and reliability of these emerging technologies for radioisotope electric generators. The information in the first three columns is taken from the cited references. The last column gives estimated specific masses for hypothetical powerplants using a 30 cm, derated xenon thruster to be discussed later in this section.

Electric generator	α_e (kg/kW)	Lifetime (yrs)	Limiting component	Powerplant α(kg/kW)
RTG	197	>17 dem.	Gradual unicouple aging	226
MOD-RTG	127	>8 est., 1.7 dem.	Dopant migration	156
RTPV	118	>10 est.	Neutron damage to GaSb cells	147
REC	42	10 est.	Rad. damage to thin-films	71
FPSE-DIPS	118	>10 est.	Two moving parts	147
AMTEC	67	>10 est., 1.6 dem.	Porous electrode grain growth	96

Table 1: Comparison of radioisotope electric generators and estimated powerplant specific masses using a 0.5 kW, 30 cm xenon thruster.

The first three devices in Table 1 involve direct conversion of heat to electricity: the conventional radioisotope thermoelectric generator (RTG) using SiGe unicouples,[2,3] the modular RTG (MOD-RTG) which uses SiGe/GaP multicouples consisting of 20 thermocouples connected in series[4,10,11] and the radioisotope thermophotovoltaic (RTPV) power system which would use GaSb infrared photovoltaic cells to directly convert radiant heat from the radioisotope to electricity.[5] The betavoltaic radioisotope energy converter (REC)[6] listed in Table 1 is a nonthermal, thin-film device with a radioisotope-laced semiconductor layer between two dissimilar metal electrodes. The decay products generate electron-hole pairs, and a current flow is induced by the work-function potential difference of the two metals. The last two devices in Table 1 are thermal engines which use a working fluid to convert radioisotope decay heat to electrical work. The free piston Stirling engine (FPSE) has been proposed as a lightweight dynamic isotope power system (DIPS) for multi-hundred watt applications.[7] Alkali metal thermoelectric converters (AMTEC)[8,12] electrochemically convert the isothermal expansion of sodium or potassium vapor to electrical work via a charge exchange by the liquid metal in a beta-alumina solid electrolyte and electron recombination at a porous metal electrode.

The specific mass of a complete propulsive powerplant has contributions from both the electric generator and ion thruster. Significant progress has been made in reducing the mass and extending the lifetime of low-power, xenon ion thrusters in recent years.[13] Xenon is used because it is inert, is easily and efficiently stored and provides a high thrust-to-power capability. Xenon thrusters have overall efficiencies exceeding 70% for specific impulses above 3000 sec, a range applicable for missions of even short thrust duration. Krypton and argon can also be used, but thruster efficiencies of 70% are not achieved until the specific impulses are above 5000 and 7500 sec, respectively.

To eliminate known life-limiting issues, 30 cm diameter, derated (low current density) thrusters are being developed for powers in the range 0.5 to 5 kW. Cathode degradation and positive and negative grid erosion all increase with input power, P_0. An empirical fit for the projected lifetime t_p of the 30 cm xenon thruster as a function of input power was determined using the information in Reference 13. For $P_0 < 2.5$ kW, the limiting component is the cathode with a lifetime $t_p(\text{khrs}) = 53.7\, P_0(\text{kW})^{-0.56}$, while for higher powers it is the negative grid with lifetime $t_p(\text{khrs}) = 112\, P_0(\text{kW})^{-1.34}$. Note that a 30 cm xenon thruster operating at 0.5 kW input power has a projected lifetime of about 9 years.

The mass of a 30 cm thruster excluding the power processing unit (PPU) is estimated at 7 kg.[13] An added gimbal assembly for maintaining proper thrust vectoring typically comprises 34% of the thruster mass (3.6 kg in this case). The specific mass of present PPU's is $\alpha_{PPU} = 8$ kg/kW of input power, though a reduction by a factor of five to ten may occur in the next decade. The specific mass of the thruster unit will depend on the input power, the total number of thrusters (including spares) N_t, the thruster mass m_t and the gimbal mass m_g. The complete powerplant specific mass is then

$$\alpha = \alpha_e + \alpha_{PPU} + N_t \cdot (m_t + m_g)/P_0. \tag{1}$$

In the last column of Table 1 are listed the estimated specific masses of hypothetical powerplants constructed from a single 30 cm thruster with 0.5 kW of input power supplied by the different radioisotope electric generators. These are probably conservative estimates since PPU specific masses will probably decrease, and electric generator efficiencies will likely increase with development. Assuming a total thruster efficiency of 75%, all the technologies beyond the standard RTG can give effective powerplant specific masses α/η_t in the range 100 to 200 kg/kW suitable for propelling small robotic probes.

Low-Thrust Program for Robotic Probes

To maximize the payload delivered for a mission along a low-thrust trajectory, both the rocket configuration (relative masses of powerplant and propellant) and the thrust program (magnitude and direction versus time) are typically optimized. Optimal programs generally involve large changes in the thrust and propellant velocity during the mission. For small robotic probes, ion thrusters with constant thrust and propellant velocity are adopted here because of their simplicity. This constrains the optimization method. For a power-limited vehicle with constant thrust, the propellant velocity and the ratio of powerplant mass to propellant mass are algebraically related. Maximizing the payload for a mission is then reduced to finding the optimal propellant velocity and the optimal thrust vectoring program.

For escape trajectories from the Solar System which only involve an acceleration phase, aligning the thrust along the velocity vector approximately maximizes the rate of energy change. The optimal thrust should actually be directed between the circumferential direction and the velocity vector during escape. Irving[14] and more recently Keaton[15] have illustrated that there is little improvement gained in payload fraction or escape time by using the optimal thrust program versus the simpler tangential thrust program. The tangential thrust program was adopted for the present study of REP powered probes.

The calculation of the prestellar mission trajectories to be described in the next section was done with a FORTRAN code incorporating a standard, double-precision differential equation solver to evolve heliocentric trajectories in the ecliptic. The simple computer code was not intended for detailed mission analysis but rather to allow a quick parameter study. The input parameters characterizing the ion rocket are the payload mass ratio M_L/M_0, the powerplant to propellant mass ratio $K = M_W/M_P$, the effective powerplant specific mass α/η_t and the total thrust time τ. Here $M_0 = M_L + M_W + M_P$ is the initial ion rocket mass after any chemical rockets are jettisoned, and η_t is the total ion thruster efficiency. The ratio M_W/M_P is related to the effective propellant velocity $v'_p = \eta_m v_p$ by $M_W/M_P = (v'_p/v'_c)^2$, where η_m is the mass utilization efficiency of the thruster and $v'_c = (2\tau\eta_t/\alpha)^{1/2}$ is the characteristic velocity of the propulsion system.[9]

Using the constant, tangential thrust program for a sample of prestellar missions, it was found that the propellant velocity which maximized the payload fraction for a given final trajectory was typically within five percent of the optimal propellant velocity given by the field-free rocket equations. Indeed using the field-free value for the optimal propellant velocity resulted in a negligible change in performance in all cases studied. The optimal rocket configuration which maximizes the payload fraction for this thrust program is then given by the approximate field-free formula[1]

$$K_{opt} = 0.26(1 + 2\ln(1 + 3(M_L/M_0)_{max})). \tag{2}$$

For example for a payload fraction of 0.25, the optimal powerplant to propellant mass ratio is 0.55.

Missions to Near-Interstellar Space

Missions have been proposed to send probes out of the Solar System in order to explore the heliopause (~100 AU) and near-interstellar space.[16] There are many scientific reasons to send probes several hundred AU from the Sun, some of which were mentioned in the Introduction. The Voyager probes, launched in the late 1970's to explore the outer planets, are leaving the Solar System at about 3 AU/year and may reach the heliopause between 2005 and 2010. However to reach distances of several hundred AU in less than fifty years requires hyperbolic excess velocities of 10 to 20 AU/year. To perform these demanding missions with chemical propulsion and gravity assist alone would involve prodigious amounts of propellant. Low-thrust electric propulsion over many years can provide the extra velocity and deliver a large payload fraction to interstellar space.

The flight time to reach 100 AU and the hyperbolic excess velocity V_∞ achievable using electric propulsion alone is considered first. Only trajectories in the ecliptic plane are considered. Missions out of the ecliptic can be accomplished with a close flyby of Jupiter to bend the trajectory out of the orbital plane. The range of powerplant specific mass studied is 20 to 200 kg/kW to allow a comparison of the performance of near-term REP and hypothetical, low specific mass powerplants of the future. Escape from low Earth orbit (LEO) is economically provided by a disposable chemical rocket since the mass of the entire REP powered probe is envisioned to be only a few hundred kilograms. Low-thrust ion propulsion starts immediately after Earth escape with the thrust vector always aligned with the velocity vector. The thrust time is ten years assuming that this is the reliability limit of the propulsion system. Shortening the thrust time can reduce the flight time to reach 100 AU slightly since the acceleration is increased ($\propto \tau^{-1/2}$) but at the expense of reducing the hyperbolic excess velocity. The converse is true if the thrust time is lengthened.

Mass ratios are refered to the initial mass of the ion rocket, M_0, after the chemical rocket is jettisoned. To determine the initial mass in low-Earth orbit, M_{LEO}, needed for the mission, the dry mass of the disposable chemical rocket, M_D, must be specified in the relation $(M_0+M_D)/M_{LEO} = \exp(-\Delta v/v_p)$, where Δv is the velocity change needed to leave orbit, and v_p is the propellant velocity of the chemical rocket. The dry mass of orbital transfer vehicles is typically about 15% of the chemical propellant mass.[17] The mass ratio in LEO is then

$$M_{LEO}/M_0 = \exp(\Delta v/v_p)/(1.15 - 0.15\exp(\Delta v/v_p)). \tag{3}$$

As an example, for minimal escape velocity from a 320 kilometer orbit ($\Delta v = 3.2$ km/s) using an oxygen-hydrogen rocket with specific impulse $I_{sp} = 450$ sec, $M_{LEO}/M_0 = 2.46$.

Figures 1 and 2 show the flight time to reach 100 AU and the hyperbolic excess velocity, respectively, as a function of α/η_t for different payload ratios M_L/M_0. Minimal Earth escape velocity is assumed so the probe will have no excess velocity relative to the Earth ($u_o = 0$), and the electric propulsion system provides all the energy for the solar escape trajectory to interstellar space. For $M_L/M_0 = 0.25$, flight times of 12 years to 23 years are achievable for specific masses in the range 20 to 100 kg/kW. The hyperbolic excess velocity varies from 17 to 6 AU/year in this specific mass range. There is a high penalty in flight time for payload ratios above 0.5 and little advantage in reducing the payload ratio below 0.25. The flight time scales roughly like $\alpha^{1/3}$, and the hyperbolic excess velocity scales like $\alpha^{-1/2}$ for small payload ratios. If only minimal Earth escape velocity is supplied to the probe, powerplant specific masses above 100 kg/kW, characteristic of near-term REP, yield very long flight times and are probably not of interest for missions to 100 AU. Flight times to reach 550 AU, the position of the first gravitational lens focus of the Sun, would be a century or more using these powerplants alone.

Reducing the flight time to reach 100 AU is difficult because of the weak dependence of this quantity on powerplant specific mass. Fast missions to several hundred AU are not possible with near-term REP unless the probe's energy can be substantially increased in the inner Solar System so as to boost the final hyperbolic excess velocity. Since the probe and its REP powerplant are assumed to have a mass of only a few hundred kilograms, this is easily done with an energetic combination of chemical propulsion and gravity assist. Low-thrust electric propulsion is used only after the probe has left the inner Solar System. This is analogous to rocket staging which enables low performance propulsion systems to produce large velocity changes. Because of its large mass and the relative ease of transfer from Earth, Jupiter is used for the primary gravity assist maneuver. Transfer to Jupiter is achieved by placing the probe on a so-called 1/2 resonance solar orbit (period 2 years, perihelion 1 AU) and performing a small Δv maneuver at aphelion (2.175 AU). This enables an Earth gravity assist (EGA) manuever 1.82 years after launch saving significant chemical propellant compared with a conventional Hohmann transfer. The transfer time to Jupiter following EGA is 1.72 years.

The Jupiter gravity assist (JGA) maneuver involves a close flyby at a radius of 1.05 Jupiter radii (75067 km) with a velocity of about 60 km/s. The encounter aligns the probe's outgoing velocity vector with that of the planet leading to an increased velocity relative to the Sun. It has long been recognized that at the high velocities achieved deep in a gravitational potential it is advantageous to perform a Δv maneuver (periapsis burn) which can dramatically increase a spacecraft's energy ($\Delta E/M = v\,\Delta v$) and hence its hyperbolic excess velocity. This requires delivering a significant mass fraction of storable, chemical propellant to Jupiter with the probe, which can be expensive in terms of initial rocket mass in low Earth orbit. For small probes however the periapsis burn at Jupiter is an economical means of augmenting the hyperbolic excess velocity.

The Δv of the periapsis burn at Jupiter is a mission parameter which is chosen to yield a desired increase in hyperbolic excess velocity. The periapsis Δv cannot be arbitrarily increased since this will make M_{LEO} unacceptably large. For the EGA-JGA transfer used here, there are two chemical propulsion units that will contribute to the mass ratio M_{LEO}/M_0. The first is the oxygen-hydrogen rocket in LEO that will transfer the probe plus its chemical rocket, with total mass M_i, to the 1/2 resonance orbit. Transfer to this orbit from LEO requires a $\Delta v = 4.24$ km/s and using Eqn.(3), the mass ratio is $M_{LEO}/M_i = 3.45$. The second mass ratio M_i/M_0 is determined by the total Δv that the probe's chemical rocket must produce up to and including the periapsis burn at Jupiter (after which it is jettisoned).

This second rocket is assumed to have a dry mass of 15% of its propellant mass and contain storable NTO-B_5H_9 liquid propellant with an I_{sp} =344 sec.[19] The total Δv required is the sum of the 1/2 resonance orbit aphelion burn (1.43 km/s), contingency for midcourse corrections (0.25 km/s) and the Jupiter periapsis burn. The mass ratio $M_{LEO}/M_0 = (M_{LEO}/M_i) \cdot (M_i/M_0)$ as a function of the Δv performed at Jupiter periapsis is shown in Figure 3. Note that the LEO mass ratio rises dramatically once the periapsis Δv exceeds 3 km/s. As an example, for a 3 km/s periapsis burn at Jupiter and an REP powered probe with mass $M_0 = 200$ kg, the required mass in LEO would be 5038 kg.

Figures 4 and 5 show the flight time to reach 100 AU and the hyperbolic excess velocity, respectively, as a function of α/η_t for different values of the Δv at Jupiter periapsis. The payload ratio of the REP powered probe is fixed at $M_L/M_0 = 0.25$ (e.g. 50 kg of payload for a 200 kg probe). The flight times include the 3.54 years needed for the Earth to Jupiter transit. If a Jupiter periapsis Δv of 3 km/s is supplied, the flight time to reach 100 AU using near-term REP with specific masses in the range 100 to 200 kg/kW is about 14 years. The hyperbolic excess velocities produced using these powerplants with the help of the powered JGA maneuver is 12 to 15 AU/year.

These hyperbolic excess velocities are high enough to consider practical robotic missions out to several hundred AU. Figure 6 shows the flight time to reach 550 AU (about 3 light-days) as a function of powerplant specific mass for different values of the Δv at Jupiter periapsis. For a periapsis Δv of 3 km/s, the flight times are 40 to 50 years using near-term REP powerplants. These times could be shortened somewhat by reducing the payload ratio M_L/M_0 below 0.25. The flight times are relatively insensitive to an increase of the periapsis Δv above 3 km/s. Note that even with hypothetical, advanced powerplants with α/η_t =20 kg/kW, the flight time to reach 550 AU can only be reduced to about 30 years.

Conclusion

The results of the present study indicate that near-term radioisotope electric propulsion is a viable candidate for sending small robotic probes on fast science missions out to several hundred astronomical units from the Sun. This propulsive capability will permit various space science themes to be investigated from new vantage points at 100 to 600 AU during extended 20 to 40 year missions.

Radioisotope electric propulsion (REP) systems based on multi-hundred watt radioisotope electric generators and ion thrusters are likely to have powerplant specific masses of 100 to 200 kg/kW if development continues during the next decade. Because of the need to avoid excessive launch masses and conserve strategic nuclear fuels, high specific mass powerplants are only practical for low power levels and small payloads. The trend in robotic spacecraft is toward complex probes of reduced mass so that 50 to 100 kg payloads will be capable of supporting several experiments. REP powered probes of total mass 200 to 400 kg can be envisioned with payload ratios of 0.25 and thrust powers of 250 to

500 watts. Of course reliable projected lifetimes for the new electric generators will be required before they can be used on long-duration electric propulsion missions. Derated ion thrusters already have projected lifetimes approaching a decade for these power levels.

Near-term REP powerplants used alone are capable of performing rendezvous missions to the outer planets in less than fifteen years but not fast prestellar missions to several hundred AU. The hyperbolic excess velocity of a small REP powered probe can be easily increased by using chemical propulsion and gravity assist while in the inner Solar System. Low-thrust propulsion is only used in the outer Solar System. This is analogous to rocket staging which enables low performance propulsion systems to produce large velocity changes. The scheme explored in this paper consists of transfer from low Earth orbit to a 1/2 resonance solar orbit with chemical propulsion, an Earth gravity assist maneuver and a powered Jupiter gravity assist maneuver to escape the inner Solar System at high velocity. Low-thrust electric propulsion commences after the Jupiter encounter once the chemical rocket has been jettisoned. Because a large mass fraction of storable chemical propellant must be delivered to Jupiter, this method is only economical for probes with a total mass of 200 to 400 kg. The ratio of rocket mass in LEO to probe mass rises dramatically for values of the Jupiter periapsis $\Delta v > 3$ km/s. For example to provide a periapsis Δv of 3 km/s for a 200 kg probe requires a mass of 5038 kg in LEO.

For a Δv of 3 km/s provided at Jupiter periapsis followed by a ten year low-thrust period, hyperbolic excess velocities of 12 to 15 AU/year can be achieved by REP powered probes with a payload ratio $M_L/M_0 = 0.25$ and specific masses of 100 to 200 kg/kW. The flight times to reach 550 AU, the position of the first gravitational lens focus of the Sun, are 40 to 50 years. The flight time is relatively insensitive to an increase of the Jupiter periapsis Δv above 3 km/s but could be shortened somewhat by reducing the payload ratio. The flight time is also insensitive to reductions in the powerplant specific mass. This suggests that unless one expects advanced powerplants with specific masses below 20 kg/kW to be available for launch in the next decade, it is advantageous to begin a deep space science program using high specific mass REP powerplants.

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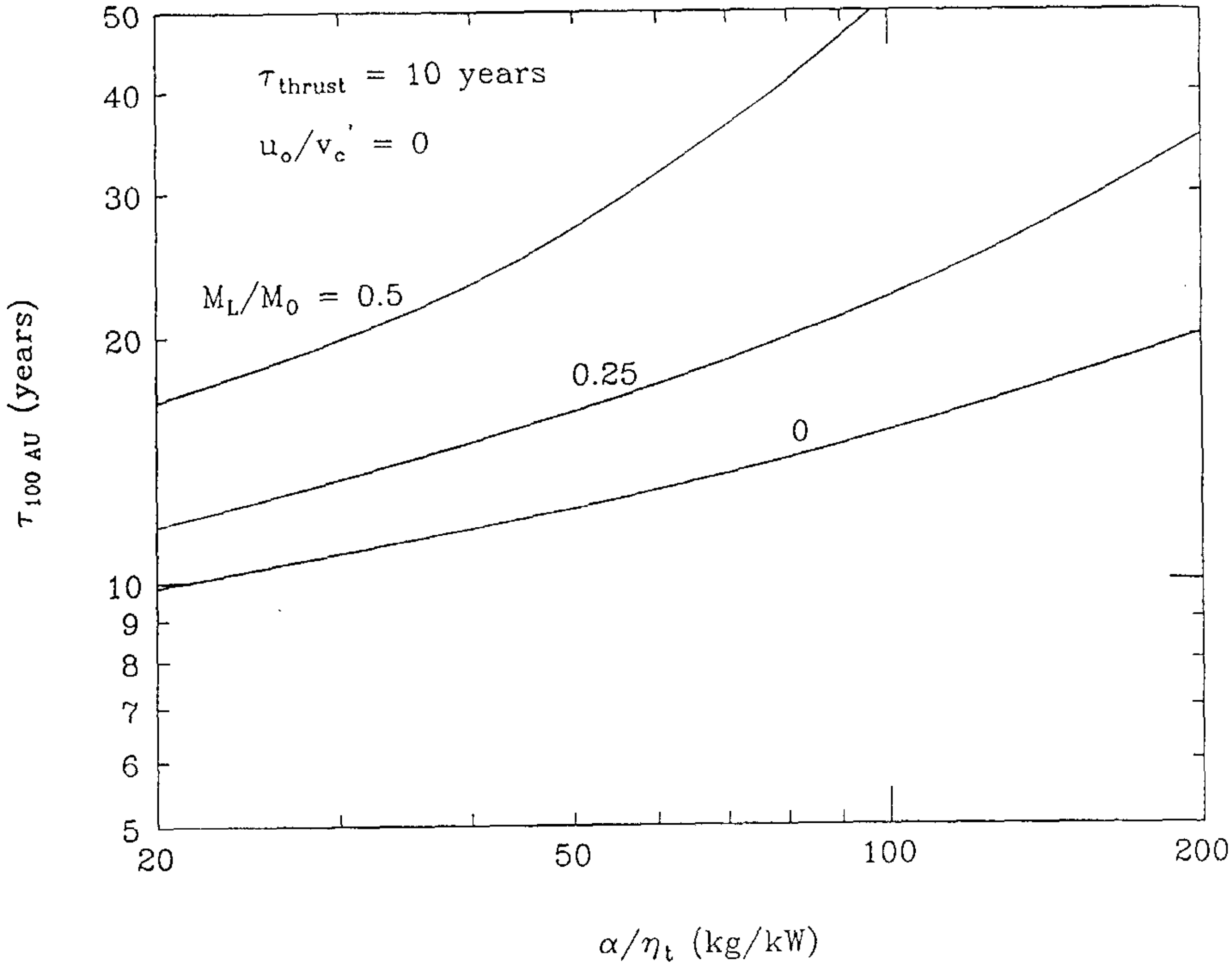

Figure 1: Flight time to reach 100 AU as a function of the powerplant specific mass for different payload mass ratios and no Earth escape excess velocity (u_o = 0). The low-thrust period is ten years commencing after Earth escape.

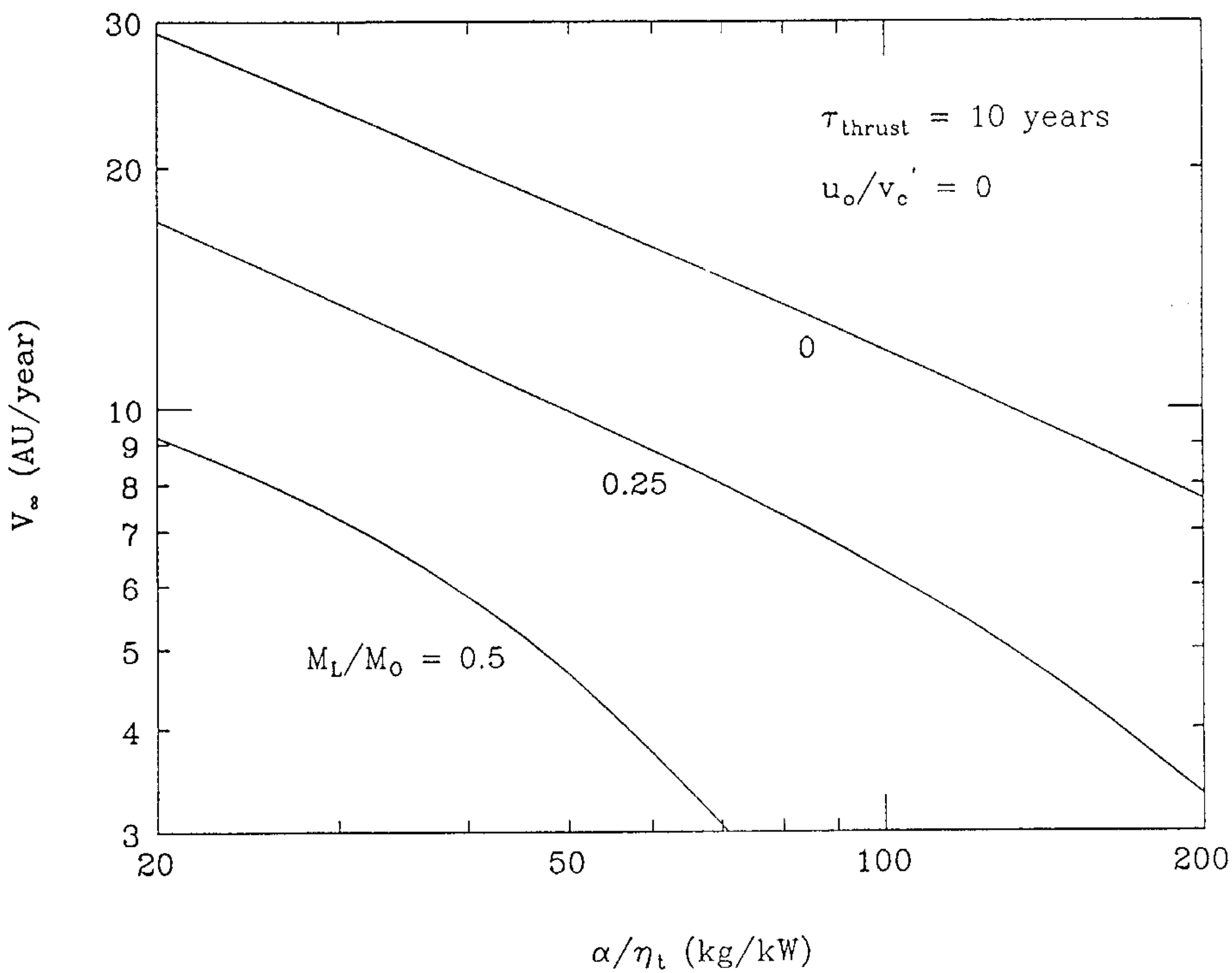

Figure 2: Hyperbolic excess velocity as a function of the powerplant specific mass for different payload mass ratios and no Earth escape excess velocity (u_o = 0). The low-thrust period is ten years commencing after Earth escape.

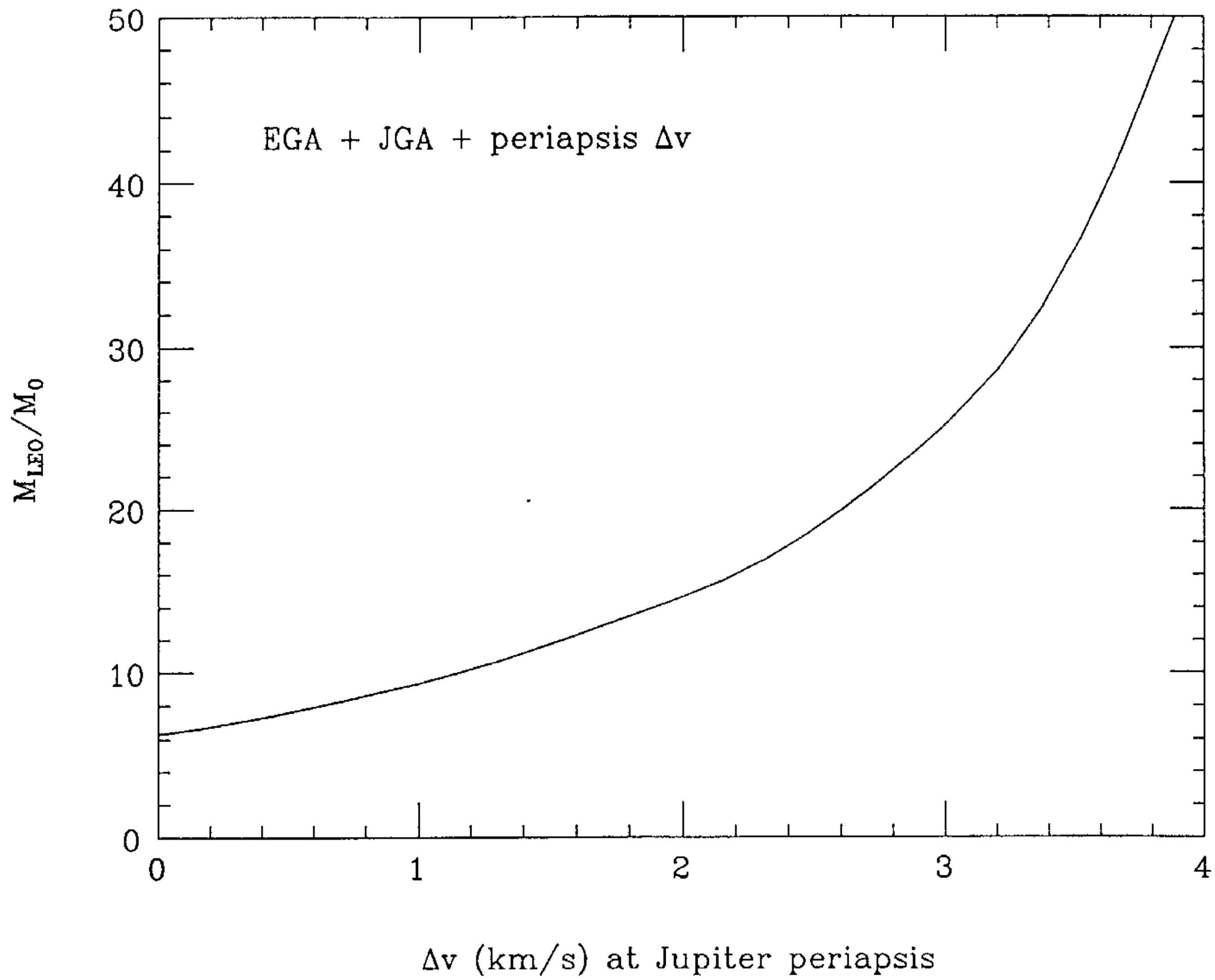

Figure 3: Ratio of the rocket mass in low Earth orbit M_{LEO} to the mass of the REP powered probe M_0 as a function of the velocity change Δv provided by the Jupiter periapsis burn. Transfer to Jupiter is affected by an Earth gravity assist maneuver.

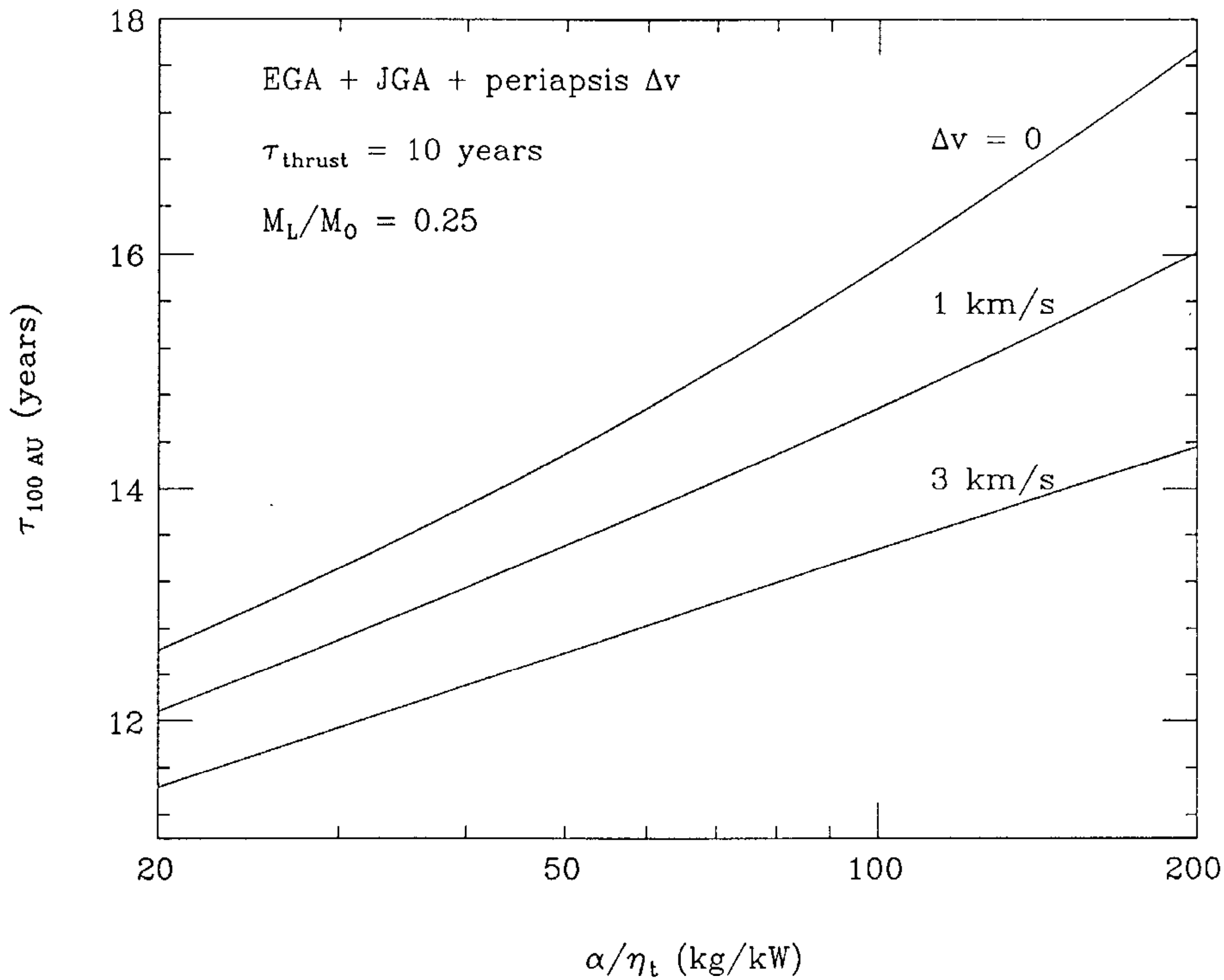

Figure 4: Flight time to reach 100 AU as a function of the powerplant specific mass for different values of the Jupiter periapsis Δv. The payload mass ratio is 0.25, and the low-thrust period is ten years commencing after the Jupiter encounter.

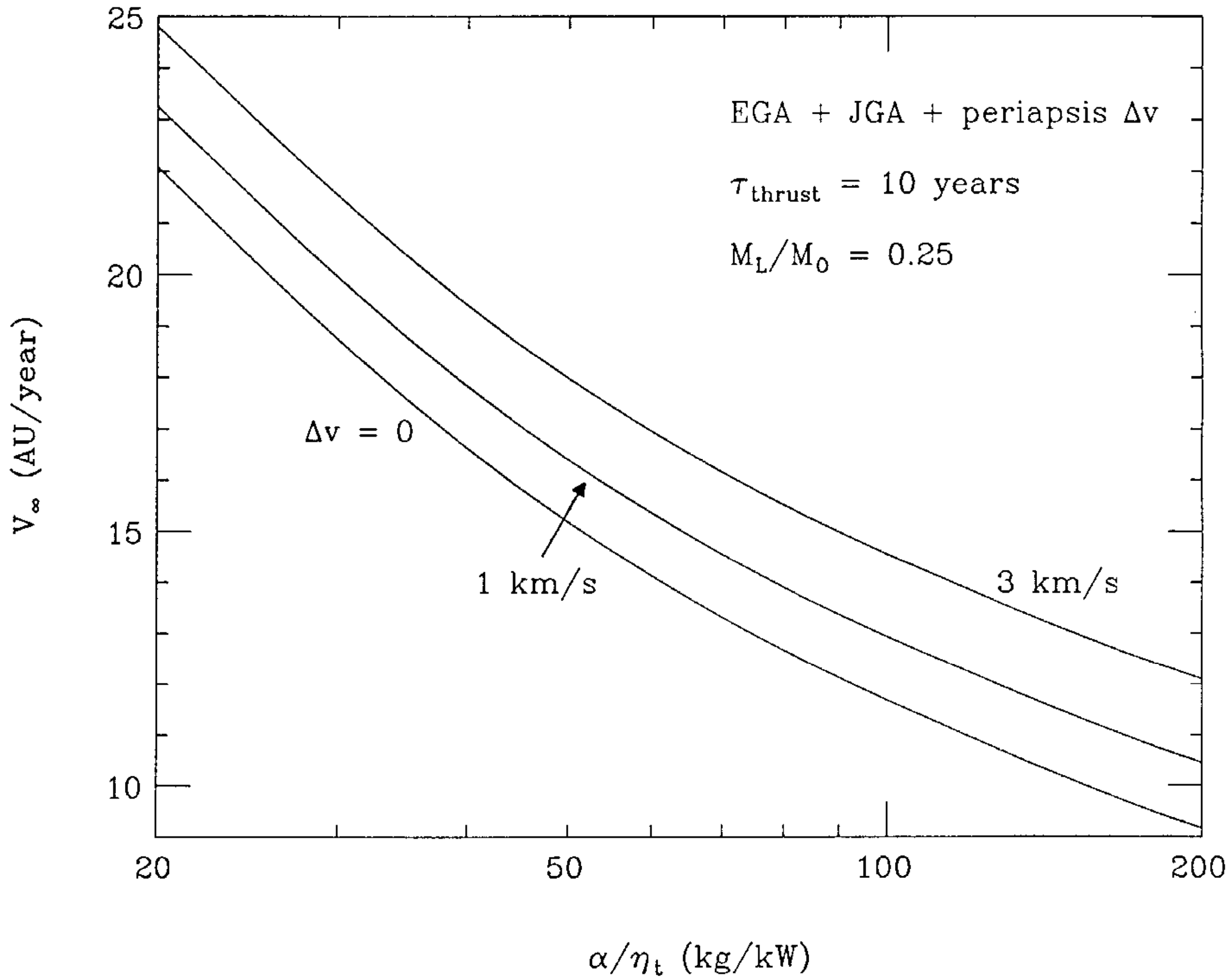

Figure 5: Hyperbolic excess velocity as a function of the powerplant specific mass for different values of the Jupiter periapsis Δv. The payload mass ratio is 0.25, and the low-thrust period is ten years commencing after the Jupiter encounter.

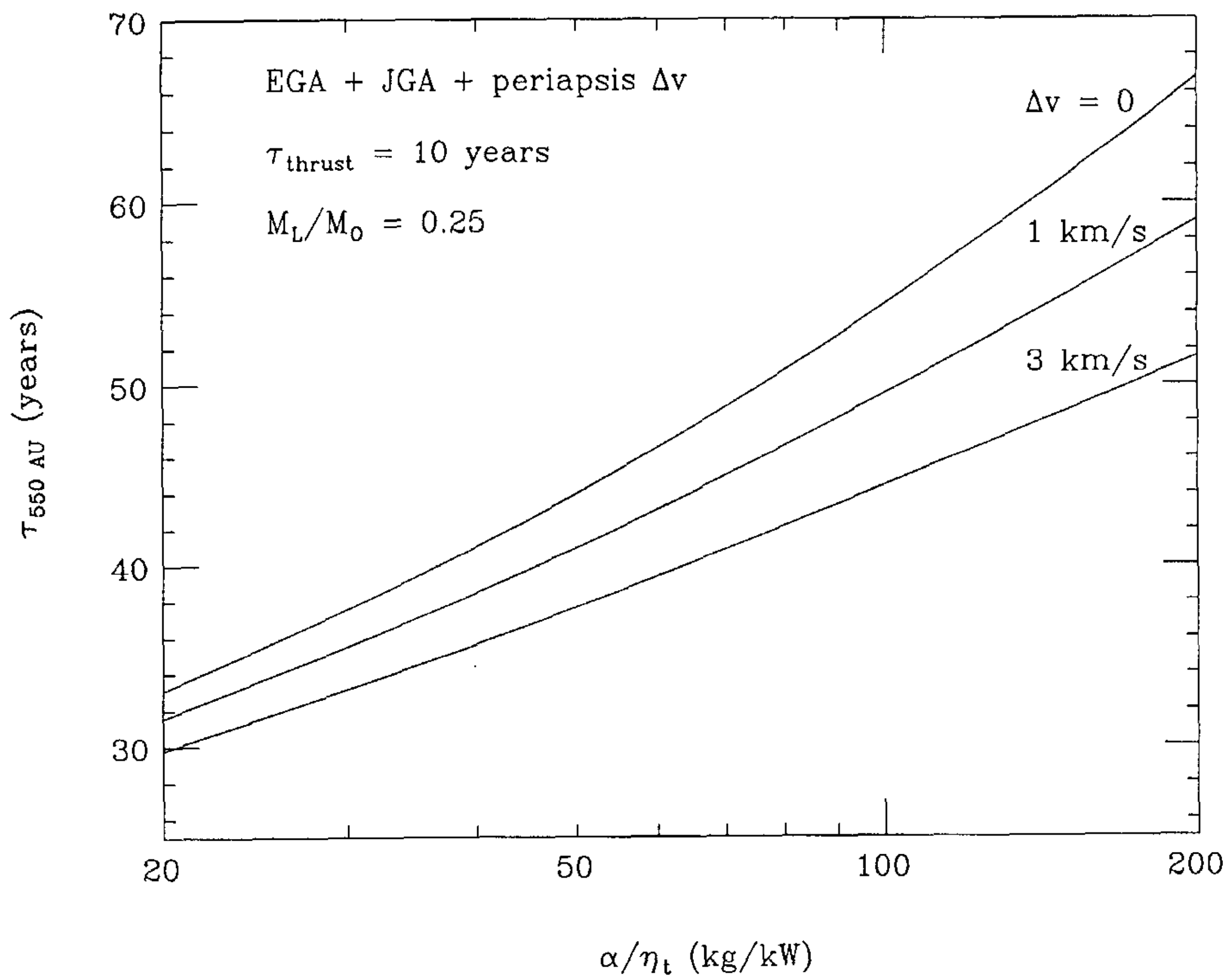

Figure 6: Flight time to reach 550 AU as a function of the powerplant specific mass for different values of the Jupiter periapsis Δv. The payload mass ratio is 0.25, and the low-thrust period is ten years commencing after the Jupiter encounter.