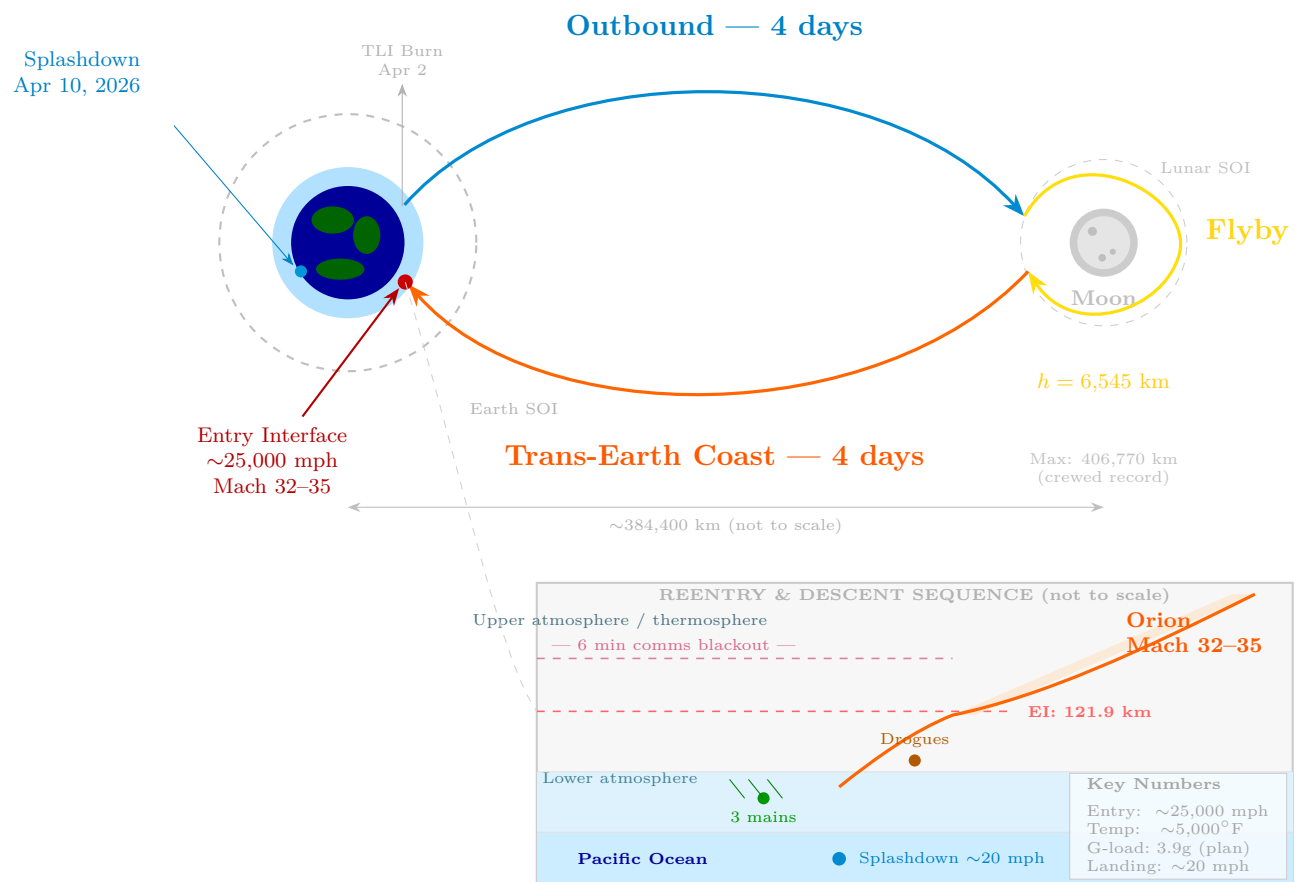


# The Mathematics of Artemis II's Return

*A Quantitative Analysis of Reentry, Descent, and Splashdown*

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**Mission:** Artemis II · April 1–10, 2026  
**Vehicle:** NASA Orion / SLS Block 1  
**Crew:** Wiseman, Glover, Koch, Hansen

**Max distance:** 252,756 mi (new record)  
**Entry speed:**  $\sim 25,000 \text{ mph}$  (Mach 32–35)  
**Splashdown:** 8:07 p.m. EDT, Pacific Ocean

## Abstract

**Introduction.** This paper presents a comprehensive quantitative analysis of the return trajectory of the Artemis II mission, independently completed by me, Christopher Gilmartin, an undergraduate student at New York University, double-majoring in Physics and Math. It aims to provide an overview of the mission's full technical physics while remaining accessible to the average reader.

**Mission Description.** On April 1, 2026, NASA's Artemis II mission launched with three NASA astronauts and one Canadian Space Agency astronaut: Reid Wiseman, Victor Glover, Christina Koch, and Jeremy Hansen, respectively. Artemis II is the first crewed mission to travel beyond low Earth orbit since the Apollo 17 mission in December of 1972. Artemis II returned to Earth after a journey of more than 694,000 miles on April 10, 2026, at 8:07 p.m. EDT with a splashdown southwest of San Diego, California. Along the way, the crew surpassed Apollo 13's record for the farthest humans have traveled from Earth, reaching a maximum distance of 252,756 miles.

**Physics Pipeline of Return.** The physics pipeline begins with the mechanics of the free-return circumlunar trajectory and the use of the patched-conic approximation to represent it, followed by the trans-Earth coast, the execution of three return trajectory correction maneuvers that guide Orion towards the entry corridor and splashdown point, crossing the entry interface at approximately 25,000 miles per hour (comparable to the Apollo lunar returns), experiencing the aerothermodynamics of reentry at speeds of Mach 32 to Mach 35, experiencing the thermochemistry of the AVCOAT ablative thermal protection system, and finally utilizing the three-phase parachute deployment for a soft landing into the ocean. In this paper, governing equations will be derived from first principles using parameters specific to Artemis II, and the results will then be compared to actual mission data. An energy accounting between the entry interface and splashdown is also presented.

Of particular interest in this paper is Artemis II's modified entry path, which replaced the long skip used on the previous mission with a much shorter one to ensure that the heat shield would function correctly, after it was found to have experienced problems during the uncrewed Artemis I mission in 2022.

All research, derivations, and analyses in this paper were conducted independently of coursework or institution-assigned projects.

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# 1 Free-Return Trajectories and the Patched-Conic Approximation

Before discussing how Orion returns to the launch site, let's solidify our understanding of exactly what enables Artemis II to get there automatically. This mission follows a free-return trajectory. In simple terms, if all the engines of Orion fail immediately after the Trans-Lunar Injection (TLI) burn, the crew will be able to reach the vicinity of the Moon and return to Earth anyway. This is an objective mathematical guarantee provided by the orbital mechanics of the entire mission, starting from the moment the TLI burn ended on April 2nd.

But, how exactly does that work?

## 1.1 The Two-Body Problem and the Vis-Viva Equation

Let's start from scratch and consider the case of a single body and a single spacecraft, with no other external forces acting on them. In this case, the space probe will travel along a conic trajectory that may be an ellipse (bound state), a parabola (escape velocity), or a hyperbola (the spacecraft escapes and never returns to the body). The total specific orbital energy  $\varepsilon$  of the spacecraft is equal to:

$$\varepsilon = \frac{v^2}{2} - \frac{\mu}{r} \quad (1)$$

where  $v$  is the speed of the spacecraft,  $r$  is the distance between the spacecraft and the center of the gravitational body, and  $\mu = GM$  is the standard gravitational parameter of this body ( $G$  is the universal gravitational constant,  $M$  is the mass of the body). Thus, for example,  $\mu_E \approx 3.986 \times 10^5 \text{ km}^3/\text{s}^2$  for Earth and  $\mu_M = 4.905 \times 10^3 \text{ km}^3/\text{s}^2$  for the Moon.

Consequently, if  $\varepsilon < 0$ , the spacecraft will move along an elliptical trajectory (it cannot leave its orbit); if  $\varepsilon = 0$ , along a parabola; if  $\varepsilon > 0$ , along a hyperbola. To find the speed of an object traveling along the orbit at any point, knowing its radius, the following equation can be derived from the above formula: The vis-viva equation:

$$v^2 = \mu \left( \frac{2}{r} - \frac{1}{a} \right) \quad (2)$$

where  $a$  is the semi-major axis of the elliptical orbit. This formula is extremely important in orbital mechanics, and we will be using it frequently throughout this paper.

So, what happens with Orion in the Artemis II case? As soon as the TLI burn is finished, the spacecraft enters an orbit characterized by such a semi-major axis that the orbit crosses the Earth's atmosphere at one of the apsides and the Moon's sphere of influence at the other.

## 1.2 Why One Gravitational Body Is Not Enough

Unfortunately, this problem is made much trickier by the fact that the Earth, Moon, and Sun affect the spacecraft's movement simultaneously. It turns out that finding a solution for the

spacecraft's motion considering all these factors is rather difficult, since it does not have a closed-form solution (Henri Poincaré proved that in 1890). Consequently, when constructing orbits and planning flights, people use the *patched-conic approach*, an approximation that divides the entire trajectory into several segments.

It is based on the following assumption: within certain zones surrounding the gravitational bodies, we consider the orbit to be a solution to the two-body problem relative to a specific body only. These zones are called **spheres of influence** (SOI). Within the sphere of influence of the Earth, the orbit is described using classical Kepler's laws. Inside the SOI of the Moon, we switch to solving the two-body problem with respect to the Moon. And at the borders of the spheres of influence, we patch two solutions together, ensuring that the position vectors and velocities are continuous at the switching point.

The radius of the sphere of influence of the Moon relative to the Earth is calculated as follows:

$$r_{\text{SOI}} = a_{\text{body}} \left( \frac{m_{\text{body}}}{m_{\text{primary}}} \right)^{2/5} \quad (3)$$

Substituting the appropriate values:

$$\begin{aligned} r_{\text{SOI}} &= 384,400 \text{ km} \times \left( \frac{7.342 \times 10^{22} \text{ kg}}{5.972 \times 10^{24} \text{ kg}} \right)^{2/5} \\ &= 384,400 \text{ km} \times (0.01229)^{0.4} \\ &\approx 66,100 \text{ km} \end{aligned} \quad (4)$$

Orion crossed into the Moon's sphere of influence on flight day 6, shortly before the April 6 flyby.

### 1.3 The Free-Return Trajectory of Artemis II

As mentioned above, the orbit in Artemis II follows a circumlunar trajectory, i.e., it allows the spacecraft to swing around the Moon and then return to Earth due to gravity alone. On 6 April 2026, Orion flew by, approaching the lunar surface to a height of 6,545 km (4,067 miles) above it at the closest approach (known as pericynthion) at 23:00 UTC. This distance is deliberately selected to provide the necessary trajectory parameters to bring the vehicle to the splashdown zone off San Diego four days after the flyby.

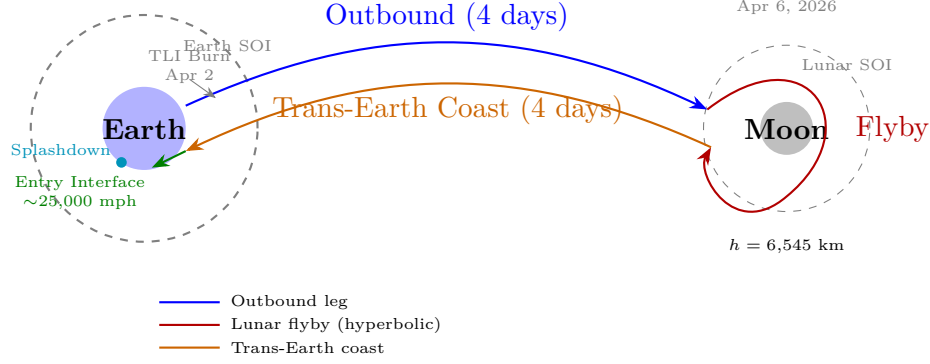


Figure 1: Schematic of the Artemis II free-return circumlunar trajectory (not to scale). Orion departs Earth on the TLI burn (blue), swings around the Moon on a hyperbolic arc (red), and coasts back to atmospheric entry interface off the California coast (orange). The return needs no major propulsive burn, only small trajectory corrections.

Within the Moon’s sphere of influence, the trajectory in the Moon’s reference frame is a hyperbola. The spacecraft arrives at the Moon in hyperbolic motion, swings around it, and departs in hyperbolic motion as well.

The excess speed  $v_\infty$  is defined as the asymptotic value of the speed when moving away from the Moon. It is calculated using the energy balance relation  $\varepsilon = v_\infty^2/2 = v^2/2 - \mu_M/r$ , where  $r$  is the distance to the Moon. At the periapsis of the hyperbola, the speed of the spacecraft reaches its maximum:

$$\frac{v_\infty^2}{2} = \frac{v_{\text{peri}}^2}{2} - \frac{\mu_M}{r_{\text{peri}}} \quad (5)$$

$$v_{\text{peri}} = \sqrt{v_\infty^2 + \frac{2\mu_M}{r_{\text{peri}}}} \quad (6)$$

For the Artemis II mission, with a periapsis height of 6,545 km above the surface (giving  $r_{\text{peri}} = 1,737 + 6,545 = 8,282$  km) and an assumed hyperbolic excess velocity relative to the Moon of  $v_\infty \approx 0.84$  km/s (a representative value; NASA has not published it), the periapsis speed equals:

$$\begin{aligned} v_{\text{peri}} &= \sqrt{(0.84)^2 + \frac{2 \times 4,905}{8,282}} \\ &= \sqrt{0.706 + 1.185} \\ &\approx \sqrt{1.891} \approx 1.375 \text{ km/s} \end{aligned} \quad (7)$$

This is well below the Moon’s escape velocity of  $\approx 2.38$  km/s at the lunar surface, but at this altitude, the local escape velocity is  $\sqrt{2\mu_M/r_{\text{peri}}} = \sqrt{2 \times 4905/8282} \approx 1.09$  km/s, meaning

Orion's periapsis speed exceeded local escape velocity, confirming it was never at risk of capture.

## 1.4 How the Moon Bends the Trajectory

The bending of the trajectory in the Moon's reference frame looks somewhat peculiar. Since energy is conserved in the Moon's reference frame, the spacecraft leaves the Moon with the same speed it arrived with, while the direction of motion changes dramatically. In other words, the Moon rotates the spacecraft's velocity vector by the angle  $\delta$ :

$$\sin\left(\frac{\delta}{2}\right) = \frac{1}{e} \quad (8)$$

where  $e$  is the eccentricity of the hyperbolic orbit of the spacecraft relative to the Moon, determined from:

$$e = 1 + \frac{r_{\text{peri}} v_{\infty}^2}{\mu_M} \quad (9)$$

For Artemis II:

$$\begin{aligned} e &= 1 + \frac{8,282 \times (0.84)^2}{4,905} \\ &= 1 + \frac{8,282 \times 0.706}{4,905} \\ &\approx 1 + 1.19 \approx 2.19 \end{aligned} \quad (10)$$

$$\begin{aligned} \sin\left(\frac{\delta}{2}\right) &= \frac{1}{2.19} \approx 0.457 \\ \frac{\delta}{2} &\approx 27.2^\circ \\ \delta &\approx 54.4^\circ \end{aligned} \quad (11)$$

The rotation of the velocity vector by an angle of about  $54.4^\circ$  is what makes Artemis II fly back to Earth instead of flying out into space. When we return to Earth-centric coordinates, we can see that the spacecraft travels along an elliptical (though hyperbolic from the Moon's point of view) trajectory that intersects with Earth's atmosphere at closest approach. The four-day trans-Earth coast begins.

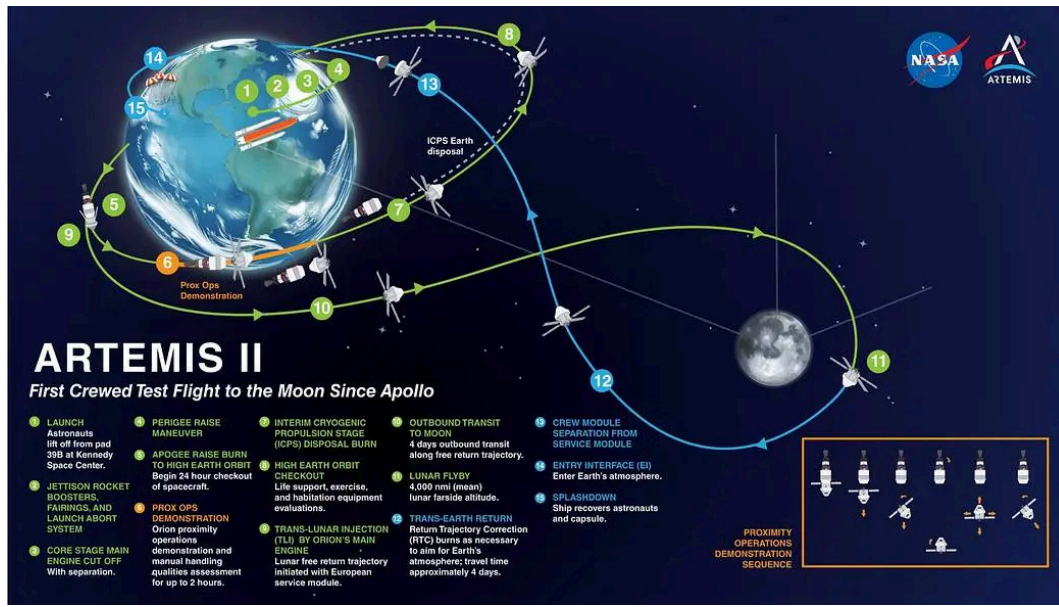


Figure 2: Official NASA mission graphic depicting the Artemis II free-return circumlunar trajectory. The figure-8 shaped path carries Orion from Kennedy Space Center, around the far side of the Moon, and back to splashdown in the Pacific Ocean. *Credit: NASA.*

## Key Takeaways

Artemis II's free-return trajectory was designed so that, even if every engine failed after it left Earth orbit, a return path to Earth was always assured. Around a single body, every spacecraft travels along a conic section (an ellipse, a parabola or a hyperbola), depending on its total energy. With too little energy to escape, it keeps orbiting forever; with just enough, it can barely escape; and with more than that, it escapes and keeps some leftover speed even far from the body.

This problem is very difficult to solve when more than two bodies are involved because the gravitational pull of Earth, the Moon, and the Sun combines to create a highly complex system. In fact, no general closed-form solution exists, as Poincaré showed in 1890. To work around this problem, engineers use what's known as a patched conic approximation: assuming that one body dominates at any one time, and solving the easier problem.

When Orion reached the Moon's sphere of influence on Day 6, the Moon dominated over Earth, causing the craft to move on a hyperbolic trajectory around it.

## 2 Trans-Earth Coast and Trajectory Correction Maneuvers

At the time Orion exits the sphere of influence of the Moon, it is no longer a problem in lunar gravity. It is now back to being a problem in Earth’s gravity, and quite a neat one at that. The spacecraft is in an elliptical orbit with a perigee within Earth’s atmosphere. Left alone, it would reach that perigee in 4 days. And that is the promise of a guaranteed free return trajectory, described in Section 1. However, while hitting the perigee is quite simple, hitting a small target area in the Pacific Ocean off San Diego is not, and it is exactly what the trans-Earth coast aims to do.

The 4-day coast from leaving the Moon’s sphere of influence is an actively observed phase, where the mission control team at Johnson Space Center continuously tracks Orion’s state vector, i.e., its position and velocity in three-dimensional space, comparing it to the planned one, using tracking data from the Deep Space Network. Deviations happen and get accumulated over time. The gravity of the Moon is not symmetrical. There is solar radiation pressure acting on the spacecraft. The atmospheric density models used in the entry interface calculations are not perfect. All these perturbations have to be compensated, and the trajectory correction maneuvers (TCMs) exist for exactly that purpose.



Figure 3: NASA’s Deep Space Network (DSN) dish array at Goldstone, California. During Artemis II’s trans-Earth coast, DSN stations provide the tracking data flight controllers use to determine Orion’s state vector and detect small trajectory deviations. *Credit: NASA/JPL-Caltech.*

## 2.1 State Vector and Orbital Perturbations

In the idealistic two-body problem discussed in Section 1, the coast from leaving the Moon's sphere of influence to reaching the entry interface will be governed by the trajectory of a pure Keplerian ellipse, which will need no corrections. In real life, however, the true equation of motion of the spacecraft in its trans-Earth coast will have the following form:

$$\ddot{\mathbf{r}} = -\frac{\mu_E}{r^3}\mathbf{r} + \mathbf{a}_{\text{pert}} \quad (12)$$

Here,  $\mathbf{r}$  is the position vector of Orion relative to Earth's center,  $\mu_E$  is Earth's gravitational constant, and  $\mathbf{a}_{\text{pert}}$  is the sum of perturbation accelerations caused by various perturbing factors. The major perturbation terms in the trans-Earth coast are:

$$\mathbf{a}_{\text{pert}} = \mathbf{a}_{\text{Moon}} + \mathbf{a}_{\text{Sun}} + \mathbf{a}_{\text{SRP}} + \mathbf{a}_{J_2} \quad (13)$$

$\mathbf{a}_{\text{Moon}}$  is the Moon's third-body gravitational acceleration on Orion. Even having left the Moon's sphere of influence, the Moon is still a perturbing body, just a weaker one.  $\mathbf{a}_{\text{Sun}}$  stands for the third-body gravitational acceleration due to the Sun, which has a similar effect as well.  $\mathbf{a}_{\text{SRP}}$  corresponds to the solar radiation pressure on the spacecraft. Lastly,  $\mathbf{a}_{J_2}$  is the Earth's oblateness effect.

Third body acceleration due to a body of mass  $m_3$  located at the position vector  $\mathbf{r}_3$  relative to the Earth's center is given by the equation:

$$\mathbf{a}_{3\text{body}} = Gm_3 \left[ \frac{\mathbf{r}_3 - \mathbf{r}}{|\mathbf{r}_3 - \mathbf{r}|^3} - \frac{\mathbf{r}_3}{|\mathbf{r}_3|^3} \right] \quad (14)$$

The first term in this expression represents the direct gravitational force exerted on the spacecraft by the third body, and the second term subtracts the acceleration of Earth due to the same force, which is necessary because our coordinate origin, Earth's center, is itself accelerated by the third body and is therefore not an inertial point of reference.

The acceleration of Orion caused by the solar radiation pressure is:

$$\mathbf{a}_{\text{SRP}} = -\frac{P_{\text{SR}} \cdot C_R \cdot A}{m} \quad (15)$$

Here,  $P_{\text{SR}} = 4.56 \times 10^{-6} \text{ N/m}^2$  is solar radiation pressure at a distance of 1 AU,  $C_R$  is the reflectivity coefficient (assumed here to be about 1.3),  $A$  is the cross-sectional area of the spacecraft facing the Sun (estimated to be around 11 square meters), and  $m$  is Orion's mass (about 25,900 kilograms after TLI, since the service module stays attached until shortly before entry). This gives us a tiny acceleration of a few times  $10^{-9} \text{ m/s}^2$ . Left uncorrected over the 4 days ( $\frac{1}{2}at^2$  with  $t \approx 3.5 \times 10^5 \text{ s}$ ), it adds up to roughly 150 meters of positional deviation at the entry interface (more if the solar arrays are counted in  $A$ ).

## 2.2 Delta-V and the Rocket Equation

Whenever it happens to be necessary, Orion uses thrusters on its European Service Module, such as its eight auxiliary engines, to apply a small velocity increment known as delta-v ( $\Delta v$ ). The rocket equation, which governs this process, is well-known:

$$\Delta v = v_e \times \ln\left(\frac{m_0}{m_f}\right) \quad (16)$$

Here,  $v_e$  is the effective exhaust velocity of the propellant,  $m_0$  is the mass of the spacecraft before the burn, and  $m_f$  is its mass after the burn. Thus, we know that mass lost due to the burn is equal to  $\Delta m = m_0 - m_f$ . Solving the rocket equation for this value gives:

$$\Delta m = m_0 \times \left(1 - e^{-\Delta v/v_e}\right) \quad (17)$$

The auxiliary engines are Aerojet Rocketdyne R-4D-11 thrusters with a specific impulse  $I_{sp}$  of roughly 312 seconds, which means that their effective exhaust velocity is:

$$v_e = I_{sp} \times g_0 = 312 \times 9.81 \approx 3,061 \text{ m/s} \quad (18)$$

Using this value, let's estimate how much fuel will be needed for applying a 1 m/s change of velocity to Orion's trajectory, assuming that the spacecraft's total mass is around 25,900 kg (NASA's post-TLI figure):

$$\begin{aligned} \Delta m &= 25,900 \times \left(1 - e^{-1/3061}\right) \\ &\approx 25,900 \times 0.000327 \\ &\approx 8.5 \text{ kg} \end{aligned} \quad (19)$$

Hence, a 1-meter-per-second correction requires around 8.5 kilograms of propellant. Clearly, these are quite tiny burns in the grand scheme of things. Indeed, the main task of the trans-Earth coast is performed by the free-return trajectory; TCMs serve only for fine adjustment.

## 2.3 TCMs During Artemis II

The Artemis II mission had three planned outbound trajectory correction opportunities and three planned return TCMs, for a total of six planned burns after TLI. In practice, the first and second outbound corrections were both cancelled as flight controllers reviewed the post-TLI trajectory and determined Orion was already on such a precise path that no correction was needed. This is a direct consequence of the accuracy of the TLI burn itself: the AJ10 engine on the European Service Module fired for 5 minutes and 49 seconds and placed Orion on a trajectory so close to nominal that two of the three outbound correction

opportunities were waved off entirely. Only a single outbound correction burn was executed, on Flight Day 5: a 17.5-second thruster firing that refined Orion’s path to the Moon.

On the return leg, three trajectory correction burns were executed. The first return TCM fired on Flight Day 7 at 8:03 p.m. EDT, lasting 15 seconds and producing a velocity change of 1.6 ft/s (0.49 m/s). The second fired on Flight Day 9 at 10:53 p.m. EDT, lasting 9 seconds and producing 5.3 ft/s (1.62 m/s). The third and final return TCM fired on Flight Day 10 at 2:53 p.m. EDT, an 8-second burn that placed Orion precisely on course for the entry corridor. Each burn refined the aim a little further, and the later burns mattered most for splashdown precision because they came closest to entry.

## 2.4 The Entry Corridor

All TCMs during the trans-Earth coast, as mentioned above, are intended to place the spacecraft into a specific orbit. The flight path angle  $\gamma$  at the entry interface defines the goal of this orbit. The entry interface for Orion is defined as an altitude of 400,000 feet above sea level, approximately 121.9 kilometers. It marks the conventional top of the part of the atmosphere that matters for Orion’s flight.

This flight path angle is a critical parameter because if the angle  $\gamma$  is too shallow (above  $-5.0^\circ$ ), the spacecraft will essentially skip off the top layer of the atmosphere and back out into a long elliptical orbit around Earth, with no guarantee of a safe return. Considering the crew onboard, this is obviously unacceptable. On the other hand, if  $\gamma$  is too steep (below  $-8.5^\circ$ ), it leads to excessive heat flux and deceleration loads. In this simplified picture, the safe range of flight angles is about 3.5 degrees wide (these bounds are representative values; Orion’s exact limits depend on its lift and guidance and are not published in this form). The whole purpose of TCMs during the trans-Earth coast is to make sure that Orion reaches the right flight path angle when it reaches the entry interface.

This angle can be defined in terms of the semi-major axis  $a$  and eccentricity  $e$  of the geocentric orbit of Orion at its intersection with the altitude of entry interface  $r_{\text{EI}} = R_E + 121.9 \text{ km}$  by the following formula:

$$\gamma = \arctan \left[ \frac{e \sin \nu}{1 + e \cos \nu} \right] \quad (20)$$

where  $\nu$  is the true anomaly at entry interface. This is the formula used by mission planners to derive orbital elements for a given target angle and, consequently, to calculate the trajectory. The values in this section lead back to that entry angle.

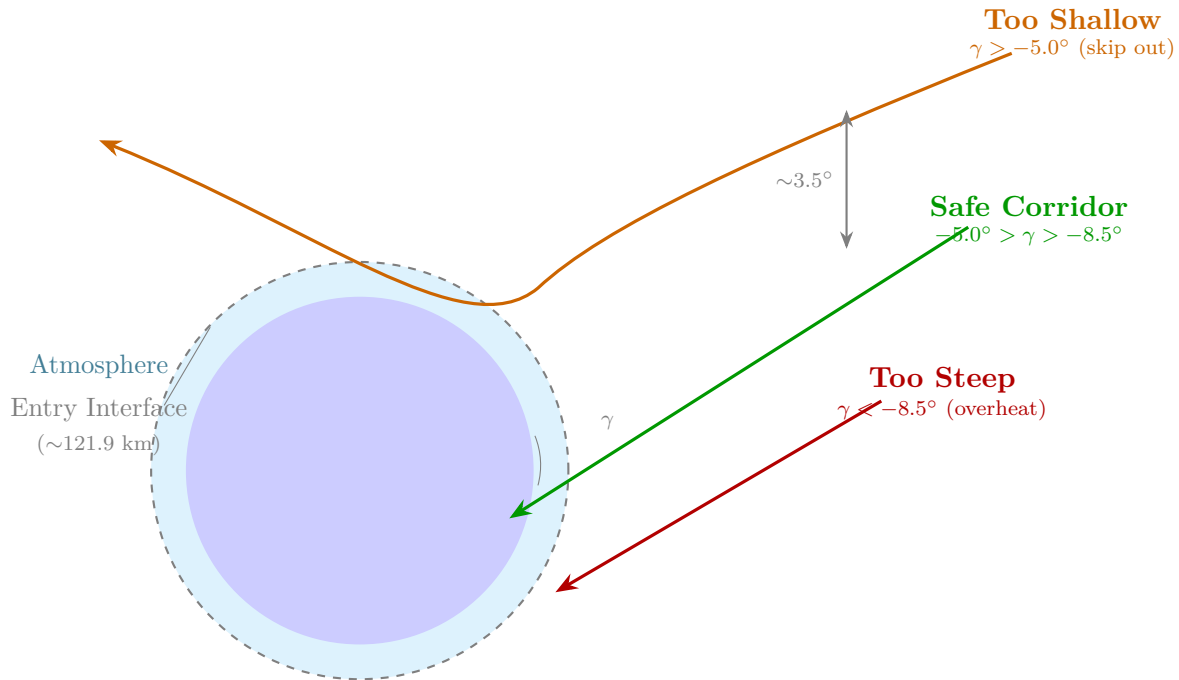


Figure 4: The Orion entry corridor at entry interface (121.9 km altitude). The safe flight path angle window is approximately 3.5 degrees wide. A shallower angle causes Orion to skip off the atmosphere; a steeper angle produces heating rates and deceleration loads beyond design limits. All TCMs during the trans-Earth coast exist to thread Orion through this corridor.

It should be noted that NASA also changed the entry itself for Artemis II, replacing Artemis I's long skip with a much shorter one. This is directly related to the heat shield fix discussed in Section 4 below.

## Key Takeaways

Once Orion broke free of the Moon's gravitational pull and began its descent back toward Earth, the easy bit was coming home, since a free-return trajectory ensured safe passage. The difficulty lay in ensuring that Orion landed in a specific spot over the Pacific Ocean, outside San Diego, rather than, say, somewhere over China.

To ensure that happened, the teams of controllers at the Johnson Space Center could determine the craft's location and speed to the last detail using the Deep Space Network, an international system of enormous radio dishes. Whenever Orion veered from its intended path, due to the inconsistent gravitation of the Moon, to solar radiation pressing against the hull, or any inaccuracies in atmospheric modeling, they would use small rockets to correct for the drift.

Trajectory correction maneuvers (TCMs) are quite small but incredibly precise: a one-meter-per-second correction uses only about 8.5 kg of propellant. Artemis II completed

four during its flight: an initial correction burn on Day 5 to tweak the trajectory past the Moon, followed by three correction burns to adjust the craft during its return on Days 7, 9, and 10. The latter burns were increasingly precise, homing in on the target splashdown site. The first two outbound burns were scrapped entirely since the trans-lunar injection burn was accurate enough.

Orion had to enter the atmosphere through a very narrow entry corridor just 3.5 degrees wide. Too shallow an angle, and it will simply skip into outer space; too steep, and the crew faces fatal temperatures and G-forces. Every TCM is performed for precisely this purpose.

### 3 Entry Interface and Hyperbolic Excess Velocity

Everything covered so far has focused on putting Orion in the correct position, at the correct angle, with the correct velocity. In Section 3, let's address this question explicitly by discussing the entry interface. The entry interface is the point at which the mission transitions from problems of orbital mechanics to those of atmospheric physics. The spacecraft stops behaving like something governed by gravity and Kepler's laws and starts becoming governed by drag, shockwaves, and thermodynamics. Entry interface is the most violent phase transition in human spaceflight, and it happens in the blink of an eye. Let's dive right into it!

Entry interface for Orion is defined as occurring at an altitude of 400,000 feet or roughly 121.9 km. This is not a physical limit to where the atmosphere ends; the transition from vacuum to atmosphere occurs much closer to Earth. The choice of 400,000 ft as the entry interface is based on the point at which the drag and lift forces acting on the craft become strong enough that they can no longer be considered perturbations to an otherwise Keplerian orbit. Above 121.9 km, you do orbital mechanics. Below it, you do aerodynamics. Of course, these are, again, oversimplifications of a very multifaceted problem in nature.

#### 3.1 Calculation of the Entry Velocity Using First Principles

The entry velocity for Orion is, of course, easily solved using the vis-viva equation from Section 1. To use vis-viva, all we need is the semi-major axis of the trans-Earth trajectory and the distance to Earth's center of gravity at the entry interface. The distance to Earth's center of gravity at the entry interface is:

$$r_{\text{EI}} = R_E + 121.9 \text{ km} = 6,371 + 121.9 = 6,492.9 \text{ km} \quad (21)$$

After its lunar flyby and gravitational slingshot maneuver, Orion follows an elliptical trans-Earth trajectory. This ellipse has Earth at one focus, and its perigee lies within Earth's atmosphere (i.e., the ellipse intersects it). Perigee occurs at a distance from Earth's center of gravity; let us call that distance  $r_{\text{peri}}$ . Apogee occurs far out, near lunar distance (we estimate it below). The semi-major axis is, therefore:

$$a = \frac{r_{\text{peri}} + r_{\text{apogee}}}{2} \quad (22)$$

However, this formulation comes with a caveat. The 'perigee' mentioned above is actually an imaginary construct; it is the point of lowest altitude if the spacecraft were to fall freely without atmosphere in the way. This means that we don't actually want to solve vis-viva at the perigee; we want to do it at the altitude of the entry interface. Vis-viva allows for this without issue:

$$v_{\text{EI}}^2 = \mu_E \left( \frac{2}{r_{\text{EI}}} - \frac{1}{a} \right) \quad (23)$$

The apogee of the trans-Earth trajectory of Artemis II happens at approximately the same place where the lunar SOI boundary would be: 376,000 km from Earth’s center of gravity (a rough choice, but the answer barely depends on it: using Orion’s actual maximum distance of 406,771 km instead changes  $v_{\text{EI}}$  by less than 0.1%). For the perigee of this hypothetical unperturbed ellipse, we assume a value of about  $r_{\text{peri}} \approx 6,200$  km from Earth (or about 170 km below the surface of Earth). Semi-major axis  $a$  is thus:

$$a = \frac{6,200 + 376,000}{2} \approx 191,100 \text{ km} \quad (24)$$

Using vis-viva at the entry interface:

$$\begin{aligned} v_{\text{EI}}^2 &= 3.986 \times 10^5 \times \left( \frac{2}{6,492.9} - \frac{1}{191,100} \right) \\ &= 3.986 \times 10^5 \times (3.081 \times 10^{-4} - 5.23 \times 10^{-6}) \\ &= 3.986 \times 10^5 \times 3.029 \times 10^{-4} \\ &\approx 120.7 \text{ km}^2/\text{s}^2 \end{aligned} \quad (25)$$

$$v_{\text{EI}} \approx 10.99 \text{ km/s} \approx 11.0 \text{ km/s} \quad (26)$$

In miles per hour:  $11.0 \text{ km/s} \times 3600/1.609 \approx 24,600 \text{ mph}$ . This is within about 0.3% of the reported entry velocity of 24,664 mph (39,693 km/h, or 11.03 km/s) during the Artemis II reentry on April 10, 2026. The small remaining gap reflects our rounded, assumed values for the apogee and perigee as well as the perturbations discussed in Section 2.

### 3.2 How 25,000 mph Compares with Earlier Missions

To understand why it is so critical that Orion enter the atmosphere at 25,000 mph (11 km/s), one needs only compare it to the entry speeds of previous missions. Returning astronauts from the ISS reenter at approximately 7.8 km/s, their orbital velocity. Re-entry speeds for the Apollo capsules were around 10.8–11.1 km/s; Apollo 10’s 24,791 mph (about 11.08 km/s) is still the fastest any humans have traveled. The Space Shuttle was capable of reaching speeds of 7.8 km/s upon re-entry, while the Artemis I capsule entered the atmosphere at about 11 km/s (NASA quoted roughly 25,000 mph).

The difference between the re-entry speeds from Apollo missions and those of Artemis II (as well as Artemis I) is caused by the difference in orbital energy of the spacecraft. Apollo missions used a free-return or powered trajectory from the Moon, arriving at Earth with comparable energies to Artemis II, the Apollo program also re-entered at around 10.8–11.0 km/s. What makes Artemis II historically notable is not the speed relative to Apollo, but the fact that it is the fastest *crewed* reentry since Apollo 17 in 1972, and the first time humans have faced this energy level with a capsule that had a known heat shield anomaly.

The specific orbital energy of a spacecraft in low Earth orbit is roughly  $-29.3$  MJ/kg. A spacecraft returning from the Moon has a specific orbital energy much closer to zero (about  $-1$  MJ/kg for the ellipse used above, since  $\varepsilon = -\mu_E/2a$ ). Orion never escaped Earth’s gravity; it climbed out to lunar distance on a very long ellipse and fell back into Earth’s atmosphere. This means that it arrived with significantly more kinetic energy than any spacecraft orbiting the Earth.

The specific kinetic energy at the entry interface is:

$$\frac{KE}{m} = \frac{v_{EI}^2}{2} = \frac{(11.0 \times 10^3)^2}{2} = 60.5 \times 10^6 \text{ J/kg} = 60.5 \text{ MJ/kg} \quad (27)$$

At LEO reentry:  $KE/m = (7.8 \times 10^3)^2/2 \approx 30.4$  MJ/kg. Orion enters the atmosphere with roughly twice the energy per unit mass as the ISS crew. All of this energy must somehow be shed; most of it will heat the shock layer and the surrounding air, while only a small fraction will reach the heat shield (Section 8 puts it well under 1%). Sections 4 and 5 of this paper focus on explaining these phenomena.

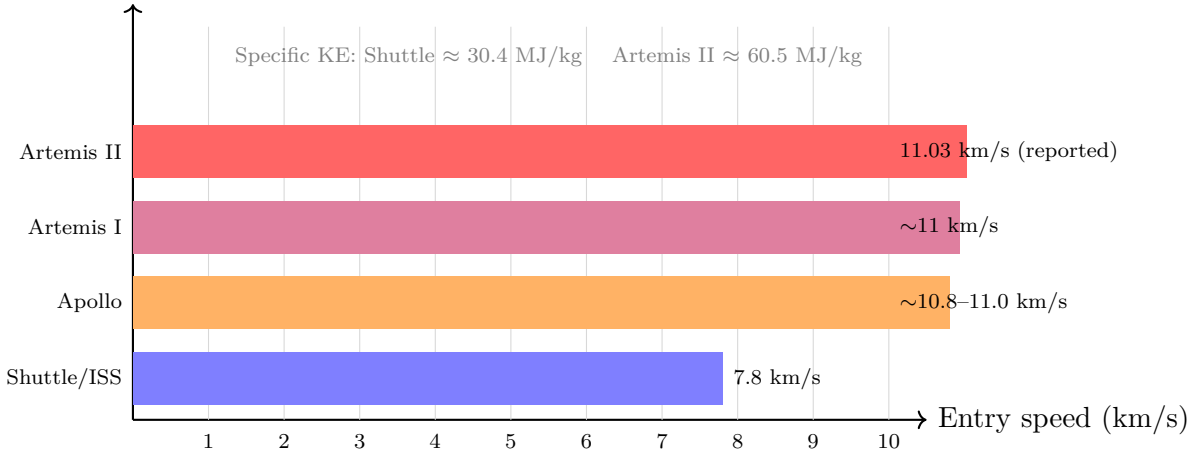


Figure 5: Comparison of atmospheric entry speeds across crewed missions. Orion’s reported entry speed of 11.03 km/s (24,664 mph) on Artemis II is the fastest crewed reentry since Apollo 17 in 1972, carrying approximately twice the specific kinetic energy of a capsule returning from the ISS.

### 3.3 The Hyperbolic Excess Velocity and Gravity Wells of Earth

There is actually a simpler way of looking at the entry speed, as it can be directly related to the trans-Earth orbital mechanics discussed in Section 1. Once Orion exits the Moon’s sphere of influence, it enters the Earth’s gravitational influence zone with a certain velocity relative to Earth. As it approaches Earth and plunges into its gravity well, it loses gravitational potential energy, gaining kinetic energy instead. The relationship between entry speed and hyperbolic excess velocity  $v_\infty$  is:

$$v_{\text{EI}}^2 = v_{\infty, E}^2 + \frac{2\mu_E}{r_{\text{EI}}} \quad (28)$$

This is, once again, the energy conservation equation used in Section 1 to find the post-lunar-flyby velocities. We might expect the spacecraft to be on a slightly hyperbolic orbit relative to Earth ( $\varepsilon \approx 0$ ), so let's check. Rearranging, we get:

$$\begin{aligned} v_{\infty, E}^2 &= v_{\text{EI}}^2 - \frac{2\mu_E}{r_{\text{EI}}} \\ &= 120.7 - \frac{2 \times 3.986 \times 10^5}{6,492.9} \\ &= 120.7 - 122.8 \approx -2.1 \text{ km}^2/\text{s}^2 \end{aligned} \quad (29)$$

Thus, the hyperbolic excess velocity of Orion with respect to Earth is imaginary, meaning that its orbit with respect to Earth is actually elliptical and not hyperbolic. The entry speed of 11 km/s is entirely a product of descending through the Earth's gravity well. Orion's trajectory around Earth is not an open hyperbola but a closed ellipse, whose (virtual) perigee lies below the entry interface.

### 3.4 Mach Number at Entry Interface

Finally, a couple of more calculations worth doing at this point that I believe fit into this section: the Mach number at the entry interface. Since the Mach number depends on the ambient temperature and density of the atmosphere, we need to determine both of these quantities at the entry interface. At the entry interface, Orion is in the thermosphere, where the standard atmosphere gives a temperature of approximately 360 K.

Assuming air to be a diatomic and ideal gas, the speed of sound in air at 360 K is:

$$c = \sqrt{\gamma_{\text{air}} \cdot R_{\text{specific}} \cdot T} = \sqrt{1.4 \times 287 \times 360} = \sqrt{144,648} \approx 380 \text{ m/s} \quad (30)$$

Therefore, the Mach number at the entry interface is:

$$M = \frac{v_{\text{EI}}}{c} = \frac{11,000}{380} \approx 28.9 \approx \text{Mach } 29 \quad (31)$$

Published figures for the actual reentry are higher: NASA's entry-day updates described Orion as traveling nearly 35 times the speed of sound, and PBS reported Mach 33. The difference is mostly a matter of which speed of sound you divide by. Lower in the atmosphere, where it is colder, the speed of sound is lower and the same speed gives a higher Mach number. At this point, the spacecraft enters extreme hypersonic flight, and the ionized plasma surrounding the capsule causes a communications blackout lasting approximately 6 minutes, a planned and expected consequence of the physics we will examine in Section 4.



Figure 6: Artist's concept of Orion during atmospheric reentry, surrounded by ionized plasma generated by the hypersonic shockwave ahead of the heat shield. Published figures for the heat shield surface range from about 3,000°F to nearly 5,000°F (2,760°C); the shock layer itself is far hotter. *Credit: NASA.*

## Key Takeaways

At an altitude of 121.9 kilometers, Orion entered the entry interface, the boundary past which orbital mechanics give way to aerodynamics. Any higher than that, and you're in space. Any lower, and the air begins to push back.

Orion crossed this boundary traveling at 24,664 miles per hour, more than ten times faster than a rifle bullet. And using nothing more than the vis-viva equation and a few rounded orbital values, we can reconstruct its velocity at the entry interface to within about 0.3 percent, an error of roughly 70 miles per hour at nearly 25,000 miles per hour, which may well be the most remarkable result to come out of this essay.

But why so high? Because Orion was falling from a quarter of a million miles away. It was sent out there by the TLI burn, turned around by the Moon's gravity, and pulled back down by Earth's. Four days of acceleration got Orion to the top of the atmosphere; every crewed mission since Apollo, from the Space Shuttle program to ISS missions, falls back to Earth from no more than several hundred miles up, while Orion fell from 252,756 miles, the source of almost double the energy per kilogram.

That's what made Orion so fast. At that speed, even air can't get out of the way fast enough. By our estimate, Orion met the upper atmosphere at about Mach 29, and published figures for the reentry put it at Mach 33 to 35.

## 4 Aerothermodynamics of Mach 33 Reentry

We left off at Mach 29 at the entry interface, by our estimate, with reported figures of Mach 33 to 35. At this stage, the air cannot even respond ahead of the craft. It gets compressed to a near-discontinuity into a thin film of super-heated plasma, right on top of the heat shield. This is referred to as the shock layer, and understanding it is crucial to explaining why reentry is the most thermally stressful part of human spaceflight.

As a spacecraft flies through the air at hypersonic speeds, our conventional ideas about aerodynamics break down in ways we have never encountered before. In subsonic flow, a pressure disturbance travels ahead of the body, and there is time for the fluid to flow out of the way before the spacecraft gets there. But Mach 33 is around 33 times the speed of sound in air (crazy, right?). The result is a normal shock wave, a near-discontinuity in pressure, density, and temperature, a short distance in front of the heat shield. Between the shock wave and the capsule surface is the shock layer.

### 4.1 The Rankine-Hugoniot Equations for Normal Shock Waves

The shock jump relations for normal shock waves can be derived from the conservation of mass, momentum, and energy. Given a calorically perfect gas with ratio of specific heats  $\gamma$ , the downstream shock quantities in terms of Mach number  $M$  are:

Pressure ratio:

$$\frac{p_2}{p_1} = \frac{2\gamma M^2 - (\gamma - 1)}{\gamma + 1} \quad (32)$$

Density ratio:

$$\frac{\rho_2}{\rho_1} = \frac{(\gamma + 1)M^2}{(\gamma - 1)M^2 + 2} \quad (33)$$

Temperature ratio:

$$\frac{T_2}{T_1} = \frac{[2\gamma M^2 - (\gamma - 1)][(\gamma - 1)M^2 + 2]}{(\gamma + 1)^2 M^2} \quad (34)$$

Substituting  $M = 33$  and  $\gamma = 1.4$ :

$$\begin{aligned} \frac{p_2}{p_1} &= \frac{2 \times 1.4 \times 33^2 - 0.4}{2.4} = \frac{3,049.2 - 0.4}{2.4} \approx 1,270 \\ \frac{\rho_2}{\rho_1} &= \frac{2.4 \times 1,089}{0.4 \times 1,089 + 2} = \frac{2,613.6}{437.6} \approx 5.97 \approx 6 \\ \frac{T_2}{T_1} &= \frac{1,270}{5.97} \approx 213 \end{aligned} \quad (35)$$

At this temperature, the ideal gas assumption breaks down entirely. The shock layer temperatures are so high that the diatomic molecules of air begin to dissociate into individual

nitrogen and oxygen atoms, which then ionize to form plasma. The chemistry involved removes energy from the hot gas as internal heating, effectively lowering the gas’s  $\gamma$  to values as low as 1.1 or 1.2. Real-gas effects like these, which complicate thermodynamics, are among the toughest parts of computational hypersonics.

With real gas effects included, the true temperature of the shock layer for a Mach 33 entry is somewhere around 8,000 to 10,000 K. Note that this is substantially lower than the “perfect gas” value of about 60,000 K (taking  $T_1 \approx 280$  K, the ambient temperature at which 11 km/s is Mach 33), as the gas takes energy to break apart and ionize rather than to heat up.

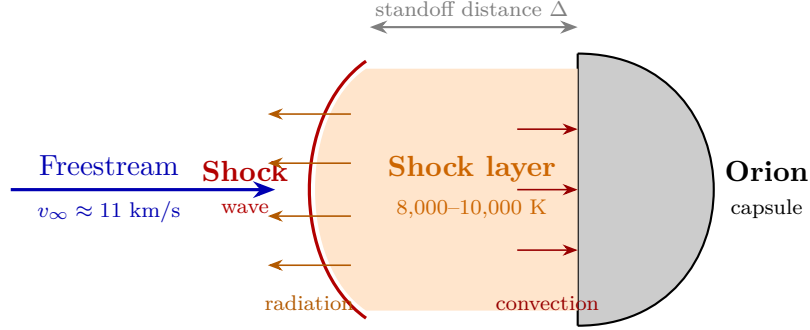


Figure 7: Schematic of the hypersonic shock layer during Orion’s reentry (not to scale). The freestream flow at  $\sim 11$  km/s (Mach 33 at peak) is abruptly compressed across the normal shock wave into the shock layer, where temperatures reach 8,000–10,000 K. Heat reaches the capsule surface by both convection (hot gas contact) and radiation (electromagnetic emission from the shock layer). For a blunt body, most of the energy deposited in the shock layer is swept around the capsule into the wake rather than reaching the heat shield.

## 4.2 Convective Heat Transfer and Radiative Heat Flux

There are two sources of heating for the vehicle: convective heating from physical contact with hot molecules in the shock layer, and radiative heating from ultraviolet and visible emission by the shock-layer gas itself. For low-Mach hypersonic flows (for instance, less than Mach 15), the convective source dominates. At lunar reentry Mach numbers, radiative heating becomes significant, a sizable fraction of the total rather than the rounding error it is for entries from low Earth orbit.

The convective heat flux at the stagnation point, the most severely heated portion of the capsule, is described by Fay-Riddell theory. A widely used simplified form for a blunt body is the Sutton-Graves relation:

$$q_{\text{conv}} = k \cdot \sqrt{\rho_{\infty}/R_n} \cdot v_{\infty}^3 \quad (36)$$

$k$  is a constant that depends on geometry and the gas type. For the sake of simplicity, assume  $k = 1$  for the case of Orion. Here  $\rho_{\infty}$  is the freestream density and  $v_{\infty}$  is the freestream velocity. The third-power dependence on velocity explains the greater severity of the lunar

return scenario, and doubling the velocity increases the convective heat transfer by a factor of 8. The radiative heat flux from the shock layer, in turn, is governed by:

$$q_{\text{rad}} = C \cdot \rho_{\infty}^{1.5} \cdot R_n \cdot f(v_{\infty}) \quad (37)$$

Here,  $R_n$  is the nose radius of the vehicle. And again assuming  $C = 1$  for simplicity,  $f(v_{\infty})$  is a strongly nonlinear function of velocity, scaling as roughly  $v^{12}$  in the regime of Artemis II. Note this nonlinearity: for the Shuttle, radiative heating plays no significant role; for Artemis I and Artemis II, which entered at nearly the same speed, it is a significant share of the stagnation-point heating.

Thus, for the Orion spacecraft, whose heat shield is a shallow spherical section with a radius of curvature of roughly 6 meters (far blunter than a sharp point nose), during Artemis II peak reentry heating the stagnation point sees somewhere in the neighborhood of 200 to 250 W/cm<sup>2</sup> of heat flux (a rough estimate; NASA has not published the measured value), compared to roughly 50 W/cm<sup>2</sup> at peak re-entry heating for the Shuttle. Artemis I entered at almost the same speed, so its peak heating was of the same order.

### 4.3 Why a Blunt Capsule Geometry?

Given all this, I caught myself asking why the Orion vehicle is such a blunt capsule. After all, sharp-nosed vehicles intuitively reduce drag and are more fuel-efficient for subsonic flight. But at high Mach numbers, drag is irrelevant; heat is the limiting factor. Blunt vehicles are better at managing heat than sharp vehicles.

For a blunt capsule, the shock wave stands farther from the surface of the craft. Most of the energy the shock deposits in the air is then carried around the capsule and left behind in the wake instead of being conducted into the surface, and the convective heating at the stagnation point falls as the nose radius grows (as  $1/\sqrt{R_n}$  in the Sutton-Graves form). The trade-off is that a thicker shock layer radiates more, which is why radiative heating grows with  $R_n$  in Equation 37. Furthermore, since blunt bodies experience large amounts of drag, they decelerate quickly and experience the worst heating for a shorter amount of time.

The drag force on the Orion capsule is:

$$F_D = \frac{1}{2} \rho v^2 C_D A \quad (38)$$

where  $\rho$  is the atmospheric density,  $v$  is the velocity of the craft,  $C_D$  is the drag coefficient (around 1.3 for Orion at hypersonic speeds), and  $A$  is the reference area (roughly  $\pi \times (16.5/2 \times 0.3048)^2 \approx 19.9$  m<sup>2</sup> given the 16.5 ft capsule diameter). The deceleration felt by the crew members is:

$$a = \frac{F_D}{m} = \frac{\rho v^2 C_D A}{2m} \quad (39)$$

in units of standard gravity ( $9.81 \text{ m/s}^2$ ). This is the g-force on the crew of the capsule. The Apollo lunar returns saw peak deceleration forces of up to about 7g; NASA’s planned entry profile for Artemis II called for up to 3.9g, well within human tolerance.

#### 4.4 Artemis II Reentry Profile: A Shorter Skip

Artemis I flew a skip guidance trajectory. Orion entered the atmosphere, shed part of its speed while remaining hypersonic, used its lift to climb back out to the thin upper atmosphere, and then descended again for the final entry. During the skip phase, the heating on the capsule dropped drastically. This led to the heat shield problem on Artemis I: the material underneath was still hot and still producing pyrolysis gases, but the outer char had become much less permeable, so the gas could not escape. Pressure built up inside the AVCOAT, and when heating resumed on the second descent, pieces of char cracked and broke away.

To avoid this, Artemis II flew a modified skip entry: an initial dip into the upper atmosphere followed by a much shorter climb back out. Keeping the heat shield under nearly continuous heating keeps the char permeable, so pyrolysis gases can escape gradually through it instead of building up inside the material. This way, the gas never gets a pause in which to build up pressure and crack the char, while still providing some of the range and precision benefits of a guided entry. The integrated heat load, where  $q$  is the total heat flux per unit area at the capsule surface ( $\text{W/cm}^2$ ), in units of  $\text{Ws/cm}^2$ , is:

$$Q = \int q(t) dt \tag{40}$$

Note the integral starts at the entry interface and finishes at parachute deployment. With the Artemis I skip guidance scheme, the curve has two peaks separated by a long cooling period; with Artemis II’s much shorter skip, the pause nearly disappears and the curve behaves like one broad peak. NASA engineers described the result as higher sustained heating during the descent, which is exactly what keeps the char permeable. The heat shield should have been designed for both.

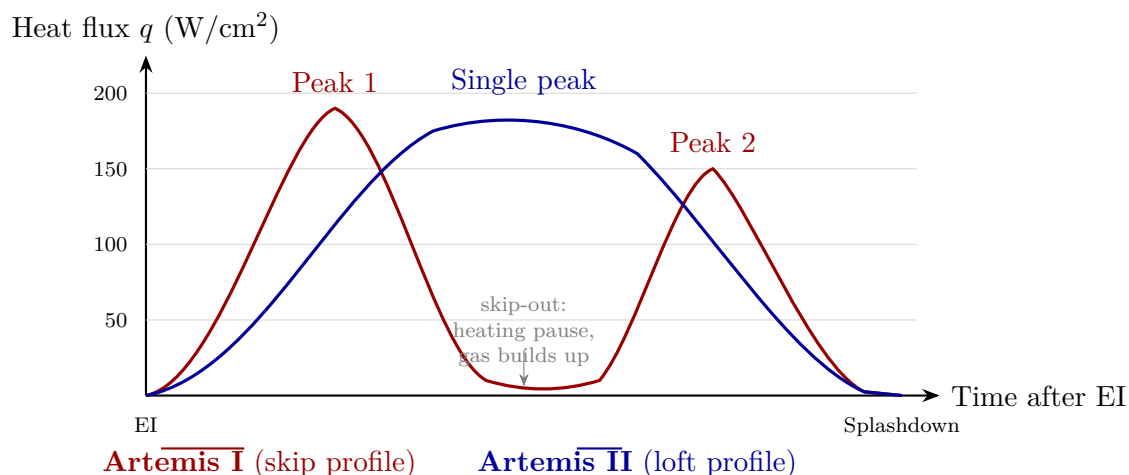


Figure 8: Schematic heat flux profiles for Artemis I (skip guidance, red) and Artemis II (loft profile, blue). The Artemis I profile has two peaks separated by a low-heating dwell phase during which gas built up inside the AVCOAT ablator and cracked the char. Artemis II's much shorter skip keeps the heating nearly continuous (drawn here as a single broad peak), so the gas never gets a pause in which to build up.



Figure 9: An Orion heat shield being lifted during ground processing. After Artemis I, post-flight inspection found unexpected loss of charred AVCOAT at more than 100 locations: the char, which is designed to erode smoothly, instead cracked and broke away after gas built up inside it during the skip entry. This anomaly directly drove the decision to shorten the skip for Artemis II. *Credit: NASA.*

## 4.5 Ballistic Coefficient and Its Impact on Trajectory

The last term worth defining is the ballistic coefficient, a term that characterizes how a body responds to aerodynamic forces:

$$\beta = \frac{m}{C_D \cdot A} \quad (41)$$

For Orion’s capsule geometry:

$$\beta = \frac{9,300}{1.3 \times 19.9} \approx 360 \text{ kg/m}^2 \quad (42)$$

The higher the ballistic coefficient, the more difficult it is to decelerate the capsule via drag. Apollo-style spacecraft had ballistic coefficients of this order of magnitude. In contrast, the ballistic coefficient of a parachute is near zero.

Combined with entry angle and speed, the ballistic coefficient determines the trajectory: the deceleration profile, when peak heating happens, how long it lasts, at what altitude subsonic flow is reached, and where parachutes can be deployed. All of these parameters are covered in the next section of the guide. But first, let’s dive deeper into the heat shield’s design and materials.

### Key Takeaways

At Mach 33, air doesn’t behave like air anymore. It can’t move out of the way fast enough, so it piles up in front of the heat shield and gets compressed so violently that it turns into plasma (the same stuff the sun is made of). The temperature in that thin layer of plasma ranges from 8,000 to 10,000 kelvin. For reference, the surface of the sun is about 5,800 K. Two things heat the spacecraft: convection, where hot gas physically touches the surface, and radiation, where the plasma glows so hot it beams energy directly onto the shield like a furnace. At Mach 33, radiation becomes a big part of the heating, something that doesn’t happen at lower reentry speeds.

Here’s the counterintuitive part: a blunt capsule is actually better than a pointed one at surviving this. A blunt shape pushes the shockwave farther out from the surface, so most of the heat stays in the air that flows around the capsule and trails off behind it instead of soaking into the heat shield. A sharp nose pulls the shock right up against the vehicle... and fries it.

The planned peak G-load on the crew was 3.9g, almost four times their body weight, pressing them into their seats for several minutes. Uncomfortable, but well within what a healthy person can survive.

## 5 Heat Shield Ablation Thermochemistry

As mentioned in Section 4, the process occurring in front of the heat shield has been considered; in this section, we discuss what happens inside it. The sacrificial ablating AVCOAT coating at the bottom of the Orion crew module is an active TPS, as the material reacts with the environment and consumes energy. The process involves certain materials chemistry phenomena that will be discussed in this section.

### 5.1 Basic Principle of Ablative TPS

Unlike heat-reflection or absorption (and thus storage) mechanisms, ablative TPS actively absorbs incoming heat energy through endothermic chemical reactions. Then, the absorbed energy is conducted to greater depths, where it is dissipated by further reactions and converted into the heat of the outgoing gas, which exits the surface by convective transport. Once the material is heated and continues to react, producing gases, the cycle continues until complete decomposition of the material takes place. Therefore, the process is self-sustaining and self-regulating with respect to energy absorption.

### 5.2 Materials and Process in Ablative TPS

The AVCOAT ablative material is composed of epoxy novolac resin reinforced with silica fibers and microballoons and filled into a fiberglass honeycomb. But the primary component responsible for the spacecraft's thermal protection is the organic resin binder. Once the material starts decomposing at temperatures above 300–500°C, epoxy novolac resin pyrolyzes and produces gases (CO, CO<sub>2</sub>, H<sub>2</sub>O, H<sub>2</sub>, hydrocarbons), as well as char mostly consisting of carbon atoms. The char layer isolates the crew module's structural material from the heat source.

The following equation describes the pyrolysis reaction schematically:



### 5.3 Energy Balance at the Ablating Surface

Three processes define the energy balance at the ablating surface: the incoming heat flux, surface radiation losses, and the heat required to conduct the energy into depth and sustain the pyrolysis reaction and convection:

$$q_{\text{net}} = q_{\text{in}} - q_{\text{rad,surface}} - q_{\text{ablation}} \quad (44)$$

where  $q_{\text{ablation}}$  is the energy absorbed by the pyrolysis reaction, gas formation, and gas convection.

## 5.4 Radiation Losses

The heat lost by surface radiation depends on its temperature:

$$q_{\text{rad,surface}} = \varepsilon \cdot \sigma \cdot T_w^4 \quad (45)$$

where  $\varepsilon = 0.85$  is the AVCOAT char emissivity (a measure of how efficiently the surface radiates energy relative to a perfect blackbody; note that emissivity and reflectivity sum to 1 for an opaque surface, so  $\varepsilon = 0.85$  means the char is a strong radiator),  $\sigma = 5.67 \times 10^{-8} \text{ W}/(\text{m}^2 \cdot \text{K}^4)$  is the Stefan-Boltzmann constant, and  $T_w$  is the surface material temperature. With the expected peak value of 3,000 K reached during the Artemis II reentry mission, this loss term equates to:

$$\begin{aligned} q_{\text{rad,surface}} &= 0.85 \times 5.67 \times 10^{-8} \times (3,000)^4 \\ &= 0.85 \times 5.67 \times 10^{-8} \times 8.1 \times 10^{13} \\ &\approx 3.9 \times 10^6 \text{ W}/\text{m}^2 = 390 \text{ W}/\text{cm}^2 \end{aligned} \quad (46)$$

Unexpectedly, the radiation losses exceed the incoming heat flux of 200–250 W/cm<sup>2</sup> estimated in Section 4. A surface cannot keep radiating more than it receives, so this mismatch tells us our inputs are rough: a surface in radiative balance at 225 W/cm<sup>2</sup> would sit near 2,600 K, and exothermic oxidation of the carbon char adds some heat at the surface as well. Either way, surface re-radiation rejects a large share of the incoming heat, and only a small part of the incoming energy reaches the crew module wall.

## 5.5 Pyrolysis Kinetics

Following the Arrhenius kinetics law, the rate of decomposition reaction can be described with the following formula:

$$\frac{d\rho}{dt} = -A \cdot \rho^n \cdot \exp\left(\frac{-E_a}{RT}\right) \quad (47)$$

where  $\rho$  is the local density of the epoxy novolac resin, decreasing as time goes,  $A$  is the pre-exponential frequency factor (proportionality constant),  $n \approx 3$  is the reaction order,  $E_a \approx 1.4 \times 10^5 \text{ J}/\text{mol}$  is the activation energy of primary reaction,  $R = 8.314 \text{ J}/(\text{mol} \cdot \text{K})$  is the universal gas constant, and  $T$  is the local material temperature. So, the rate of decomposition grows exponentially with rising temperature. The mass flow rate of outgoing pyrolysis gases is:

$$\dot{m}_g = \int \left( -\frac{d\rho}{dt} \right) dx \quad (48)$$

where the integral covers the entire pyrolysis zone. There are two important aspects to mention regarding the gases released during pyrolysis. First, their flow absorbs heat as they exit towards the surface. Second, they form a protective layer of gases, reducing incoming heat through the blocking effect.

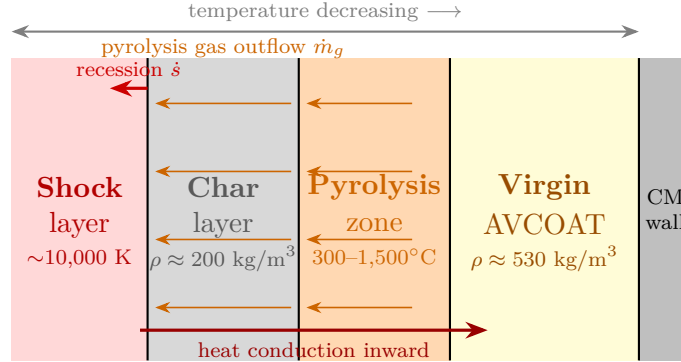


Figure 10: Cross-section of the AVCOAT ablative heat shield during peak reentry heating (not to scale, heat flux arrives from the left). The shock layer plasma heats the char surface to  $\sim 3,000$  K. Pyrolysis gases produced in the decomposition zone flow outward through the porous char, absorbing heat and providing a blocking effect. The virgin AVCOAT and crew module wall stay far cooler than the surface throughout reentry.

## 5.6 The Problem Identified in the Artemis I Mission

The Artemis I problem comes down to exactly this gas flow. The char layer left after the pyrolysis reaction should be porous enough to let gas escape easily. NASA’s post-flight investigation found that the AVCOAT on Artemis I did not let enough of these gases escape. As the spacecraft made its skip entry, Orion entered the atmosphere and started heating and ablating. When it climbed back out, the heating eased and the outer char became even less permeable, while the hot material beneath kept producing gas. Since the gases could no longer escape easily, pressure built up inside the material and, once heating resumed, caused spalling, namely internal cracking and fragmentation of the char layer.

For Artemis II, the fix described in Section 4 was a modified entry with a much shorter skip, which keeps the heat shield under nearly continuous heating so the char stays permeable and the gases keep venting, while still providing some of the range and precision benefits of a guided entry.



Figure 11: Technicians working on an Orion crew module heat shield during spacecraft assembly. The Artemis II heat shield consists of 186 pre-machined AVCOAT blocks bonded onto a carbon fiber substrate, covering the full 16.5-foot diameter base of the capsule. *Credit: NASA/Lockheed Martin.*

## 5.7 Thickness, Mass Loss Rate, and Thermal Margin

The AVCOAT coating is 1 to 3 inches thick, with maximum thickness at the stagnation point. Its total mass on the heat shield for the Artemis II mission is estimated at 700 to 800 kg, spread across 186 blocks covering the 16.5-foot-diameter bottom surface. The recession rate of the ablating surface is given by:

$$\dot{s} = \frac{\dot{m}_w}{\rho_{\text{char}}} \quad (49)$$

where  $\dot{m}_w$  is the mass flow rate of outgoing material (char and pyrolysis gas) and  $\rho_{\text{char}} \approx 200\text{--}300 \text{ kg/m}^3$  is the density of char (significantly less than the density of initial material,  $\rho \approx 530 \text{ kg/m}^3$ ). If the stagnation-point recession rate were held at a peak value of 0.1–0.3 mm/s (a rough estimate) for all 400 seconds of heating, the recession would be 40 to 120 mm, as much as or more than the whole 1-to-3-inch (25 to 76 mm) coating. That cannot be the real picture: the rate is near its peak for only a short part of the heating pulse, so the actual recession is a fraction of this bound, which is what leaves a margin of unreacted material.

## 5.8 Ablation Margin

Ablative margin is the thickness of the unreacted AVCOAT coating left at the end of the heat stage of reentry. It is a crucial parameter in assessing the safety of the Artemis II mission. According to NASA engineers, even with a reduced thermal margin due to the problem identified in the Artemis I mission, the carbon-fiber structure of the crew module would withstand reentry. With the revised Artemis II trajectory, this margin is to be much greater.

Early post-splashdown looks at the Artemis II heat shield were encouraging. Following recovery of the capsule, images showing apparent discoloration on the underside of Orion circulated widely, prompting concern. NASA Administrator Jared Isaacman clarified that the discoloration was not liberated material, the white coloring corresponded to the compression pad area and was consistent with local geometry, AVCOAT byproducts, and transitional heating environments that had been observed and predicted during arc jet ground testing. NASA's entry-day timeline put the expected heat shield temperature at about 3,000°F (1,650°C), while other reports cite nearly 5,000°F. Either way, the heat shield protected the crew and crew module throughout descent. Full data analysis is ongoing and will inform future Artemis heat shield design.

### Key Takeaways

The heat shield doesn't actually have to reflect or absorb heat. Its task is self-destruction, in which the AVCOAT material undergoes a controlled decomposition (it chars rather than burns), absorbing heat from the surface and releasing it into the airflow. It is not possible to reuse this heat protection system.

In more detail, at temperatures above 300°C, the AVCOAT material begins to decompose, forming gases and charring. They exit through porous char and play two roles at once. On the one hand, they absorb part of the heat energy; on the other, they create a protective gas screen that absorbs some of the incoming heat radiation. In addition, the insulating char layer protects the capsule's underlying carbon fiber structure from high temperatures throughout the descent.

However, in the Artemis I mission, it turned out that the heat shield's char was not porous enough, so during the skip entry, when the heating paused, trapped gas built up and cracked the char, and material broke away in over 100 spots on the shield. Artemis II addressed this with a much shorter skip that kept the heating nearly continuous, so the gas never had a chance to build up.

Early post-flight looks found nothing alarming: the discoloration seen in photographs matches behavior NASA had already observed in arc jet tests on Earth, and a full inspection is still underway.

## 6 Drogue Chutes, Mains, and Terminal Descent

When the parachutes deploy, the difficult part is technically over. Orion has gone through its most thermally and structurally challenging minute of space travel, lost almost all of its entry kinetic energy due to atmospheric drag and heat-shield ablation, and slowed from Mach 33 to subsonic speeds. All that remains is slowing the capsule from about 330 mph at chute deployment to a splashdown velocity of about 17–20 mph, which is still no small feat! Imagine slowing down a 9,300 kg capsule using only fabric and air. However, the physics is more straightforward than the hypersonic aerothermodynamics of Section 4, and hence, a very nice topic to end on.

Orion uses a three-stage parachute sequence, starting with three forward-bay cover chutes that pull away the forward-bay cover, exposing the rest of the system. Two drogue chutes then deploy, stabilizing the capsule and initiating deceleration. Finally, three pilot chutes pull out the three main chutes, which bring the spacecraft down to splashdown. The whole process is quick: on April 10, the drogues opened at 8:03 p.m. EDT at 23,400 feet, the mains at 8:04 p.m. at 5,400 feet, and Orion splashed down at 8:07 p.m.

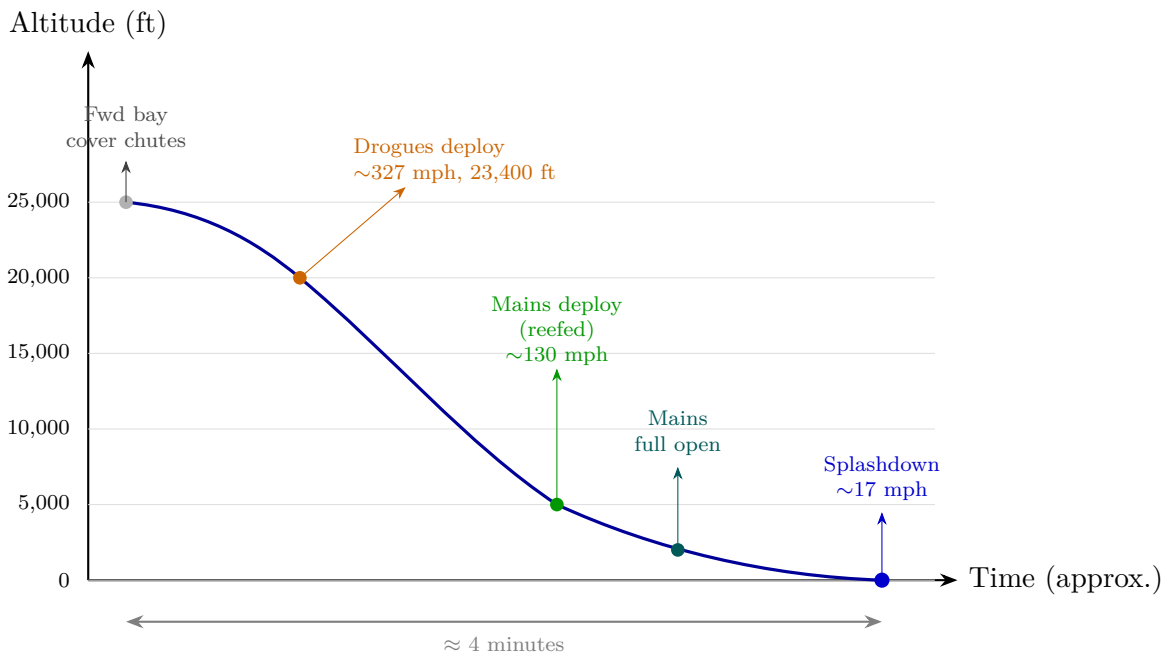


Figure 12: Schematic of the Orion parachute deployment sequence from forward bay cover jettison to splashdown (not to scale). As confirmed during the Artemis II reentry on April 10, 2026, drogue parachutes deployed at 23,400 feet and the three main ringsail parachutes deployed at 5,400 feet, carrying Orion to a splashdown at roughly 17–20 mph (NASA’s quoted range).

## 6.1 The Drag Force Equation at Low Velocity

As for the previous stages of the descent, the basic equations governing the last leg remain unchanged. The force exerted on a parachute in flight is described by the drag force equation derived in Section 4:

$$F_D = \frac{1}{2}\rho v^2 C_D A \quad (50)$$

At terminal velocity, this drag force is perfectly balanced by the gravity force on the capsule:

$$F_D = mg \quad (51)$$

Setting these forces equal to each other and solving for terminal velocity:

$$mg = \frac{1}{2}\rho v_t^2 C_D A \quad (52)$$

$$v_t = \sqrt{\frac{2mg}{\rho C_D A}} \quad (53)$$

And this is the formula for terminal velocity. All parachute engineering revolves around minimizing this quantity, meaning all effort goes into increasing  $C_D \times A$ , the effective drag area.

## 6.2 Drogue Parachutes

The two drogue chutes deployed at 23,400 feet and about 479 feet per second (roughly 327 mph, or Mach 0.47), as NASA reported during the actual Artemis II reentry on April 10, 2026. As mentioned earlier, they are not used to reach terminal velocity for splashdown, since they are too small for that task. Instead, their function is to stabilize the spacecraft, slow it from about 330 mph to about 150 mph, and provide conditions for deploying the main chutes.

The shape of each drogue parachute is a conical ribbon, with an approximate nominal diameter of 23 feet. This design was deliberately selected because it provides greater dynamic stability during deployment, as a solid conical canopy would experience dangerously large loads at such high velocities. At the same time, ribbons allow some airflow through the chute, mitigating them. The maximum loads a parachute experiences occur at the moment of its deployment. The following formula describes the force of such loads:

$$F_{\text{open}} = C_x \cdot q_{\text{dyn}} \cdot A_{\text{ref}} \quad (54)$$

where  $C_x$  is an opening coefficient (taking into account the dynamics of the chute opening),  $q_{\text{dyn}} = \frac{1}{2}\rho v^2$  is the dynamic pressure at the moment of deployment, and  $A_{\text{ref}}$  is the reference

area of the parachute. At Orion drogue-deployment velocity and altitude,  $q_{\text{dyn}}$  is considerable; hence, these forces are critical for parachute design.

For Orion’s drogue parachutes deploying at 23,400 feet (with  $\rho \approx 0.581 \text{ kg/m}^3$  from the standard atmosphere and  $v \approx 146 \text{ m/s}$ ):

$$\begin{aligned} q_{\text{dyn}} &= \frac{1}{2} \times 0.581 \times 146^2 \\ &= \frac{1}{2} \times 0.581 \times 21,316 \\ &\approx 6,192 \text{ Pa} \approx 6.2 \text{ kPa} \end{aligned} \tag{55}$$

That is only about 6 percent of an atmosphere, but spread over a 23-foot canopy (about  $38 \text{ m}^2$ ) it already adds up to roughly 240 kN before the opening coefficient  $C_x$  is applied. The structural design of the drogue must withstand this stress.

### 6.3 Main Parachutes

Having slowed Orion down from about 330 mph to roughly 130 to 150 mph, the drogues release at approximately 5,400 feet altitude, and the three main parachutes deploy in a two-stage procedure, first partially reefed (smaller canopy), then fully open. Reefing ensures lower loads during chute opening. In the first moments, it keeps the canopy diameter small, forcing gradual deployment.

The main chutes are ring-sail designs, with a nominal diameter of approximately 116 feet (almost the size of a large building). Each parachute covers approximately 11,000 square feet ( $1,022 \text{ m}^2$ ) of canopy area, as confirmed by Artemis II parachute system manager Jared Daum. With three main chutes fully deployed, the drag area of the entire assembly becomes:

$$C_D \cdot A_{\text{total}} = 3 \times C_{D,\text{main}} \times \frac{\pi}{4} D^2 \tag{56}$$

The drag coefficient for a ringsail at low velocities is roughly 0.97. Using the confirmed canopy area of  $1,022 \text{ m}^2$  per parachute:

$$C_D \cdot A_{\text{total}} = 3 \times 0.97 \times 1,022 \approx 2,974 \text{ m}^2 \tag{57}$$

And using the formula for terminal velocity:

$$\begin{aligned}
v_t &= \sqrt{\frac{2 \times 9,300 \times 9.81}{1.225 \times 2,974}} \\
&= \sqrt{\frac{182,466}{3,643}} \\
&= \sqrt{50.1} \approx 7.1 \text{ m/s} \approx 25.4 \text{ km/h} \approx 15.8 \text{ mph}
\end{aligned} \tag{58}$$

This is in the same range as NASA’s quoted splashdown speed of about 17–20 mph (we will use 19 mph, or 8.5 m/s, below). The gap most likely comes from our drag area: three clustered canopies interfere with one another and produce less drag than three isolated ones, so the effective drag coefficient is lower than 0.97 (a value between about 0.6 and 0.85 reproduces 17–20 mph). The capsule is not still slowing down at impact; under the mains it settles to terminal velocity within seconds.

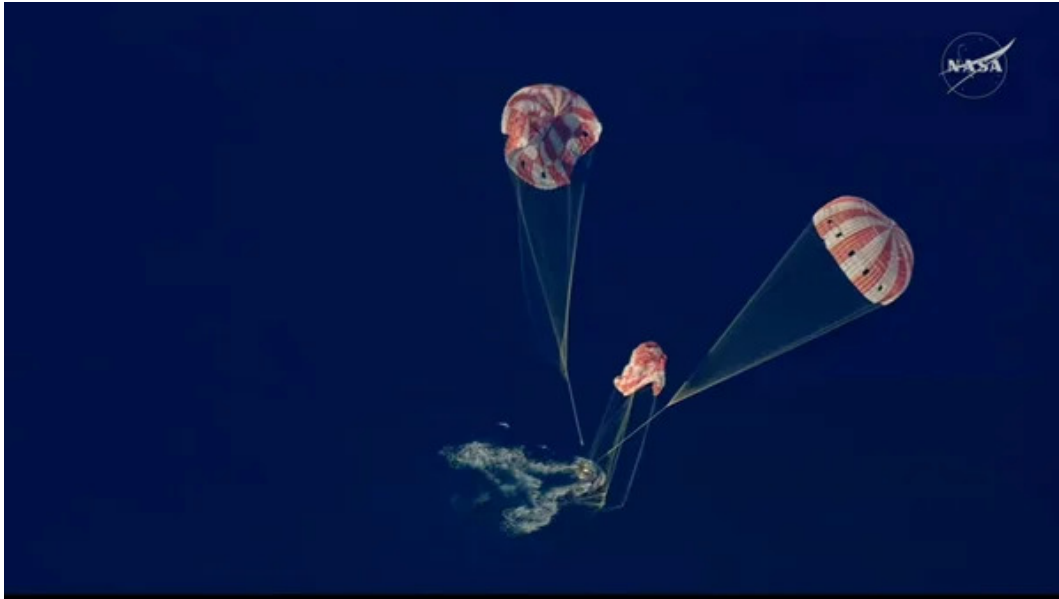


Figure 13: Orion descending under its three main ringsail parachutes during splashdown. Each parachute has a nominal diameter of 116 feet and covers approximately 11,000 square feet (1,022 m<sup>2</sup>) of canopy area; together they reduce the capsule’s descent speed to approximately 17–19 mph at splashdown. *Credit: NASA.*

## 6.4 Impact Forces in Water

The landing at 17 mph is still not soft in the usual sense as hitting the water still creates a significant shock loading for the crew. However, Orion’s design includes special crew seats where they sit in their suits in a reclined position. These seats are engineered to mitigate the impact load. Peak deceleration at impact with water depends on the angle of impact and sea state at the splashdown location.

In the case of a near-vertical water landing at a splashdown speed of approximately 19 mph ( $\approx 8.5$  m/s), the average deceleration can be estimated from the work-energy relation. The average deceleration over stopping distance  $d$  is:

$$a = \frac{v^2}{2d} \quad (59)$$

If one assumes  $d \approx 1$  m (a reasonable value for the initial water depth penetrated before the load stops the capsule):

$$a = \frac{(8.5)^2}{2 \times 1} = \frac{72.25}{2} \approx 36.1 \text{ m/s}^2 \approx 3.7g \quad (60)$$

This is an average over the stopping distance, and it is modest compared with the entry loads; the reclined seats soften it further. The crew were confirmed to be in good health immediately following splashdown, with commander Wiseman radioing “four crew green” to Mission Control.

## Key Takeaways

When the parachutes open, the toughest job is done. In about ten minutes, the spacecraft has slowed from a mind-boggling 25,000 mph to subsonic speeds, meaning it’s flying slower than the speed of sound.

However, decelerating a 9,300 kg payload from about 330 mph to just 17–20 mph using nothing but fabric and air isn’t an easy task either. Orion employs a system of 11 parachutes in three phases: first, small parachutes pull away the forward bay cover; then, two drogues at 23,400 feet reduce the capsule’s speed to about 150 mph. Finally, at 5,400 feet, pilot parachutes pull out three ringsail parachutes, each 116 feet in diameter and with an area of 11,000 sq feet, which slow Orion down to the water.

The science behind it is simple: at terminal velocity, drag force equals gravitational force. The larger the parachutes and the more drag they generate, the slower the descent. Our analysis suggests that, in Orion’s case, terminal velocity should be about 15.8 mph, a bit under NASA’s quoted 17 to 20 mph. The small gap comes from our generous drag estimate: three parachutes flying close together produce a little less drag than three on their own.

Even at 19 mph, a collision with water results in an acceleration of about 3.7g, meaning the crew experiences about 4 times their body weight. To mitigate such forces, Orion is equipped with custom-designed reclining seats.

## 7 Splashdown Targeting and Ocean Recovery

All aspects of the analysis thus far, from the free-return trajectory to the TCMs to the entry corridor to the heat shield to the parachutes, built up for one goal: delivering a 9,300 kg capsule to a particular region in the Pacific Ocean, approximately 40 to 50 miles off the coast of San Diego (confirmed as the actual splashdown location on April 10, 2026). Because of this, I’ve divided this section into three parts: splashdown targeting, the targeting process itself to achieve such accuracy during a capsule’s descent, and the process of recovering the capsule from the ocean.

### 7.1 Defining the Splashdown Target

It should be emphasized that the splashdown zone is defined as the set of outcomes that simultaneously satisfy multiple criteria, for which there may be conflicting solutions. To start, the splashdown zone must be reachable under the constraints discussed in Section 2 (the flight-path angle at the entry interface limits the achievable ground track, and once the parachutes open, the capsule cannot be significantly redirected). In addition, the splashdown zone must lie in the Pacific Ocean (splashdowns in the Atlantic would require flying over populated areas during the heating phase). It must be accessible to the recovery ship in a reasonable travel time from its home base at San Diego, the sea conditions in that zone must be favorable for diver operations, and it must lie outside any shipping lanes. NASA also sets strict meteorological “go” conditions for the splashdown zone: wave heights below 6 feet (1.8 m), winds below 28.7 mph (46 km/h), and no rain or lightning within 30 nautical miles of the recovery area.

As it turns out, for Artemis II, all the above criteria narrow the location to an ellipse approximately 40 to 50 miles southwest of San Diego. This ellipse is the so-called landing dispersion ellipse, and the uncertainties in the entry condition entirely determine its parameters. Mission Control confirmed a “perfect bullseye splashdown” on April 10, 2026, with the capsule landing right on target.

### 7.2 The Landing Dispersion Ellipse

The landing dispersion ellipse indicates that, due to uncertainties in the entry conditions, the actual landing site will differ somewhat from the expected one. There are two major sources of dispersion for the capsule’s position upon splashdown: the uncertainty in the flight path angle  $\gamma$  at entry interface, and the uncertainty in the cross-range position (i.e., the distance along a line perpendicular to the ground track).

Let  $\gamma_0$  and  $v_0$  be the nominal values of flight path angle and the entry speed of the capsule (which can be calculated using Equation 26 from Section 3). Assuming the capsule has a lift-to-drag ratio of  $L/D$  (approximately 0.3 for Orion) at the entry interface, a perturbation in the flight path angle  $\delta\gamma_0$  results in a downrange dispersion  $\delta R$  given approximately by:

$$\delta R \approx \frac{2v_0^2}{g} \cdot \frac{L}{D} \cdot \delta\gamma_0 \cdot \cos(\gamma_0) \quad (61)$$

Thus, for Artemis II with  $v_0 \approx 11,000$  m/s,  $L/D \approx 0.3$ ,  $\gamma_0 \approx -6.5^\circ$ , and a nominal uncertainty in flight path angle  $\delta\gamma_0 \approx 0.1^\circ$ :

$$\begin{aligned}\delta R &\approx \frac{2 \times (11,000)^2}{9.81} \times 0.3 \times \frac{0.1\pi}{180} \times \cos(-6.5^\circ) \\ &\approx 2.467 \times 10^7 \text{ m} \times 0.3 \times 0.001745 \times 0.9936 \\ &\approx 12,900 \text{ m} \approx 8 \text{ miles}\end{aligned}\tag{62}$$

In other words, a 0.1-degree uncertainty in the flight path angle  $\gamma$  results in an 8-mile spread of the capsule’s final location. By tightening the uncertainty  $\delta\gamma$ , the dispersion ellipse shrinks, increasing the probability of landing within the splashdown area.

The cross-range uncertainty can be estimated similarly, though this is complicated by the fact that atmospheric density changes along the trajectory, thereby affecting the lift-to-drag ratio. Both kinds of error can be corrected in flight. At hypersonic speeds, Orion can steer by changing the direction of its lift with a bank angle: the vertical part of the lift stretches or shortens the downrange distance, and the sideways part corrects cross-range. Thus, the guidance system steers the capsule sideways toward the target splashdown point as it descends through the atmosphere.

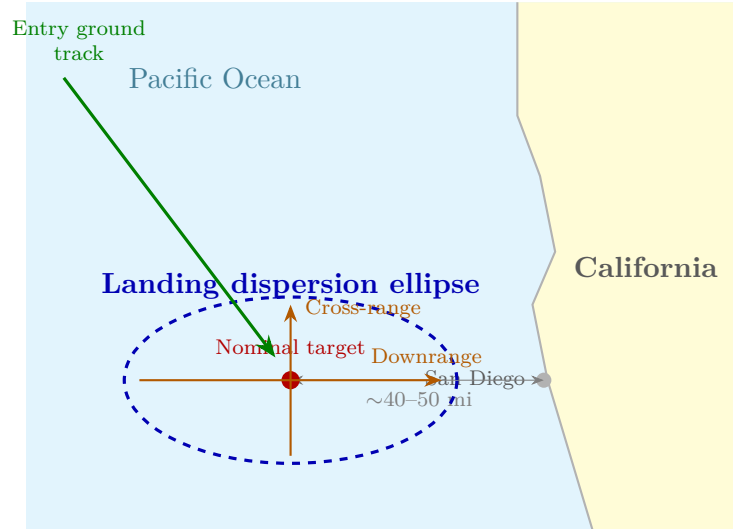


Figure 14: Schematic of the Artemis II splashdown targeting geometry (not to scale). The actual splashdown occurred approximately 40–50 miles southwest of San Diego, confirmed as a “perfect bullseye” by Mission Control. The downrange axis of the ellipse is set by flight path angle uncertainty at entry interface; the cross-range axis is set by lateral velocity uncertainties and atmospheric density variations. TCMs during the trans-Earth coast shrink both axes of the ellipse.

### 7.3 Orion's Entry Guidance

As already mentioned, during atmospheric entry, Orion has some maneuvering capabilities. More specifically, from the entry interface until the parachutes deploy, Orion's guidance software continuously seeks the optimal bank angle that minimizes the miss distance to the target. One possible algorithm is similar to the one used for the Apollo program's final guidance. The basic idea is to compute the predicted splashdown point, compare it to the target, and send commands to the guidance system to adjust the bank angle until the difference falls below a certain threshold. The bank angle controls the direction of the lift: at  $\phi = 0$ , it is maximally upward; at  $\phi = 180^\circ$ , it is maximally downward; and at intermediate values of  $\phi$ , the lift is partially sideways, thus providing the needed steering capability.

Given a bank angle  $\phi$ , the lift and drag forces acting on the capsule are:

$$L = \frac{1}{2}\rho v^2 C_L A_{\text{ref}} \quad (63)$$

$$D = \frac{1}{2}\rho v^2 C_D A_{\text{ref}} \quad (64)$$

where  $C_L$  and  $C_D$  are the respective lift and drag coefficients (for Orion  $C_L \approx 0.4$ ,  $C_D \approx 1.3$ ). As noted above, the lift-to-drag ratio of Orion at hypersonic speeds is  $L/D \approx 0.3$ . The motion of the capsule along the atmospheric trajectory (in the along-track, cross-track, and vertical coordinates) satisfies the following set of ordinary differential equations:

$$\frac{dv}{dt} = -\frac{D}{m} - g \sin \gamma \quad (65)$$

$$\frac{d\gamma}{dt} = \frac{L \cos \phi}{mv} - \left( \frac{g}{v} - \frac{v}{r} \right) \cos \gamma \quad (66)$$

$$\frac{d\psi}{dt} = \frac{L \sin \phi}{mv \cos \gamma} \quad (67)$$

Here,  $v$  is the capsule's speed,  $\gamma$  is its flight path angle,  $\psi$  is the heading angle, and  $r$  is the distance from the capsule to the center of the Earth. Solving the above set of equations is probably what Orion's guidance computer does regularly to steer the spacecraft to its splashdown zone.

### 7.4 Recovery Operations

Once the splashdown site is accurately determined, the USS John P. Murtha sets out for that area, accompanied by other recovery assets (four helicopters from Helicopter Sea Combat Squadron 23, rigid-hull inflatable boats, and Navy divers from Explosive Ordnance Disposal Group 1). The recovery choreography is highly scripted: approximately two hours before splashdown, all surface assets are in position; helicopters are airborne one hour prior. Two of

the four helicopters carry special imaging equipment to track the descending capsule through the final minutes of parachute descent.

The calculation of the splashdown time requires solving the equations of motion with the parachutes deployed, which we have effectively done in Section 6 (see Equation 53). Using the confirmed main chute deployment altitude of 5,400 feet (1,646 m) and a splashdown speed of  $\approx 19$  mph (8.5 m/s):

$$t_{\text{splashdown}} \approx \frac{h}{v_t} = \frac{1,646 \text{ m}}{8.5 \text{ m/s}} \approx 194 \text{ s} \approx 3.2 \text{ min} \quad (68)$$

after the main chute deployment. Adding the roughly one minute from drogue deployment at 23,400 feet to main chute deployment at 5,400 feet gives a total of about 4 to 5 minutes, consistent with NASA’s timeline (drogues at 8:03 p.m., mains at 8:04 p.m., splashdown at 8:07 p.m. EDT).

On April 10, 2026, the USS John P. Murtha was stationed near the predicted splashdown point. Following the confirmed splashdown at 8:07 p.m. EDT, a brief communications issue between recovery teams and the capsule caused a slight delay in approach, but the joint NASA and Navy team stabilized the capsule using a sea anchor and an inflatable collar, installed the “front porch” inflatable raft at the side hatch, and had all four crew members out of the capsule by about 9:34 p.m. EDT, roughly an hour and a half after splashdown.



Figure 15: Navy divers and recovery personnel securing the Orion crew module following splashdown during an Underway Recovery Test (URT). The recovery choreography involves four helicopters, rigid-hull inflatable boats, and open-water Navy divers who enter the water after splashdown. The USS John P. Murtha served as the primary recovery vessel for Artemis II. *Credit: NASA.*

## Key Takeaways

For Orion to return to Earth, it needed not only to endure reentry but also reach a certain point above the ocean where it could be safely recovered. As mentioned earlier, that specific point is somewhere above the waters around San Diego, which makes sense given the accessibility of the Pacific Ocean, the presence of the Navy there, favorable weather, and a proper geometric trajectory.

That target location isn't one particular point. Rather, it is an elliptical area, since any small imperfection in the orientation of the entry will translate to a big mistake in terms of the splashdown. An error of 0.1 degrees in flight path angle can lead to an error of about 8 miles on the surface, which is why the TCMs from Section 2 matter so much.

It should be noted that during reentry, Orion's guidance could make some corrections using the capsule's aerodynamic characteristics. In other words, the capsule was able to roll slightly to adjust the descent trajectory onto the correct course. That led to what was called a perfect bullseye splashdown.

USS John P. Murtha was there, waiting near the splashdown point. Navy divers stabilized the spacecraft and attached an inflatable life raft to the capsule door, allowing four astronauts to be quickly transferred to helicopters and then to the ship.

## 8 The Complete Energy Budget

So far, each segment of this paper has considered one particular aspect of the descent journey in isolation. Section 8 ties all the segments together into one question: where did the energy go? A spacecraft returning from the Moon arrives at the entry interface with an extraordinary amount of kinetic and potential energy. By splashdown, nearly all of it is gone. Where the energy went, and in what proportions, is a useful check on every calculated value from previous sections, since energy must be conserved.

### 8.1 Total Mechanical Energy at Entry Interface

The specific mechanical energy of Orion at the entry interface is equal to the sum of its specific kinetic and specific gravitational potential energy, as measured from Earth's surface:

$$E_{\text{EI}} = \frac{1}{2}v_{\text{EI}}^2 - \frac{\mu_E}{r_{\text{EI}}} + \frac{\mu_E}{R_E} \quad (69)$$

The first part above is the specific kinetic energy. The second part is the gravitational potential energy at  $r_{\text{EI}}$ . The third part adds back the potential energy at Earth's surface, so the energy is measured relative to sea level rather than infinity. Plugging in numbers:

$$KE_{\text{specific}} = \frac{v_{\text{EI}}^2}{2} = \frac{(11,000)^2}{2} = 60.5 \times 10^6 \text{ J/kg} = 60.5 \text{ MJ/kg} \quad (70)$$

$$\begin{aligned} PE_{\text{specific}} &= \mu_E \left( \frac{1}{R_E} - \frac{1}{r_{\text{EI}}} \right) \\ &= 3.986 \times 10^{14} \times \left( \frac{1}{6.371 \times 10^6} - \frac{1}{6.4929 \times 10^6} \right) \\ &= 3.986 \times 10^{14} \times (1.570 \times 10^{-7} - 1.540 \times 10^{-7}) \\ &= 3.986 \times 10^{14} \times 2.96 \times 10^{-9} \approx 1.18 \text{ MJ/kg} \end{aligned} \quad (71)$$

Thus, the total specific mechanical energy at the entry interface is:

$$E_{\text{EI}} \approx 60.5 + 1.18 \approx 61.7 \text{ MJ/kg} \quad (72)$$

For the capsule with a full mass of 9,300 kg, the total energy is:

$$E_{\text{total,EI}} = 61.7 \times 10^6 \times 9,300 \approx 5.74 \times 10^{11} \text{ J} \approx 574 \text{ GJ} \quad (73)$$

To put the above into context, 574 gigajoules is approximately the energy released by 137 tons of TNT, or roughly the electricity about 15 average U.S. households use in a year. All of that energy must somehow be accounted for by the time the capsule floats into the Pacific.

## 8.2 Energy at Splashdown

At the moment of splashdown, Orion is moving at only about 8.5 m/s. Thus, its specific kinetic energy is:

$$KE_{\text{splash}} = \frac{v_{\text{splash}}^2}{2} = \frac{(8.5)^2}{2} = 36.1 \text{ J/kg} \quad (74)$$

And its gravitational potential energy, again as measured relative to sea level, is negligibly small. Therefore, the total mechanical energy at splashdown is:

$$E_{\text{splash}} = 9,300 \times 36.1 \approx 336,000 \text{ J} \approx 0.34 \text{ MJ} \quad (75)$$

The above amount of energy is negligible compared to the 574 GJ at the entry interface. Essentially, all of the entry energy has been spent by splashdown. The question is how?

## 8.3 Where Did the Energy Go?

Three main mechanisms dissipate the energy of entry and account for its dissipation from the entry interface to splashdown.

First and foremost, the energy is dissipated as aerodynamic heating of the shock layer and of the atmosphere through which the spacecraft travels. During entry, the spacecraft decelerates due to drag forces. Virtually all of the work done by drag goes into heating the surrounding air: the shock layer heats up, and that hot gas is swept around the capsule and left behind in the wake, where it cools by mixing and radiating. This heat absorption comes from the shock layer surrounding the spacecraft; Orion's blunt-body shape accounts for the overwhelming fraction of the dissipated energy. The energy dissipated by aerodynamic drag from the entry interface to the end of descent can be expressed as:

$$W_{\text{drag}} = \int F_D \cdot v \, dt = \int \frac{1}{2} \rho(t) v(t)^3 C_D A \, dt \quad (76)$$

where the above integral extends from the entry interface to the moment of deploying parachutes. The integral above cannot be analytically solved due to the coupling between  $\rho(t)$  and  $v(t)$  via the equations of motion. As the craft descends, the air gets denser and drag rises; as it slows, drag falls. The two effects are tied together, which is why the problem needs numerical integration. However, we already know the result. It equals about all of the 574 GJ above.

Secondly, some of the energy is dissipated as heat ablation by the heat shield. The energy absorbed by the AVCOAT heat shield through pyrolysis and re-radiation is calculated in Section 5. The maximum peak heat flux at the stagnation point is estimated at 200–250 W/cm<sup>2</sup>. The energy dissipated by the heat shield equals the energy transferred from the atmosphere to the heat shield through the peak heat flux acting on the stagnation point for about 400 seconds:

$$Q_{\text{shield}} \approx 225 \times 10^4 \frac{\text{W}}{\text{m}^2} \times 0.2 \text{ m}^2 \times 400 \text{ s} = 1.8 \times 10^8 \text{ J} = 180 \text{ MJ} \quad (77)$$

Scaling this up to the entire heat shield (about 19.9 m<sup>2</sup>) at the peak flux for the full 400 seconds would give about 18 GJ, but that is a strict upper bound: the flux away from the stagnation point, and away from the peak moment, is much lower. A few gigajoules, roughly 0.3–0.7% of the total entry energy, is a more realistic figure. Once again, the heat shield is the last resort to prevent the relatively small fraction of shock-layer heat from reaching the capsule.

Thirdly, some energy is dissipated as deceleration through parachute drag. The energy lost to the parachutes can be expressed as:

$$W_{\text{chutes}} = \int F_{D,\text{chutes}} \cdot v \, dt \quad (78)$$

From the drogue deployment at 23,400 feet (7,130 m) to splashdown about four minutes later, the parachutes have to remove the kinetic energy Orion still has at drogue deployment (about 146 m/s) plus all of the potential energy of the drop:

$$W_{\text{chutes}} \approx \frac{1}{2}mv_0^2 + mgh \approx \frac{1}{2} \times 9,300 \times 146^2 + 9,300 \times 9.81 \times 7,130 \approx 0.75 \text{ GJ} \quad (79)$$

The above is only about 0.13% of the total energy dissipated in entry. In absolute terms, parachutes are a negligibly small energy dissipation device relative to the scale of the energy dissipated during entry, and they merely bring the capsule from a few hundred miles per hour down to 8.5 m/s.

## 8.4 The Complete Budget

Combining the parts above:

| Mechanism                        | Energy      | Fraction  |
|----------------------------------|-------------|-----------|
| Total at entry interface         | ~574 GJ     | 100%      |
| Atmospheric drag & shock layer   | ~570 GJ     | ~99.2%    |
| Heat shield ablation & radiation | ~2–4 GJ     | ~0.3–0.7% |
| Parachute drag                   | ~0.75 GJ    | ~0.13%    |
| Residual KE at splashdown        | ~0.00034 GJ | ~0.00006% |

Table 1: Complete energy budget for Artemis II’s return from entry interface to splashdown. The atmosphere accounts for over 99% of all energy dissipation; the heat shield and parachutes together handle less than 1%.

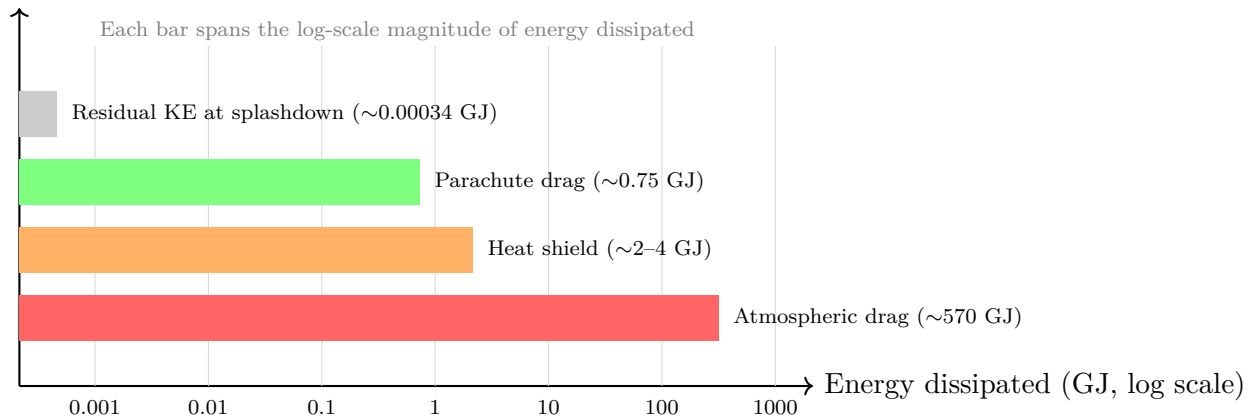


Figure 16: Log-scale energy budget for Artemis II's return. Atmospheric drag and shock layer radiation (red) account for over 99% of all energy dissipation. The heat shield (orange) and parachutes (green) are orders of magnitude smaller contributors. The residual kinetic energy at splashdown (gray) is essentially zero on this scale.

The numbers make it obvious who does the majority of the dissipation work. While the heat shield is one of the most advanced technical components of the thermal protection system, it dissipates less than 1% of the total energy. It is not designed to absorb energy but to withstand it while the atmosphere does the work. Similarly, parachutes are irrelevant for dissipating energy, which occurs mostly in the atmosphere. By splashdown, almost all of the energy gathered on the fall back from a quarter of a million miles away is gone.

This is the energy budget of Artemis II's return. 250,000 miles of orbital mechanics, a hypersonic fireball at Mach 33, 186 blocks of ablative material, three huge ringsail parachutes, and a Navy vessel standing by in the Pacific. All aimed to decelerate four people from 25,000 mph to splashdown in the Pacific Ocean. At 9:58 p.m. EDT on April 10, 2026, just under two hours after splashdown, all four crewmembers were safely aboard the USS John P. Murtha, completing a 694,481-mile journey that began nine days earlier on a Florida launchpad.



Figure 17: Recovery personnel with an MH-60S Seahawk helicopter on a ship's flight deck. After the April 10, 2026 splashdown, Navy divers from Explosive Ordnance Disposal Group 1 and helicopters from Helicopter Sea Combat Squadron 23 extracted all four crewmembers (Reid Wiseman, Victor Glover, Christina Koch, and Jeremy Hansen) and flew them to the USS John P. Murtha, where they were aboard by 9:58 p.m. EDT, completing the 694,481-mile Artemis II mission. *Credit: NASA.*

## Key Takeaways

Now, where does 25,000 mph go? When Orion reentered Earth's atmosphere, it had around 574 gigajoules of potential and kinetic energy combined, or in other words, the equivalent energy output of 137 tons of TNT, or roughly the electricity about 15 average U.S. households use in a year. By the time of splashdown, it was virtually depleted. Where did it go?

Ninety-nine percent of this energy went into the atmosphere. The drag and shockwaves created during re-entry turned almost all of Orion's kinetic energy into heat, most of it left behind in the air along the capsule's path. Less than one percent of it went into the heat shield, which was considered the star of the show in terms of technology used by Orion. Its purpose wasn't to dissipate energy; it merely had to withstand the small percentage that passed through.

A smaller fraction of energy still went into the parachutes. They played a relatively insignificant role in slowing the capsule, because it was already slowed down by atmospheric drag. All they had to do was finish the braking process.

Finally, the splashdown impact energy was 0.34 MJ, out of a total energy budget of 574,000 megajoules. That is 0.00006%. In conclusion, atmospheric braking was the key point; the rest was just polishing.

## 9 Model Estimates Compared with Reported Mission Values

Throughout this paper, we developed equations based on physical principles and solved them using Artemis II mission parameters to estimate various physical quantities of interest. This paper was written after the mission, so these estimates are not forecasts: they are a post-mission reconstruction, and several inputs (the flyby altitude, the peak Mach number, and the parachute deployment altitudes) were taken directly from mission reports. What we can do is compare the model values with the data reported for the April 1–10, 2026 flight. This checks how well simple methods reproduce the real mission, and it shows honestly where the two-body patched conics approximation and the real mission differ.

All of the model values in this paper were computed using the two-body vis-viva equation, the Tsiolkovsky rocket equation, standard parachute drag equations, and Rankine-Hugoniot shock relations. None of these includes any form of numerical integration of the full perturbed trajectory, real-gas dynamics, or parachute inflation modeling. Where a quantity was computed rather than copied from mission data, the difference between the model and the reported value shows how far simple classical physics can go on an extremely complex engineered system.

The table uses the following color coding:

- **Green**: model within  $\pm 5\%$  of the reported value, or qualitatively correct
- **Yellow**: model within  $\pm 15\%$ , or minor qualitative discrepancy
- **Red**: model more than  $15\%$  from the reported value, or significant qualitative discrepancy
- Uncolored: not an independent check (the value was taken from mission data, or nothing comparable has been reported)

| Quantity             | Model                      | Reported                   | %<br>Err. | Note / Improvement                                                                                                          |
|----------------------|----------------------------|----------------------------|-----------|-----------------------------------------------------------------------------------------------------------------------------|
| Entry velocity at EI | 10.99 km/s<br>(24,600 mph) | 11.03 km/s<br>(24,664 mph) | 0.3%      | Close agreement. Vis-viva with an assumed TEC apogee and perigee is sufficient; the result barely depends on those choices. |
| Mach number at EI    | Mach 29 (at 360 K)         | Nearly Mach 35 (NASA)      | N/A       | Reported Mach figures depend on which speed of sound is used, so they are not a direct check of the 360 K estimate.         |

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(Table continued from previous page)

| Quantity                          | Model                                          | Reported                                                             | %<br>Err. | Note / Improvement                                                                                                                                  |
|-----------------------------------|------------------------------------------------|----------------------------------------------------------------------|-----------|-----------------------------------------------------------------------------------------------------------------------------------------------------|
| Peak Mach number                  | Not computed<br>(Mach 33 used<br>as an input)  | Mach 33<br>(PBS) to 35<br>(NASA)                                     | N/A       | The Rankine-Hugoniot calculation in Section 4 takes the reported Mach 33 as its input, so this row is not an independent check.                     |
| Peak G-load on crew               | Not computed                                   | 3.9g (NASA planned profile)                                          | N/A       | The paper gives the drag equation but does not model the deceleration profile, so there is no independent estimate.                                 |
| Lunar SOI radius                  | 66,100 km                                      | Not measured                                                         | N/A       | The SOI is a construct of the patched-conic method, not something the mission measures.                                                             |
| Hyperbolic turn angle $\delta$    | 54.4°                                          | Not publicly released                                                | N/A       | Consistent with observed free-return geometry. Full navigation data not yet public.                                                                 |
| Lunar flyby altitude              | Not computed<br>(reported value used as input) | 6,545 km                                                             | N/A       | Set by NASA's trajectory design; this paper uses the reported value rather than deriving it.                                                        |
| Drogue deploy altitude            | Not computed<br>(reported value used)          | 23,400 ft                                                            | N/A       | The dynamic-pressure estimate in Section 6 uses the reported altitude and speed.                                                                    |
| Main chute deploy alt.            | Not computed                                   | 5,400 ft                                                             | N/A       | NASA's pre-flight timeline said about 6,000 ft; this paper does not model it.                                                                       |
| Terminal velocity (3 main chutes) | 15.8 mph                                       | About 17–20 mph (NASA figures)                                       | 7–21%     | Our drag area is likely too large: clustered canopies produce less drag than isolated ones. The capsule is at terminal velocity well before impact. |
| Heat shield surface temp.         | ~3,000 K                                       | ~3,000°F ( $\approx 1,920$ K) to nearly 5,000°F ( $\approx 3,030$ K) | 1–56%     | Published figures disagree, so this is not a clean check. The shock layer ( $\sim 8,000$ – $10,000$ K) is distinct from the wall.                   |
| Water impact G-load               | 3.7g (average)                                 | Not published                                                        | N/A       | Rough work-energy estimate; the crew landed safely.                                                                                                 |

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| Quantity           | Model  | Reported              | %<br>Err. | Note / Improvement                                                                       |
|--------------------|--------|-----------------------|-----------|------------------------------------------------------------------------------------------|
| Total entry energy | 574 GJ | Not directly measured | N/A       | Thermodynamically consistent with confirmed entry speed and mass. No contradictory data. |

*Table: Model values from this paper compared with reported Artemis II mission data (a post-mission reconstruction, not a set of forecasts). Green rows agree within 5%; yellow rows show minor discrepancies; uncolored rows are not independent checks. The entry velocity estimate (about 0.3% error) is the standout result of this analysis.*

## Summary of Results

The most notable outcome represented here is the entry velocity. Using nothing beyond the vis-viva equation, an assumed trans-Earth apogee and perigee, and the two-body patched conic approach, the model entry velocity of 10.99 km/s differs from the reported 11.03 km/s by about 0.3%, roughly 70 miles per hour out of nearly 25,000. The inputs were chosen after the mission, but the result barely depends on them, since at the entry interface the speed is set almost entirely by the depth of Earth’s gravity well. That is a nice demonstration of why the vis-viva equation remains a workhorse of orbital mechanics.

Several rows are not independent checks at all. The flyby altitude, the peak Mach number, and the parachute deployment altitudes were taken from mission reports and used as inputs, and the paper was updated as confirmed data came in. The flyby altitude in particular comes out of NASA’s high-precision trajectory optimization, which cannot be reproduced from first principles and published data.

The terminal velocity disagreement most likely comes from our drag area. We multiplied the full canopy area of each parachute by a drag coefficient of 0.97, but three clustered canopies interfere with one another and produce less drag than three isolated ones; an effective drag coefficient between about 0.6 and 0.85 would reproduce the reported 17–20 mph. The capsule itself is not still slowing down at impact: under the mains it settles to terminal velocity within seconds, long before it reaches the water.

Overall, where the simple first-principles models could be checked against the mission, they landed close to the reported values, and where they could not, the table says so.

## 10 Author's Note

If you made it this far, truly, thank you so much. This is what I do in my free time, and I'm glad to share it with you and receive constructive feedback. I hope you learned something along the way. This project has taken  $\sim 50$ – $60$  hours, so please feel free to share my work with anyone who might be interested, and stay tuned because this is just the start! Thank you again for all the recent support, and I will keep sharing the math and science behind whatever is going on in our world.

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