EE-585 – Space Mission Design and Operations

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Ecole Polytechnique Fédérale de Lausanne

Week 06 – 18 Oct 2024

Today's outline

Phasing

Relative motion

Applications of rendezvous, proximity operations and docking (RPOD)

Space resident object population & debris problem

Phasing

Phasing

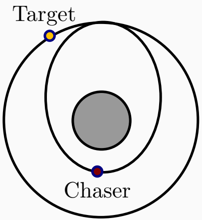
Goal of phasing: get 2 satellites close to each other (“Rendezvous”) such that they can interact (“proximity operations”)

- Possible interactions in Rendezvous and proximity Operations (RPOs): inspection, close-formation, in-orbit servicing, docking. . .

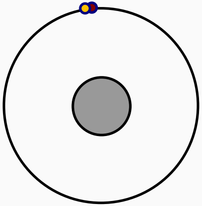
The spacecraft must be on similar orbits ($\rightarrow$ remember that inclination change is very costly!). *Assume co-planarity in the following*.

Typical case: a target object that is passive (ISS) and an approaching active spacecraft, chasing the target (e.g., Crew Dragon).

Initial situation



Final situation

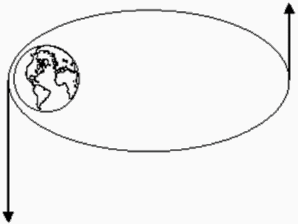


Reminder: velocity as altitude changes

As the semi-major increases, the orbital period increases and orbital velocity decreases

$$a \nearrow \iff T = 2\pi\sqrt{\frac{a^3}{\mu}} \nearrow \iff V = \sqrt{\frac{\mu}{r}} \searrow$$

The velocity of an elliptical orbit is greatest at perigee and least at apogee



Vis Viva equation:

$$V = \sqrt{\frac{2\mu}{r} - \frac{\mu}{a}}$$

$$\text{if } r \searrow \implies V \nearrow$$

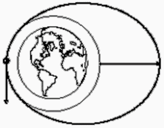
$$\text{if } r \nearrow \implies V \searrow$$

Reminder: effects of in-plane burns

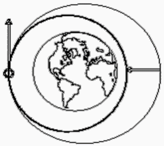
Posigrade burns increase altitude 180° from the burn $v + \Delta v \implies r_{apo} \nearrow$ (with burn at peri)

Retrograde burns decrease altitude 180° from the burn $v - \Delta v \implies r_{peri} \searrow$ (with burn at apo)

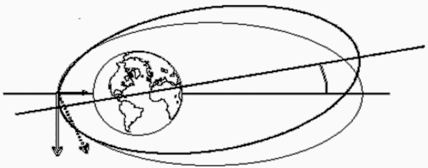
Radial burns change the argument of perigee



Posigrade burn



Retrograde burn

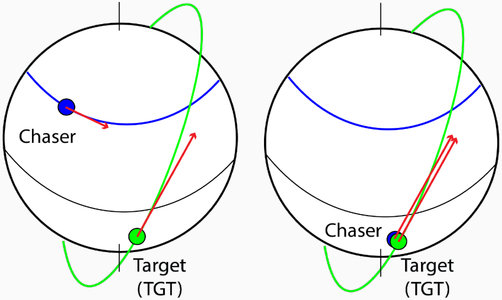


Radial burn

Rendezvous problem

The 2 spacecraft may be on very different orbits. The first stage of the rendezvous is to bring the 2 spacecraft into close proximity by performing a phasing manoeuvre.

Initial condition | Final condition



If the orbits of the chaser and the target are too different, the trajectory of chaser must be changed such that

$$r_{\text{chaser}} = r_{\text{TGT}}$$
$$v_{\text{chaser}} = v_{\text{TGT}}$$

if the orbits are too different, the Lambert problem can be solved to find the best/most efficient transfer orbit.

For a new spacecraft, the target inclination of the orbit must be equal or larger than the latitude of the chaser's launch site.

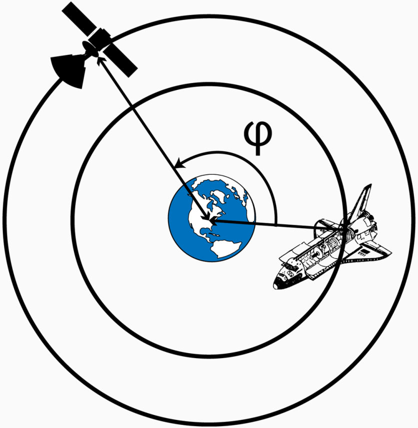
Phase angle

Assume two objects on coplanar orbits but different altitudes (remember that nodal regression will degrade coplanarity – differential nodal regression may be used to make orbits coplanar).

The phase angle φ is the angle between the chaser and target, measured from the centre of the Earth.

The phasing rate or catch-up rate is the rate at which phase angle changes.

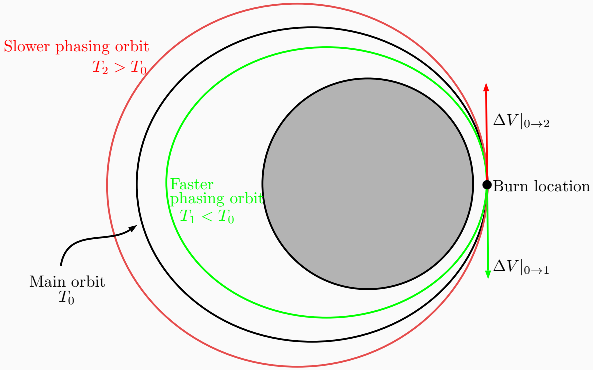
The phasing rate (synodic period) is a function of the differential altitude (third law of Kepler).



A transfer trajectory can be computed to bring the chaser to the altitude of the target and reduce the phase angle to $\varphi = 0$.

Phasing manoeuvres: Hohmann transfer

A Hohmanm transfer allows a spacecraft to return to its main orbit within a specific time.

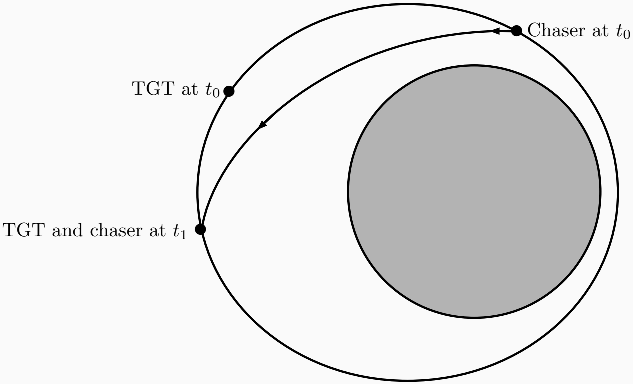


The slower phasing orbit will make the spacecraft return later because $a_2 > a_0$ and therefore $T_2 > T_0$ as $T = 2\pi\sqrt{\frac{a^3}{\mu}}$. This requires a posigrade burn.

This could be used to for one satellite to approach another one on similar orbits or be used in GEO to make very fast relocations (but not used in practice).

Phasing manoeuvres: Fast manoeuvres

While Hohmann transfer are energy-efficient, they are slow. Intercept trajectories can be found that are much faster, but costlier than Hohmann.



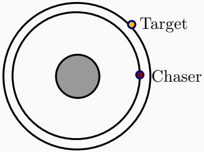
This is also a 2-burn manoeuvre $\rightarrow$ at the velocity vector of the chaser at t_1 must be aligned with the target.

Phasing manoeuvres: synodic period

Synodic period is the time between conjunctions of 2 objects, that is their nearest approach.

For the Jupiter and Earth system, the synodic period is about 13 months.

The synodic period is used to determine when 2 objects will be close to each other (e.g. for observation) or in the right geometry before a RPO manoeuvre.



The synodic period is computed by

$$\frac{1}{T_{\text{syn}}} = \frac{1}{T_1} - \frac{1}{T_2}$$

Remember that $T = \frac{2\pi}{n}$ is the mean motion (in e.g. rad/s or rev/d), so

$$n_{\text{syn}} = n_1 - n_2$$

$\rightarrow$ *This is an important concept for interplanetary travel too.*

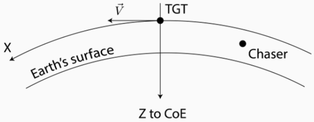
Local reference frame

The 6 Keplerian parameters are not well suited to describe the relative motion between a chaser and a target.

Let's put ourselves in a local coordinate system attached to the target: LVLH (Local Vertical, Local Horizontal)

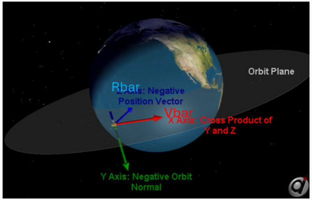
Assume a circular orbit of the target for simplicity:

- Z/Rbar: Oriented in the direction of $-\vec{r}$ (points to center of Earth) - Local Vertical
- Vbar: In the direction of the velocity vector - Local Horizontal



In a general case:

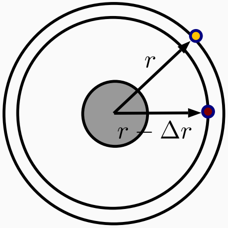
- Z/Rbar: Oriented in the direction of $-\vec{r}$ (points to center of Earth) - Local Vertical
- Y: Negative to the orbit normal $-\vec{h} = -(\vec{r} \times \vec{v})$ (Angular momentum)
- X-axis: Right-handed coordinate system - Local Horizontal



Credits: ai solutions

Catch up/overtake rate for nearby circular orbits

Initial situation



Two objects are on circular orbits of radius r and $r - \Delta r$ respectively, with $\Delta r \ll r$.

After one full orbit, the lower object will have moved forward with respect to the upper one by a distance Δx

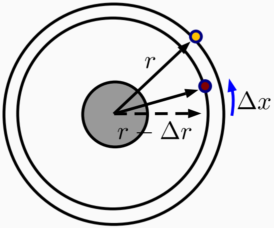
$$\boxed{\Delta x \cong 3\pi \Delta r} \qquad \text{for } \Delta r \ll r$$

The horizontal distance between the two objects changes by $\Delta x \cong 3\pi\Delta r \approx 10\Delta r$ *each orbit*.

Example: in LEO for a $\Delta r = 100$ m, the lower-orbit object would catch-up the higher object by $\Delta x \sim 1$ km/orbit or ~ 15 km/d.

If the chaser is above the target, the chaser will fall behind by Δx.

Final situation



Derivation of the catch up/overtake rate

Idea: use the linearised difference of orbital period and path travelled.

$$T(r) = 2\pi\sqrt{\frac{r^3}{\mu}}$$

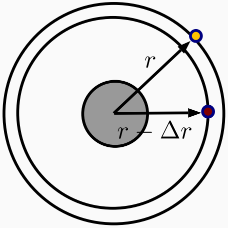
$$T(r+\Delta r) = 2\pi\sqrt{\frac{(r+\Delta r)^3}{\mu}} \approx 2\pi\sqrt{\frac{r^3 + 3r^2\Delta r}{\mu}} \overset{\text{Taylor}}{\approx} T(r)\left(1 + \frac{3}{2}\frac{\Delta r}{r}\right)$$

$$\implies \Delta T = T(r+\Delta r) - T(r) \approx \frac{3}{2}\frac{\Delta r}{r} T(r)$$

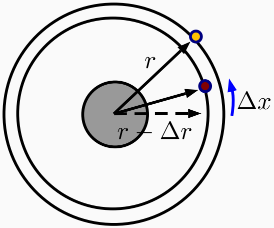
The orbital velocity is $V = \sqrt{\frac{\mu}{r}}$, thus

$$\boxed{\Delta x} = V\Delta T = \sqrt{\frac{\mu}{r}} \cdot \frac{3}{2}\frac{\Delta r}{r} \cdot 2\pi\sqrt{\frac{r^3}{\mu}} \boxed{= 3\pi\Delta r}$$

Initial situation



Final situation



Catch up rate for an elliptical orbit of the chaser

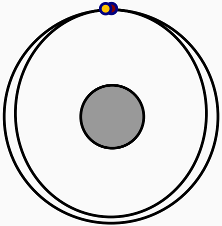
If the chaser is on an elliptical orbit, with semi-major axis $a < r$ and the two objects are initially co-located, the chaser will have moved forward by

$$\boxed{\Delta x \approx 3\pi \left(r_{\text{TGT}} - a_{\text{chaser}}\right)} \qquad \text{for } |r - a| \ll r$$

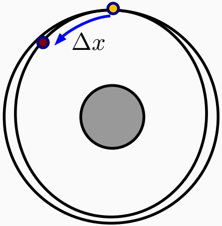
This is the horizontal distance difference per orbit.

Similarly to the circular case, if $a > r$, i.e. the semi-major axis of the chaser is higher than the target, the target will fall behind ($\Delta x < 0$).

Initial situation



Final situation



Energy considerations

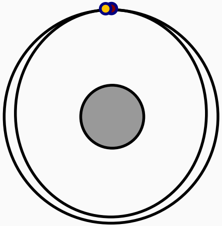
If you are in a spacecraft (say Soyuz or Crew Dragon) on an initial circular orbit of radius r, together with the ISS, for instance, and you reduce your velocity by a small amount, you also reduce the energy of your orbit, and you come to a new elliptical orbit with a semi-major axis $a < r$.

Your new orbit will have a shorter period, so you will lead ISS after one full orbit by a value of $\Delta x \sim 3\pi(r - a)$.

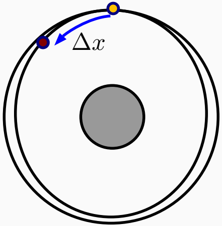
This is NOT intuitive!

You initially reduce your kinetic energy, so you reduce your total energy as well, so you reduce the size of your orbit ($a < r$), as a consequence you reduce the period of your orbit, and you move forward vs. ISS, despite reduced initial energy!

Initial situation



Final situation

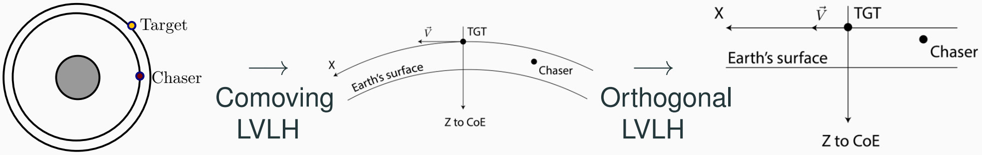


Relative motion

Projecting the motion in the LVLH frame

Let's fix our origin to the target and project the motion on the local vertical local horizontal frame (LVLH).

RPOs generally start with the chaser behind and below. You could start from ahead and above, but you would have to first raise you orbit above you target.

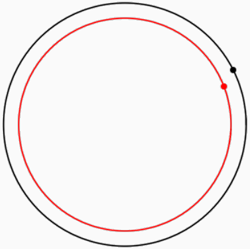


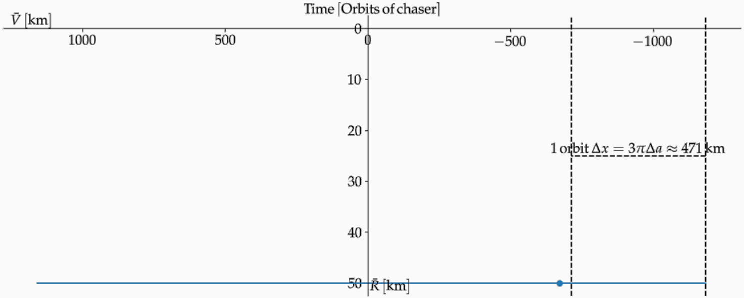
Mathematical description of the relative motion is possible but out of the scope of the course. A first order approximation of the chaser's motion is given by the Clohessy-Wiltshire equations.

Relative motion: chaser on circular orbit

Chaser 50 km lower than the target

$\rightarrow$ Constant catch-up rate





Relative motion: chaser on elliptical orbit

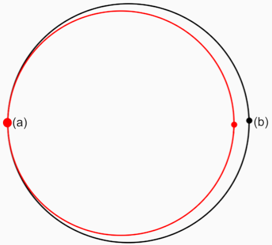
Chaser's semi-major axis 25 km lower than the target

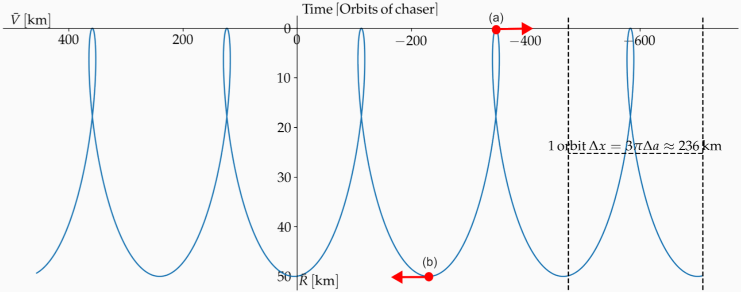
$\rightarrow$ Catch-up rate on average is $\Delta x \approx 3\pi(r - a)$

v_{chaser} changes according to Vis Viva equation:

$\rightarrow$ at (a) v_{chaser} is minimum and here $v_{\text{chaser}} < v_{\text{TGT}}$

$\rightarrow$ at (b) v_{chaser} is maximum and $v_{\text{chaser}} > v_{\text{TGT}}$

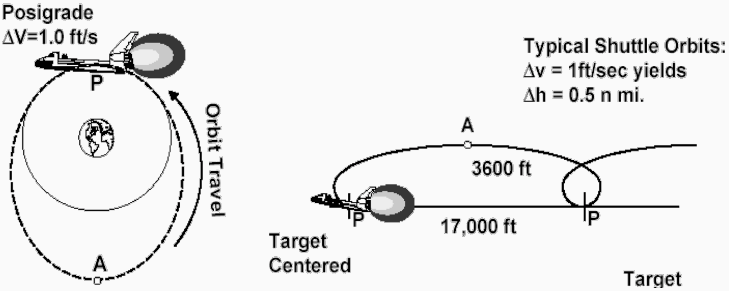




Relative motion: further examples

$\rightarrow$ Animated relative motions

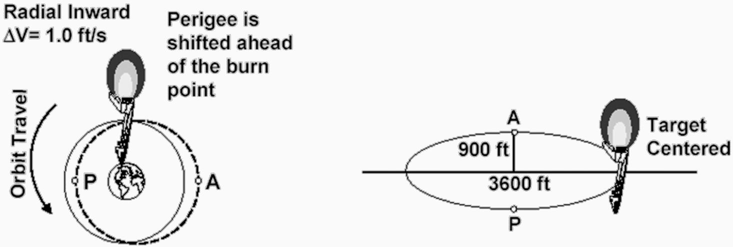
Effect of posigrade burn on relative motion



Initial condition: Shuttle and ISS are colocated.
On the left: the Shuttle is moving away from the ISS after a posigrade burnand transiting to a higher energy elliptical orbit. The Space Shuttle reaches the apogee (A) after half an orbit ($=$ 45 min), and then comes back to the same altitude as it had originally.
On the right: resulting motion of the Shuttle versus ISS, going over and behind.

For a retrograde burn, the shuttle will be on a shorter period-orbit and will move away in front of the ISS.

Effect of inward radial burn on relative motion



A radial burn is a burn perpendicular to the velocity vector: the amplitude of the velocity vector $|\vec{V}|$, the semi-major axis a, the energy ε and the period T remain unchanged.

The perigee is reached later on the new orbit.

There is no tendency to move forward or aft of ISS, the spacecraft goes on an elliptical or circular relative orbit in the vicinity of ISS.

Applications of rendezvous, proximity operations and docking (RPOD)

Motivations for RPOD

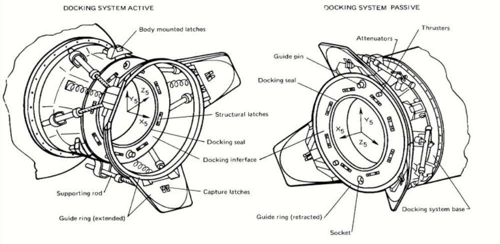
- Cooperative rendezvous:
  - Getting a crew to a space station (ISS, CSS) and rotating them.
  - Spacecraft rearrangement (e.g. Apollo LEM/command module docking), station building.
  - Formation-flying: some applications require $\lesssim 1$ km-distance formations (e.g. TerraSAR-X & TanDEM-X)
  - Servicing: propellant delivery (aka “petrol station in space“), delivery of subsystems (e.g. new propulsion module [MEV-1 mission])
- Uncooperative rendezvous:
  - Close-up inspection of object (e.g. Astroscale).
  - Rendezvous with asteroids (e.g. DART impact on Dimorphos or mining), comets (Rosetta to 67P/Churyumov-Gerasimenko), . . .
  - Active Debris Removal (ADR).

Navigation for rendezvous and proximity operations

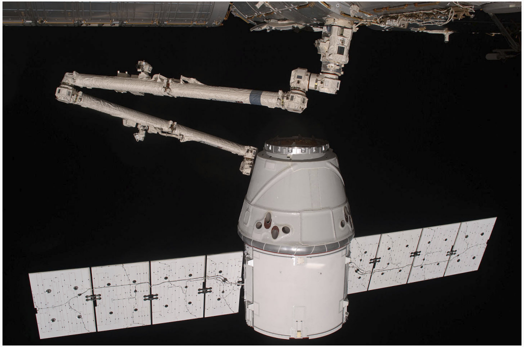
- Absolute navigation (i.e. relative to an inertial frame) is necessary during the phasing and far approach.
  - $\rightarrow$ Orbit determination
- Relative navigation (i.e. relative to the chaser or the target) has better precision in the final stages of the rendezvous.
  - Angle-only navigation: using small telescope and cameras, determine the attitude of the chaser and the position of the target. Works well for far range operations. Stereo cameras can be used at short distances.
  - Radar: works well for short & medium distances. Requires much energy.
  - Lidar/laser to recover the range.
- Pose estimation (i.e. the relative orientation of the chaser & target).
  - Attitude of the target might not always be controlled ($\rightarrow$ Uncooperative rendezvous).
  - Comparison of an optical image to a 3D model.
  - Machine-learning assisted pose estimation.

Docking

Progress about to dock with ISS. (Credits: NASA)

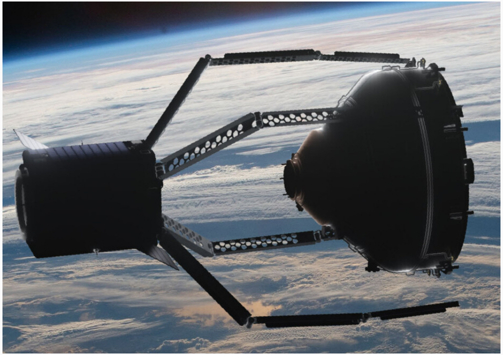


Apollo/Soyuz docking system (1975)

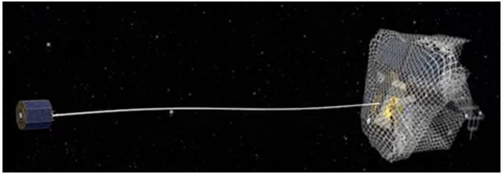


Canadarm2 grapples an early Dragon in 2012. (Credits: NASA)

Docking and servicing



ClearSpace 2020 concept. (Credits: ClearSpace)



Alternative capture mechanisms (Credits: ESA)

To facilitate docking, companies now propose standard docking plates. Not yet fully verified on orbit. (Credits: Astroscale)

RPO profiles: Automated Transfer Vehicle (ATV) missions (1/6)

This picture of the ISS was taken in 2011. It shows the Space Shuttle Endeavour (docked to the forward portion of the USOS [US Orbital Segment]), ATV-2, docked to the aft of the ROS (Russian Orbital Segment). A Soyuz and and Progress cargo.

The ATV brought resupply equipment, fuel, water, payloads and food for the crew, but it could also reboost the Station, firing thrusters to bring the Station to a higher altitude.

RPO profiles: ATV missions (2/6)

Designation	Name	Launch date	Docking date	Re-entry date
ATV-001	Jules Verne	09 Mar 2008	03 Apr 2008	29 Sep 2008
ATV-002	Johannes Kepler	16 Feb 2011	24 Feb 2011	21 Jun 2011
ATV-003	Edoardo Amaldi	23 Mar 2012	28 Mar 2012	04 Oct 2012
ATV-004	Albert Einstein	05 Jun 2013	15 Jun 2013	02 Nov 2013
ATV-005	Geores Lemaitre	29 Jul 2014	12 Aug 2014	15 Feb 2015

The re-entry of ATV was automatic after de-docking and destructive. The crew was loading ATV with trash before de-docking and re-entry.

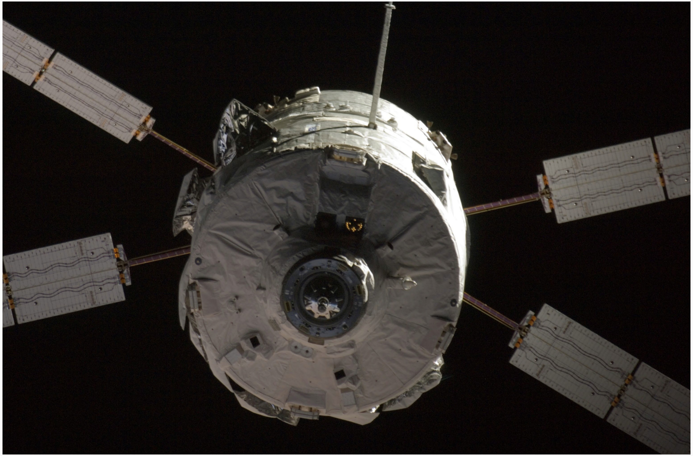
It lives on as the basis of the basis for European Service Module for the Artemis programme.

RPO profiles: ATV missions – docking (3/6)

This is a view of the ATV approaching the back of ISS, to the Russian segment.

It was using various videometers, retroreflectors, and GPS navigation to do an entirely automatic approach to the Station and docking.

There was a possibility for the crew on board the Station to monitor the approach, and command an abort if needed. It never happened.

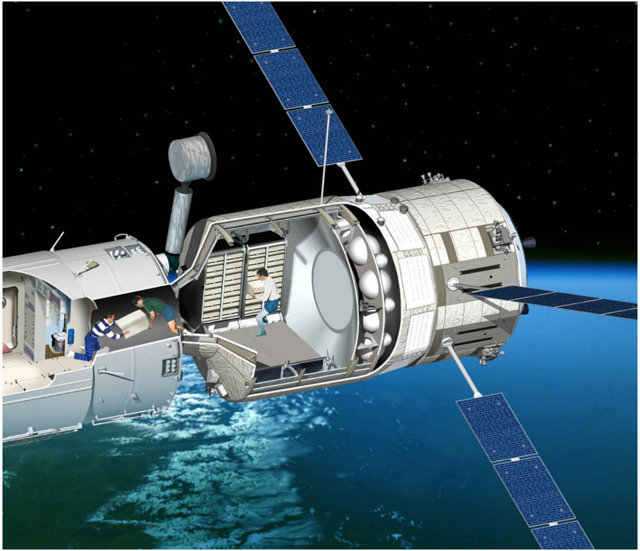


Credits: NASA

RPO profiles: ATV missions – interior (4/6)

One compartment of ATV was accessible by the crew, typically to provide food, equipment and payloads.

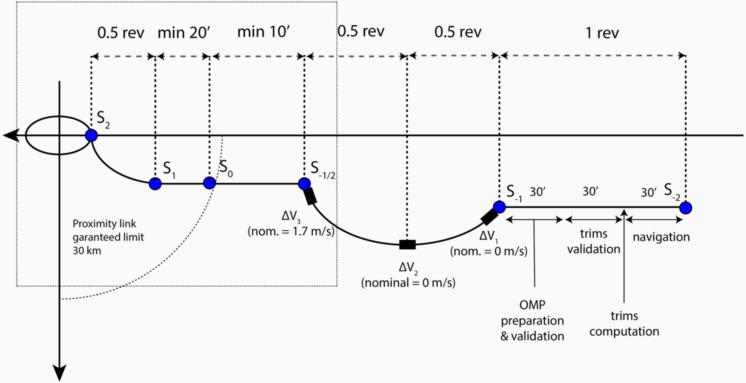
The back of the ATV contained water tanks to generate oxygen inside the ISS, also fuel and thrusters that were used to reboost ISS.



Credits: ESA, D.Ducros

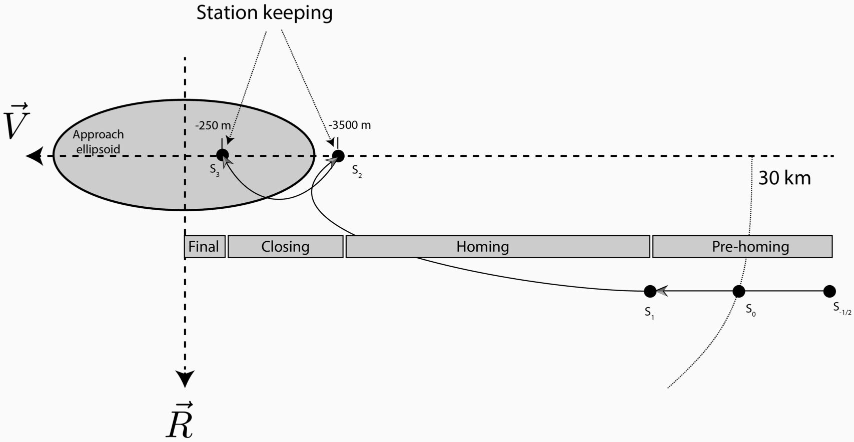
ATV was coming from behind and below the Station.

The timing was arranged so that ATV would arrive at the Station in daylight, for better visibility of the crew to the approaching re-supply vehicle.



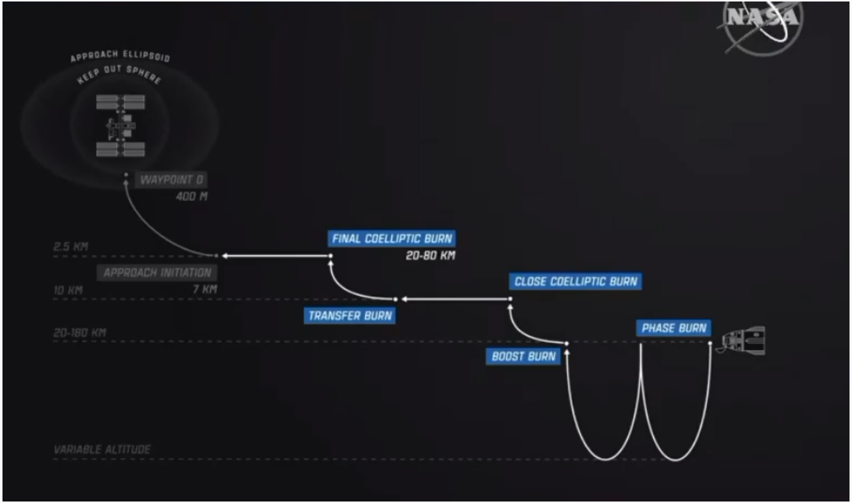
Credits: Adapted from ESA

At the S_2 point, a manoeuvre was performed to have the last relative orbit reaching S_3.



Credits: Adapted from ESA

RPO profiles: Crew Dragon/ISS – final rendezvous profile (1/3)

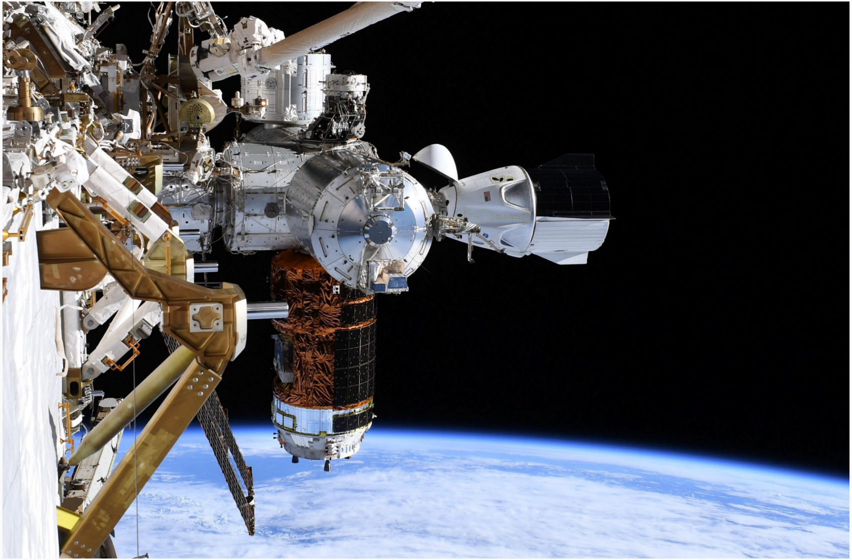


Credits: NASA TV

RPO profiles: Crew Dragon/ISS – Automatic approach to the ISS (2/3)

Credits: NASA TV

RPO profiles: Crew Dragon/ISS – Automatic approach to the ISS (3/3)



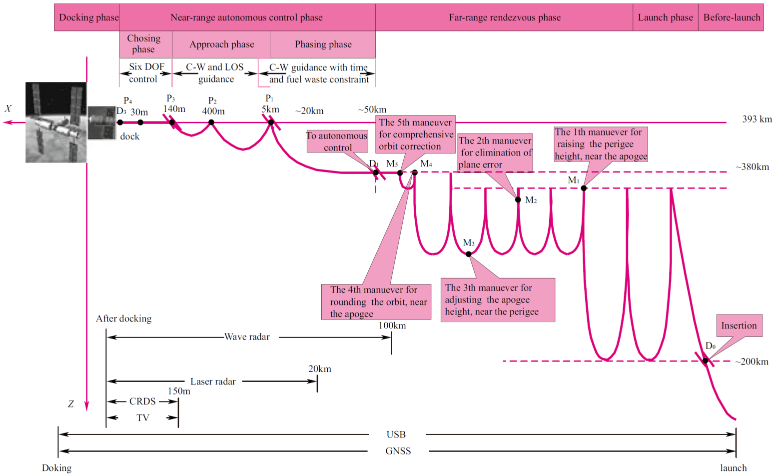
RPO profiles: Shenzhou 8 docking to CSS in 2011

China's first automatic space rendezvous and docking.

Far-range rendezvous: ground control, close range and docking automatic.

The autonomous phase divided into a homing phase, an approach phase, and a rendezvous phase. Four holding points at 5 km, 400, 140, and 30 m

Guidance based on the Clohessy-Wiltshire equation.



Credits: Xie et al., *GNC for S/C Rendezvous and Docking: Theory and Methods*, 2021

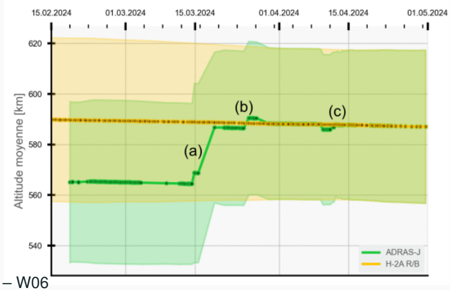
RPO profiles: Astroscale's ADRAS-J mission (1/2)

A commercial entity – Astroscale – launched the inspection satellite ADRAS-J on 18.02.2024. It inspects closely a discarded upper stage from a H-2A Japanese rocket launched in 2009. It characterises the debris to prepare a future active debris removal mission.

(a) commissioning and orbit matching

(b) first approaches, at $\sim$ 2 km min

(c) start of many rendezvous and close inspections ($<$ 100 m)

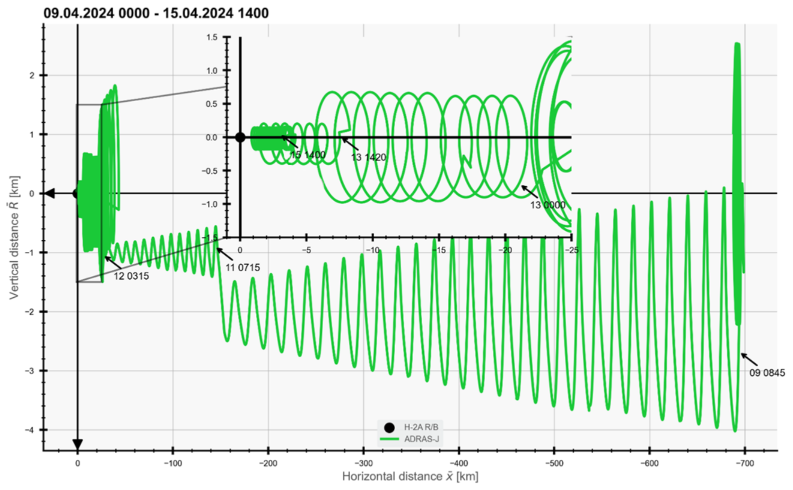


RPO profiles: Astroscale's ADRAS-J mission (2/2)

Real-life challenge: the circular orbit assumption does not hold for the target → triangular-shaped curve.

Astroscale approached in distinct phases with holds at 700 km behind the target, 25 km and ∼ 2 km. It maintained a < 1 km separation with the upper stage for ∼ 3 weeks.

The close approach, inspection and pullouts continue to this day.

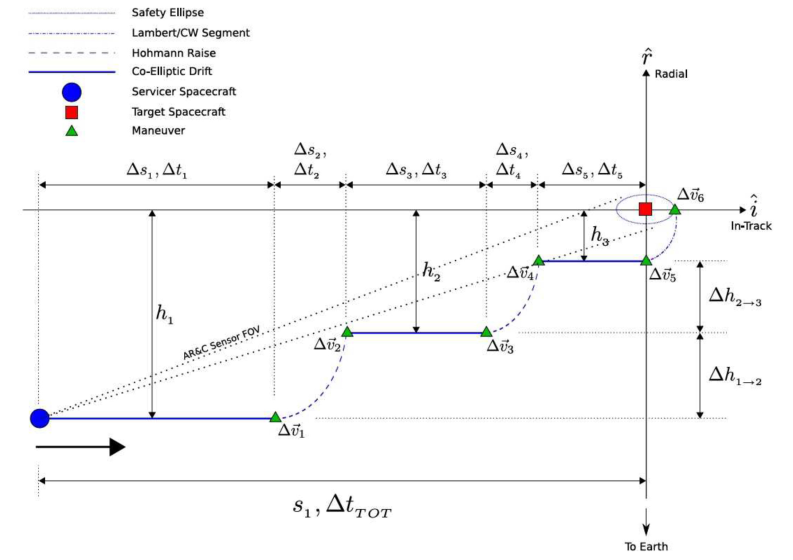


Closer images:

RPO profiles: USA vs rest of the World

US convention for the relative plot usually inverts the $\bar{x}$ axis → make sure you understand the convention.

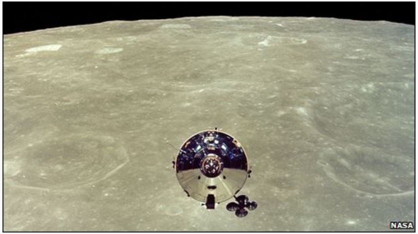
You can could compute the movement in a co-elliptic reference frame (i.e. no longer LVLH with $\bar{x}$ aligned with the velocity vector) to avoid the triangular shapes of the previous slide.



Credits: Barbee et al, 2010

Rendezvous around other bodies

It is important to realize that the RPO strategies and concepts presented here are applicable anywhere, and not only on LEO (except the LEO approximations).



They can be used in case of orbits around the Moon or planets in the Solar System. We can also derive approximations for any celestial body.

These concepts and strategies can be used for heliocentric trajectories as well, but, if we take the case of a mission from a planet to another planet, we have to take into account the gravitational effect of the departure planet at the beginning of the journey, and the same with the destination planet at the end of the trip.

→ *We will handle this in the next lectures.*

Space resident object population & debris problem

Number of successful launches per year since 1957

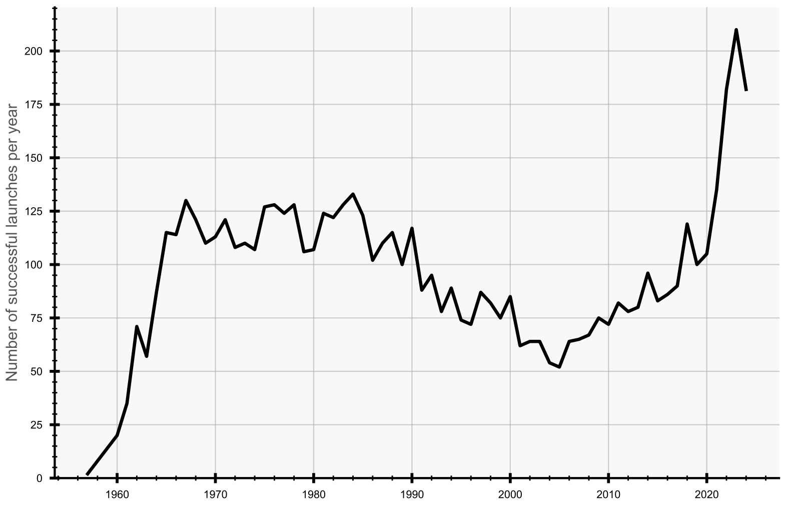
This shows the number of launches per country that reached orbit and deployed satellites.

The number of launches first was consistently high during the cold war.

It dipped to a minimum in the mid-2000s.

The number increased by a factor of 2 between 2020 and 2023.

Data as of 17 Oct 2024.



Number of satellites launched since 1957

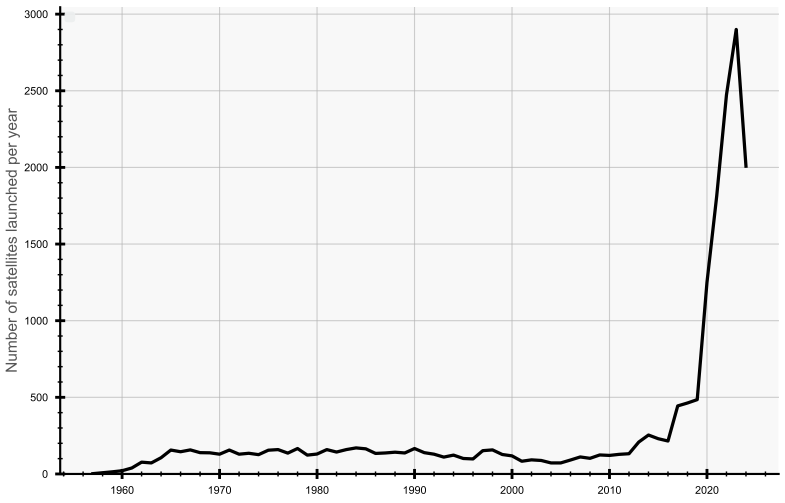
This shows the number of payloads launched per year.

Mid-60s to early-2010s, 1 launch $\cong$ 1 satellite. Rideshare launches increased since $\sim$ 2010.

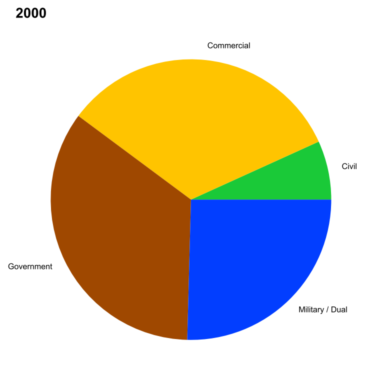
$3\times$ more operational satellites in 2024 as compared to 2021!

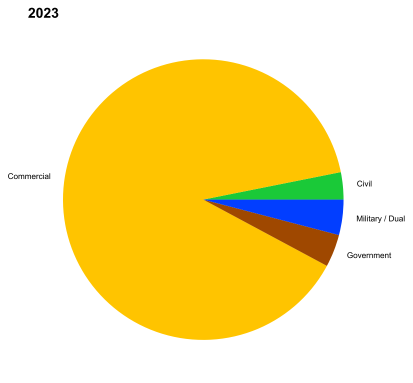
Explosion of satellites launched mainly due to Starlink (60 satellites/launch in 1st generation).

Data as of 17 Oct 2024.



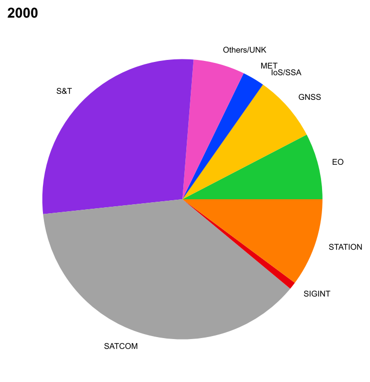
Description of the satellites launched: type of use

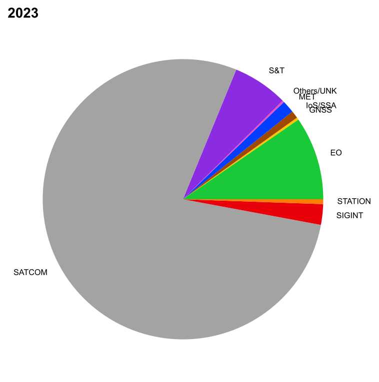




Big shift from governmental & military satellites in 2000 towards commercial utilisations in 2023.

Description of the satellites launched: function





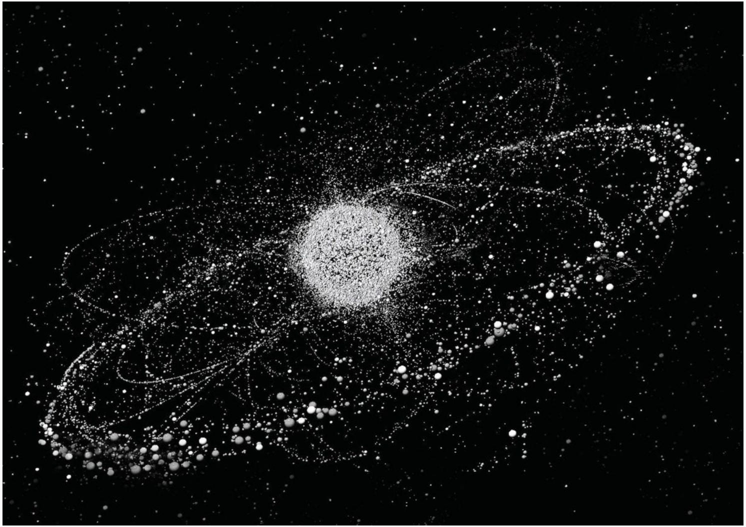
Big shift from governmental-driven science & technology and crewed mission to SATCOM. Rise in the relative fraction of Earth observation satellites (EO, SIGINT) for governmental and military purposes.

Operational satellites and debris

Space debris consist in detached fragments of satellites and many rocket upper stages.

A satellite which is not working any longer is also considered a debris (either failed, or end of life).

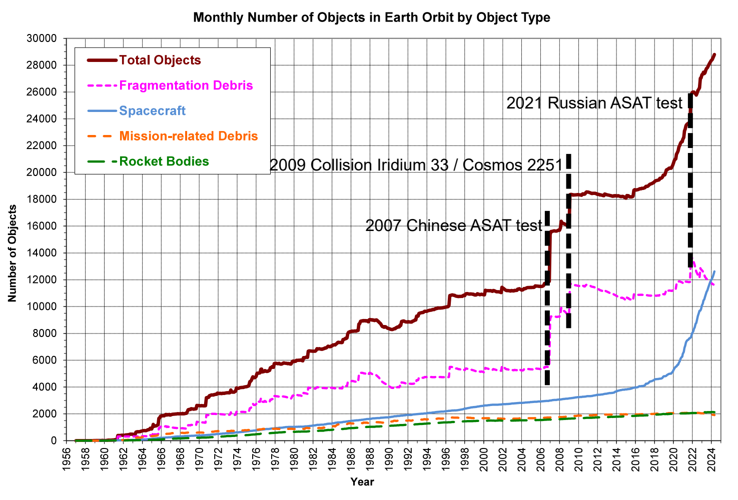
In LEO, there are many more debris than satellites! (about 7,500 satellites and $> 20,000$ debris more than 10 cm in size).



The number of debris/non-operational object is soaring too

Debris-generating events:. . .

- launch (rocket bodies, adaptors, . . .)
- end of life (no re-entry)
- breakups / fragmentations
- collisions
- intense space weather events
- anti-satellite tests



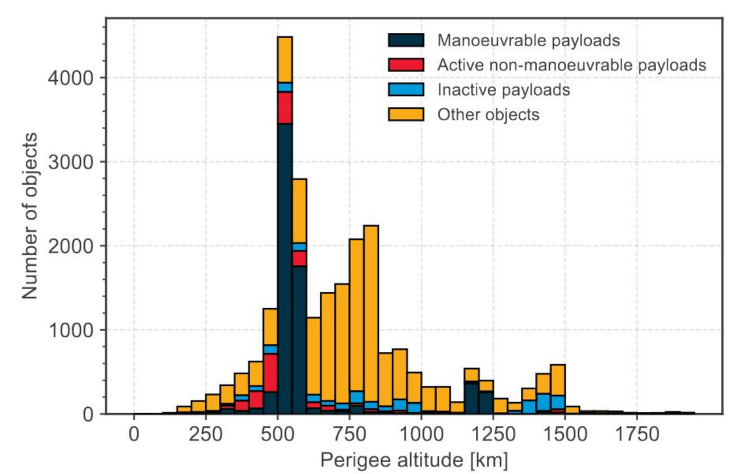
Data: June 2024, Credits: NASA Debris Programme Office

Altitude distribution of payloads and debris

Objects are not distributed uniformly with altitude or inclination.

First generation of Starlinks at $\sim 520 - 570$ km

Higher peak at $\sim$ 800 km from the Irdium 33/K2251 collision.



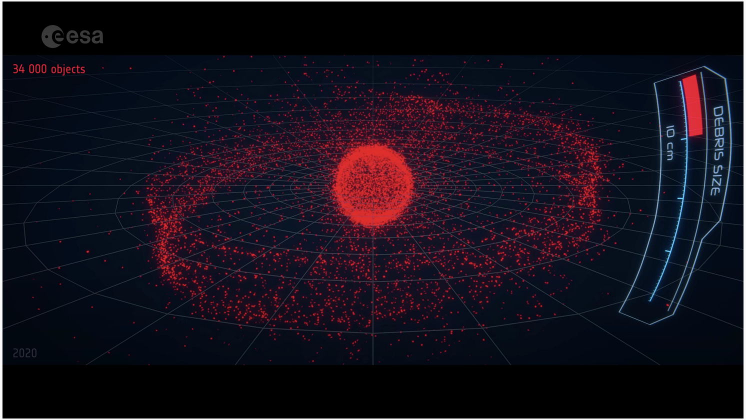
Credits: ESA Space environment report 2024

Estimation of number of debris

Objects must be typically larger than $\sim$ 10 cm to be tracked in LEO $\rightarrow$ Many un-tracked objects!

10^6 lethal and mostly non-trackable objects, size $\sim 1 - 10$ cm

$130 \cdot 10^6$ mostly non lethal and non-trackable objects, size $\sim 0.1 - 1$ cm

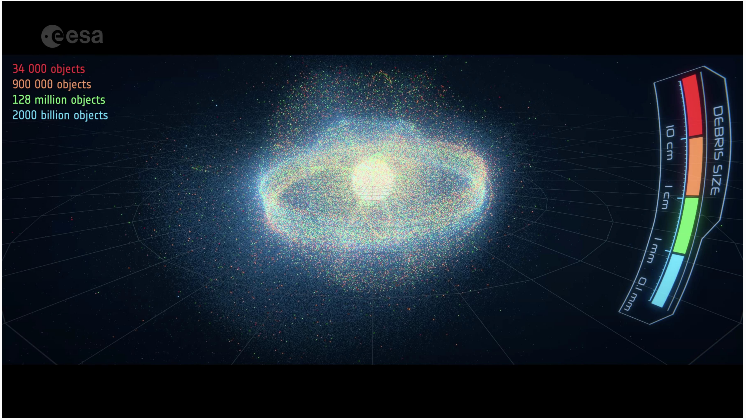


Estimation of number of debris

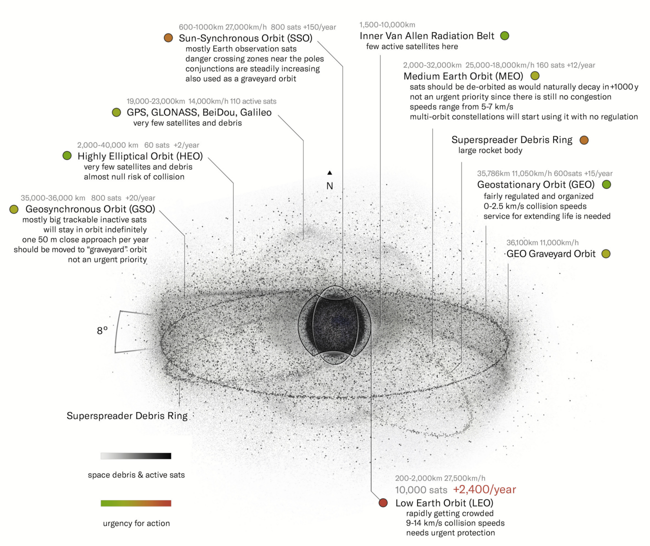
Objects must be typically larger than $\sim$ 10 cm to be tracked in LEO $\rightarrow$ Many un-tracked objects!

$\sim 10^6$ lethal and mostly non-trackable objects, size $\sim 1 - 10$ cm

$\sim 10^8$ mostly non lethal and non-trackable objects, size $\sim 0.1 - 1$ cm



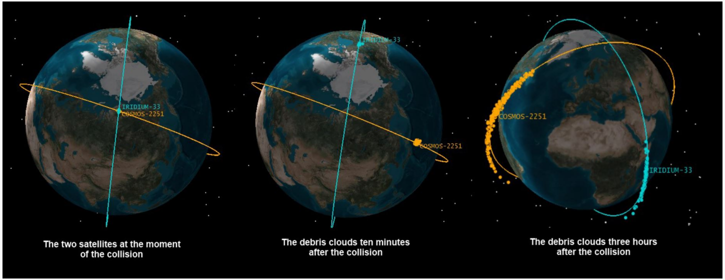
Estimation of number of debris



Credits: Wikipedia, Pablo Carlos Budassi

2009 Iridium 33 / Cosmos 2251 collision

On 10 Feb 2009 an operational Iridium satellite collided with a non-operational Cosmos 2251 at about 800 km and 11.7 km/s. $\sim$ 2000 pieces of debris generated.



The debris cloud will remain on orbit for decades.

Although the orbital data was available, there was no daily screenings for possible collisions and no procedure to quickly manoeuvre.

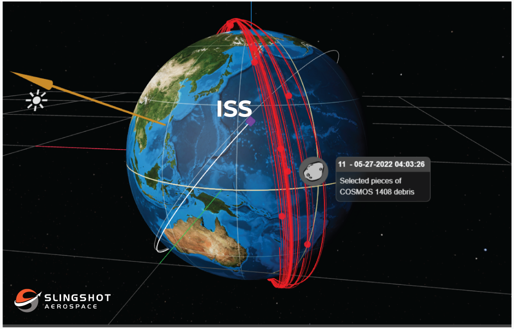
Legally, the launching state is responsible, but K2251 was launched by the Soviet Union, so unclear situation.

2021 Russian direct-ascent anti-satellite (ASAT) test

On 15 Nov 2021, a missile was launched from Plesetsk (Close to St Petersburg, Northern Russia).

It intercepted Cosmos 1408, a Soviet-era non-op. SIGINT launched in 1982 at 480 km/$i = 83°$.

A cloud of debris was generated which prompted the 7 ISS crew members to shelter in their descent craft.



Some of the debris were on a orbit that is close to the ISS $\rightarrow$ reccurent approches.

At least 1800 pieces, some with lifetimes of > 10 years.

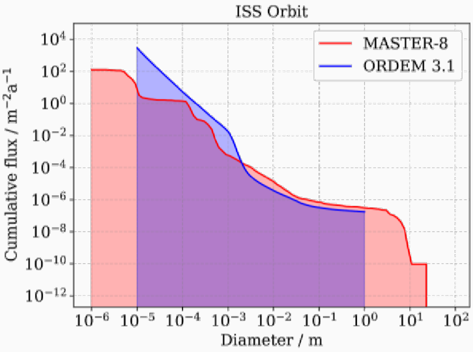
The USA, China, India and Russia have performed ASAT test. There is a moratorium for ASAT tests now, but without China, India and Russia.

Orbital debris flux at the orbit of the ISS

The flux of meteoroids is less than the flux of orbital debris for LEO for most sizes.

Meteoroids are light and small, but have typically $v \sim 17$ km/s $\rightarrow$ kinetic energy is substantially greater than with orbital debris.

Models of the debris population can be used to simulate the flux (ORDEM by NASA, DRAMA by ESA).

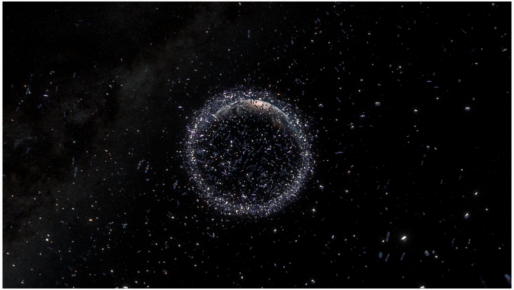


Horstmann et al, 2021

Mitigations and disposal guidelines

Mitigations at the design stage:

- Go to the right orbit
- Shield the critical systems
- Redundancy for the important systems
- Go small
- Minimise the cross-section, A_N
- Zero-debris generation
- Active/Passive debris removal?



Credits: ESA

Post-mission disposal guidelines:

- LEO: re-entry within 5 yr (with 90% successful disposal rate).
- LEO: must have passive de-orbit mechanisms for high-risk S/C (e.g. deployable solar sails).
- GEO: go to a graveyard orbit above GEO, make sure the S/C stays outside of protected zone ($\implies \sim z_{\mathrm{GEO}} + 300$ km), passivation.

Interactive quiz

→ EchoPoll platform

- You can scan a QR code or go to the link
- EchoPoll is the EPFL-recommended solution
- You do not have to register, just skip entering a username and/or email address