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DOCUMENT

Guidelines for the computation of Delta-V and propellant budget

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1 INTRODUCTION

1.1 Objective of the document

The objective of this document is to provide guidelines to compute the ΔV and the propellant budgets for Low Earth Orbit satellites during phases A/B/C/D/E.

These guidelines shall be adopted as reference for the computation ΔV and the propellant budgets, however in case of conflicts with the requirement established in the mission Satellite System Requirement Document (SSRD) shall take precedence.

1.2 Structure of the document

The document is composed of two main sections.

The first section gives the general assumptions for the computation of the ΔV by defining the typical mission phases, the common definitions and phase-specific definitions.

The second section provides the procedure to compute the propellant mass. The computation starts from end of life operations proceeding backward in the mission timeline until beginning of life operations.

2 REFERENCE DOCUMENTS

RD[01] ECSS-E-ST-10-04C – Space environment, 15 Nov 2008

RD[02] ESA/ADMIN/IPOL(2014)2, Space Debris Mitigation Policy for Agency Projects, 28 Mar2014

RD[03] ECSS-U-AS-10C, Space Sustainability: Adoption Notice of ISO 24113: Space Systems - Space Debris Mitigation Requirements, 10 Feb 2012

3 DEFINITIONS

Satellite dry mass
The satellite dry mass is defined as the mass of the satellite including unit margins and the system margin according to the policy specified in the SSRD.

Aerodynamic drag coefficient
The coefficient of drag, C_d, is a dimensionless quantity which reflects the satellite's susceptibility to drag forces.

Cross-section area
The cross-section area, A, is defined to be the area which is normal to the satellite's velocity vector.

Ballistic Coefficient
The ballistic coefficient, $BC = m/(C_dA)$, is another measure of a satellite's susceptibility to drag effects. With this definition, a low BC means drag will affect the satellite a lot, and vice versa.

Trailing formation
In case the mission is composed of various satellites flying in the same orbital plane or on the same ground track this is called trailing formation.

4 ASSUMPTIONS FOR ΔV COMPUTATIONS

4.1 Mission phases

For a spacecraft in LEO, during its mission lifetime, three main phases can be identified:

- Beginning of life operations
- Mission operations
- End of life operations

The mission ΔV is computed by simple addition of the ΔV contributions from all manoeuvres performed during the mission lifetime. All the following contributors shall be taken into account in the computation of the total ΔV budget:

- Beginning of life
  - Launcher injection errors correction
  - Orbit acquisition
- Mission operations
  - Orbit transfer manoeuvres
  - In-plane and out of plane orbit control manoeuvres
  - Constellation/Formation Maintenance
  - Collision Avoidance Manoeuvres (CAM)
  - Attitude control based on thrusters
- End of life
  - Clearance from operational orbit
  - Uncontrolled re-entry
  - Controlled re-entry

The list shall be complemented with mission-specific entries when applicable.

4.2 Aerodynamic drag coefficient

In absence of more accurate methods for computing the drag coefficient (and its evolution if used for the decay prediction) a constant value of C_d = 3.0 shall be assumed.

4.3 Cross-section area

The cross-section area may be different depending on the mission phase therefore the following assumption shall be considered for its computation:

- Beginning of life until start of end of life operations
  - The projection of the satellite normal to the velocity vector (for drag) and normal to the Sun vector (for solar radiation pressure). The average shall be calculated over 15 orbits to account for changes in attitude profile over one day.
- End of life
  - The cross-section area for de-orbiting is needed to compute the target orbit to be reached to comply with space debris mitigation requirements. If it is demonstrated that during the decay from the de-orbiting altitude to the actual re-entry in the atmosphere the spacecraft have a stable aerodynamic position,

the cross-section area resulting from such stable position shall be assumed. Otherwise, it shall be assumed that the spacecraft will be tumbling in a random manner during the decay. An average cross section area from a tumbling situation shall be then calculated. The ESA tool DRAMA shall preferably be used for the computation.

4.4 Launch epoch

The launch date to consider is the one targeted at the time when the computation is carried out. Worst-case scenarios for delayed launch dates (up to 5 years) shall also be considered in order to assess the impact of the extra propellant needed on the selected tank.

4.5 Mission lifetime

The mission duration specified in the SSRD (or any other applicable document) shall be used for the computation of ΔV.

4.6 Space environment

RD[01] is the applicable standard for space environment. The values of geomagnetic activity and solar activity indexes provided in table A-1 of the RD[01] shall be used as reference and extended by repetition of the 11-year cycle assuming September 2020 as start of cycle 25.
In particular:

- Mean values of the geomagnetic activity and solar activity indexes shall be considered for de-orbiting manoeuvres
- Maximum values of geomagnetic activity and solar activity indexes shall be considered for all other types of manoeuvres

4.7 Margins on ΔV

No margins shall be applied to the computation of the ΔV, except for the final burn in case of a controlled re-entry, where a margin of 15% shall be applied.
A factor 2 shall be applied on the number of collision avoidance manoeuvres, as further explained in section 4.10.4.

4.8 Manoeuvres duration

For the purpose of ΔV computation all manoeuvres shall be considered as impulsive. The gravity losses associated to the long firing of the thrusters, such e.g. during the controlled re-entry, will be accounted for in the propellant budget, as explained in section 5.5.1.

4.9 Beginning of life operations

4.9.1 Launcher injection errors correction

Launcher injection errors shall be considered as per launcher user manual specifications unless otherwise stated in the applicable SSRD. 2σ values shall apply if no specifications are given in the SSRD or any other applicable document.

The launcher injection errors to be corrected depend on the specific orbit requirements of the mission. If they were to be corrected the following guidelines applies:

Semi-major axis and eccentricity vector
The eccentricity vector needs to be corrected whenever a specific eccentricity (e.g. frozen eccentricity) is targeted. To compute the correction, the argument of perigee shall be assumed to be off target by 180 deg, unless specific information is provided by the launcher authorities. The correction of semi-major axis and eccentricity vector is done together by means of in-plane manoeuvres. For budgeting purposes, the individual contribution of correcting the semi-major axis and the eccentricity vector shall be calculated, including on the budget only the largest amount of the two of them.

Inclination
The correction of the inclination shall be computed separately from the correction of the semi-major axis / eccentricity vector, unless otherwise specified in the SSRD.

Right ascension of ascending node (RAAN)
RAAN injection errors shall be corrected if the resulting mean local solar time (MLST) of the orbit is not within the tolerance specified in the SRD or any other applicable document. If RAAN correction is performed by manoeuvring the satellite to a drifting orbit the maximum time allowed for the satellite in the drifting orbit shall not exceed the duration of the platform commissioning, unless otherwise stated in the SRD.

4.9.2 Orbit acquisition

The amount and type of orbit acquisition manoeuvres is mission dependent and may include both out of plane and in plane manoeuvres.
The maximum allowed period to acquire the nominal orbit shall be considered when defining the needed orbit acquisition manoeuvres. In case no maximum period is defined, a value shall be proposed by the contractor and approved by the Agency.

4.10 Mission operations

4.10.1 Orbit transfer manoeuvres

In case the mission foresees different orbits, the ΔV associated to all the orbit transfer(s) shall be computed and taken into account in the budget.

4.10.2 In-plane and out of plane orbit control manoeuvres

In-plane and out of plane orbit control manoeuvres shall be performed to maintain the orbit deviation from the nominal one within the conditions, if any, specified in the SSRD.

In-plane manoeuvres shall be computed taking into account the foreseen period of operations as derived from the assumptions on launch epoch (4.4) and mission lifetime (4.5).

The computation of the atmospheric density shall take into account the index values related to such foreseen period of operations and the assumptions of section 4.6.
Out of plane manoeuvres shall be computed separately from in-plane ones.

4.10.3 Constellation/Formation Maintenance

In case the mission is composed of various satellites (constellation or tandem), the required ΔV for the formation maintenance shall be computed. The maintenance manoeuvres shall include, when applicable:

- Orbit maintenance manoeuvres performed to follow the manoeuvres of the "master" satellite of the constellation.
- Reacquisition of the constellation in case the geometry is lost due to a contingency.
- Additional orbit manoeuvres needed to comply with the constellation requirements, if and as applicable.

The assumptions to be considered for the cross-section area of all the spacecraft are the same described in 4.3, unless additional information is available for any of the satellite of the constellation (e.g. as it could be the case for a master-slave constellation in which the ballistic coefficient of the master is already known).

4.10.4 Collision Avoidance Manoeuvres (CAM)

Collision avoidance manoeuvres shall be performed in order to avoid collision with space debris and with any active spacecraft (including constellation or formation flying cases).

The number of CAM depends on the size of the spacecraft and on the space debris distribution for the relevant orbit. The latest version of the ESA maintained tool DRAMA (https://sdup.esoc.esa.int/web/csdtf/home) shall be used to compute the amount of CAM needed during the mission operational lifetime. An accepted collision probability level (ACPL) value of 10^-4 shall be used as input for the analysis in DRAMA.

In order to account for uncertainties such as major events which may lead to a sudden debris population increase, the analysis shall be conducted considering the expected debris population at the end of the mission lifetime and a margin factor of 2 on the total number of manoeuvres.

In all cases the ΔV needed for each CAM shall be calculated assuming an altitude change manoeuvre of 200 m (not followed by any circularisation) and a subsequent manoeuvre to place the spacecraft back into its nominal orbit.

4.10.5 Attitude control based on thrusters

The ΔV necessary for any attitude control based on thrusters (e.g. safe mode, sun avoidance) and wheels off-loading (in case this is performed using propulsion instead of magnetic actuation) shall be considered and computed according to the estimated number of these events occurring during the mission lifetime, if and as applicable.

This entry shall also take into account the ΔV needed for orbit re-acquisition after the occurrence of thruster-based events.

4.11 End of life operations

In accordance with RD[02] and RD[03] every spacecraft operating in the LEO protected region shall limit its post-mission presence in the LEO protected to a maximum of 25 years from the end of mission. Furthermore, the maximum acceptable casualty risk for any re-entry event (controlled or uncontrolled) shall not exceed 1 in 10,000.

Should a (set-of) de-orbiting manoeuvre(s) be necessary to comply with these requirements the following assumptions shall be considered for the computation of the de-orbiting ΔV.

4.11.1 Clearance from operational orbit

The operational approach to perform the end of life disposal may foresee one or more manoeuvres to move the satellite away from its operational orbit. For budgeting purposes, a semi-major axis drop of 2 km TBC shall be considered.

4.11.2 Uncontrolled re-entry

All the manoeuvres associated with an uncontrolled re-entry shall be considered for the computation of the total ΔV budget.

4.11.3 Controlled re-entry

If the satellite will undergo a controlled re-entry the ΔV shall be computed considering all the manoeuvres needed to reach the target perigee.
The altitude of the last perigee before the last burn shall be such that the attitude control of the satellite is still possible. From this perigee, controlled re-entry shall be achieved in a single manoeuvre. The losses associated to the fact that the manoeuvres are not impulsive shall be properly accounted for in the ΔV budget.
The 15% margin on the last burn comes on top of the losses previously computed.

5 ASSUMPTIONS FOR PROPELLANT MASS COMPUTATION

The propellant computation procedure starts with the computation of the propellant required to perform the de-orbiting manoeuvre and then proceeds backward in the mission timeline ending with the propellant needed for the correction of launcher injection errors.

In order to simplify the propellant mass computation, all the ΔV contributors shall be summed-up and aggregated according to the mission phase to which they belong, as explained below:

$$\Delta \boldsymbol{v}_{\boldsymbol{BOL}} = \Delta v_{launcher\ inj\ errors} + \Delta v_{orbit\ acquisition}$$

$$\Delta \boldsymbol{v}_{\boldsymbol{OP}} = \Delta v_{orbit\ transfers} + \Delta v_{formation\ maintenance} + \Delta v_{orbit\ control} + \Delta v_{CAM}$$

$$\Delta \boldsymbol{v}_{\boldsymbol{EOL}} = \Delta v_{clearance} + \Delta v_{re-entry}$$

Mission-specific entries shall be added when applicable. The mapping between ΔV and propellant mass is shown in Table 1.

Phase	ΔV contribution	Total ΔV	Propellant Mass
Beginning of life	Launcher injection errors correction	ΔV_{BOL}	$Prop_{BOL}$
	Orbit acquisition		
Mission operations	Orbit transfer manoeuvres	ΔV_{OP}	$Prop_{OP}$
	In-plane and out of plane orbit control manoeuvres		
	Constellation/Formation Maintenance		
	Collision Avoidance Manoeuvres (CAM)		
FDIR	Attitude control based on thrusters	ΔV_{FDIR}	$Prop_{FDIR}$
End of life	Clearance from operational orbit	ΔV_{EOL}	$Prop_{EOL}$
	Uncontrolled re-entry		
	Controlled re-entry		

Table 1: Mapping between ΔV and propellant budget contributors

5.1 Satellite dry mass

The satellite dry mass shall include all unit margins, plus the system margin according to the policy specified in the SSRD. A 15% system margin shall apply if no specifications are given in the SSRD or in any other applicable document.

5.2 Margins on propellant

No margins shall be applied to the computation of the propellant, except for attitude control based on thrusters, where a margin of 100% shall be applied.

5.3 Propellant residuals and gauging

Typically, a small amount of propellant remains trapped in valves, injectors of the propulsion subsystem or attached to the wall of the tank and cannot be used for producing thrust. A value of 1% of the maximum propellant load allowed in the selected tank shall be considered as propellant residuals in the propellant budget ($Prop_{RES}$).

A correct estimation of the remaining propellant at the end of the mission is difficult and depends for instance on factors such as propellant loading uncertainty and gauging accuracy. A value of 2% of the maximum propellant load allowed in the selected tank shall be taken into account in the propellant budget ($Prop_{UNC}$).

5.4 Thruster efficiency

In order to take into account all factors that degrade the actual performance of the thrusters an efficiency parameter η is defined as:

$$\eta = cos(\alpha) \cdot \eta_{misalignment} \cdot \eta_{plume\_loss} \cdot \eta_{modulation} \cdot \eta_{gravity\_loss}$$

being:

- *cos(α)*: the scalar product between the unit vector describing the orientation of the thruster in a spacecraft centred frame and the unit vector defining the desired direction of thrust.
- $\eta_{misalignment}$: a factor taking into account the effect of actual misalignment of the thrusters with respect to their nominal direction of thrust. It is computed as the cosine of the misalignment angle. This will include also the effect of attitude errors during the thrust phase, if and as applicable.
- η_{plume_loss}: a factor taking the thrust efficiency reduction due to thrusters plume impinging on satellites exposed surfaces (e.g. solar arrays). It shall be expressed as an absolute number.
- $\eta_{modulation}$: a factor taking into account the effect of the thrusters on/off modulation on the total thrust. It shall be expressed as an absolute number.

If different thrusters are used for different manoeuvres, the efficiency η shall reflect such change.
The efficiency parameter can have different values depending on the type of manoeuvre under consideration (e.g. η of disposal can be different from η of nominal operation due to the likely presence of gravity losses during the re-entry manoeuvres).

5.5 Computational procedure

5.5.1 End of life

The following formula shall be used to compute the propellant needed for end of life manoeuvres:

$$Prop_{EOL} = \left(m_{dry} + Prop_{RES} + Prop_{UNC}\right)\left(e^{\frac{\Delta V_{EOL}}{V_{e\_EOL}}} - 1\right)$$

being:

- m_{dry} = dry mass of the satellite as indicated in section 5.1
- ΔV_{EOL} = total ΔV needed for the end of life manoeuvres, including the margins indicated in section 4.7.
- $V_{e\_EOL} = I_{sp\_EOL} \cdot \eta_{EOL} \cdot g_o$: exhaust velocity at end of life.
- I_{sp_EOL} = average specific impulse at end of life.
- η_{EOL} = thruster efficiency associated to end of life manoeuvres.
- g_o = standard gravitational acceleration, equal to 9.80665 m/s².

5.5.2 Mission operations

The following formula shall be used to compute the propellant needed for mission operations manoeuvres:

$$Prop_{OP} = \left(m_{dry} + Prop_{RES} + Prop_{UNC} + Prop_{EOL}\right)\left(e^{\frac{\Delta V_{OP}}{V_{e\_OP}}} - 1\right)$$

being:

- ΔV_{OP}: total ΔV needed for mission operations manoeuvres.
- $V_{e\_OP} = I_{sp\_OP} \cdot \eta_{OP} \cdot g_o$: exhaust velocity during mission operations.
- I_{sp_OP}: average specific impulse during mission operations.
- η_{OP}: thruster efficiency associated to mission operations manoeuvres.

5.5.3 Attitude control based on thrusters

The following formula shall be used to compute the propellant needed for attitude control based on thrusters:

$$Prop_{FDIR} = 2 \cdot \left(m_{dry} + Prop_{RES} + Prop_{UNC} + Prop_{EOL}\right)\left(e^{\frac{\Delta V_{FDIR}}{V_{e\_FDIR}}} - 1\right)$$

being:

- ΔV_{FDIR}: total ΔV needed for attitude control based on thrusters.

- $V_{e\_FDIR} = I_{sp\_OP} \cdot \eta_{FDIR} \cdot g_0$: exhaust velocity during attitude control based on thrusters.
- I_{sp_OP}: average specific impulse during mission operations.
- η_{FDIR}: thruster efficiency associated to mission attitude control based on thrusters.

A factor 2, representing a margin of 100% is applied here to account for uncertainties in mission design and system performance.

5.5.4 Beginning of life

The following formula shall be used to compute the propellant needed for beginning of life manoeuvres:

$$Prop_{BOL} = \left(m_{dry} + Prop_{RES} + Prop_{UNC} + Prop_{EOL} + Prop_{OP}\right)\left(e^{\frac{\Delta V_{BOL}}{V_{e\_BOL}}} - 1\right)$$

being:

- ΔV_{BOL}: total ΔV needed for beginning of life manoeuvres.
- $V_{e\_BOL} = I_{sp\_BOL} \cdot \eta_{BOL} \cdot g_0$: exhaust velocity at beginning of life.
- I_{sp_BOL}: average specific impulse at beginning of life.
- η_{BOL}: thruster efficiency associated to beginning of life manoeuvres.

5.5.5 Total propellant mass

The total propellant mass is therefore:

$$Prop_{TOT} = Prop_{EOL} + Prop_{OP} + Prop_{FDIR} + Prop_{EOL} + Prop_{RES} + Prop_{UNC}$$

6 SUMMARY TABLE

All the entries for the ΔV and propellant budget shall be summarized in a table as the one shown in Table 2. The list given in the mission phase column shall be complemented with mission-specific entries when applicable.

<table>
<tr><th>Phase</th><th>ΔV contribution</th><th>Total ΔV [m/s]</th><th>Propellant [kg]</th><th>Margins</th><th>I_{sp} [s]</th></tr>
<tr><td rowspan="2">Beginning of life</td><td>Launcher injection errors correction</td><td rowspan="2">ΔV_{BOL}</td><td rowspan="2">$Prop_{BOL}$</td><td rowspan="2">None</td><td rowspan="2">I_{sp} at BOL</td></tr>
<tr><td>Orbit acquisition</td></tr>
<tr><td rowspan="4">Mission operations</td><td>Orbit transfer manoeuvres</td><td rowspan="4">ΔV_{OP}</td><td rowspan="4">$Prop_{OP}$</td><td rowspan="3">None</td><td rowspan="5">Average I_{sp} of mission operational phase</td></tr>
<tr><td>In-plane and out of plane orbit control manoeuvres</td></tr>
<tr><td>Constellation/Formation Maintenance</td></tr>
<tr><td>Collision Avoidance Manoeuvres (CAM)</td><td>100% on the number of CAM</td></tr>
<tr><td>FDIR</td><td>Attitude control based on thrusters</td><td>ΔV_{FDIR}</td><td>$Prop_{FDIR}$</td><td>100% on propellant</td></tr>
<tr><td rowspan="3">End of life</td><td>Clearance from operational orbit</td><td rowspan="3">ΔV_{EOL}</td><td rowspan="3">$Prop_{EOL}$</td><td rowspan="2">None</td><td rowspan="3">I_{sp} at EOL</td></tr>
<tr><td>Uncontrolled re-entry</td></tr>
<tr><td>Controlled re-entry</td><td>15% only the final burn ΔV</td></tr>
<tr><td rowspan="2"></td><td>Propellant Residuals</td><td>N/A</td><td>$Prop_{RES}$</td><td>1% of max tank load</td><td>N/A</td></tr>
<tr><td>Propellant uncertainties and gauging</td><td>N/A</td><td>$Prop_{UNC}$</td><td>2% of max tank load</td><td>N/A</td></tr>
<tr><td>TOTAL</td><td></td><td></td><td></td><td></td><td></td></tr>
</table>

Table 2 Summary table for DV and propellant budget

ANNEX A - CALCULATION EXAMPLE

As an example, TEST mission is proposed to illustrate the delta-V and propellant calculation. It is highlighted that the numbers provided in this example may not be accurate.

Delta-v computation

TEST mission hypothesis

The TEST satellite would fly in trailing formation, between 60 to 80 seconds behind Sentinel-2, following the same ground-track. The orbit parameters are defined by:

TEST reference orbit definition	
Repeat cycle	10 days
Cycle Length	143 Orbits
Inclination	Sun-synchronous
MLST at descending node	S2 MLST + 60 to 80 s
Eccentricity vector	Frozen
Longitude of 1st Ascending node	Same as Sentinel-2

The satellite characteristics are defined by:

TEST characteristics	
Dry mass (including system margins)	730 kg
Cd	3
Cross section area	4.5 m²
Thruster efficiency	0.985
Propellant tank	200 kg

The TEST satellite will be launched in 2029 and has a lifetime of 5 years.
At least one repositioning from one Seintel-2 to another shall be possible during the mission lifetime.
The TEST satellite will need to perform a controlled re-entry at end of life.

Launch injection & Orbit Acquisition

It is assumed that:

- The SRD requires to correct launcher injection errors at a 3-sigma levels
- In order to acquire the trailing position with respect to Sentinel-2 in the allocated time, the TEST satellite will be launched 10km below its nominal orbit.
- The target inclination will be the nominal sun-synchronous inclination of the final orbit, therefore no inclination manoeuvre is needed for the orbit acquisition.

Considering a 15 km 3-sigma semi-major axis error on the launcher and a targeted semi-major axis 10km lower than nominal, the maximum correction of the semi-major axis would be 25 km. The total delta-v for the injection error and the orbit acquisition with the assumed 3-sigma launcher accuracy would therefore be:

TEST 3-σ injection accuracy + orbit acquistion			
Semi-major axis (error + acquisition)	15+10 km	13 m/s	13.0 m/s*
Eccentricity vector	0.0025	9.3 m/s	
Inclination	0.15 deg		19.5 m/s
MLST	5 sec		---**
TOTAL			**32.5 m/s**

*Only the maximum of the eccentricity vector correction or the total semi-major axis correction (error + orbit acquisition) shall go to the delta-v budget.
**The error on the MLST is smaller than the 20 seconds tolerance, therefore it does not need to be corrected.

Orbit transfer manoeuvres

At least one transfer from one Sentinel-2 to another shall be possible. It is assumed that:
- The transfer shall be possible in less than 60 days.
- The maximum drifting phase is 180 degrees. Only 50 days are available for that drift, as the transfer phase may start and end with an out of plane manoeuvre, and some margin for the last adjustments is considered. The orbit needs to be lowered/raised by 3.3 km in semi-major axis to start the drift followed by the raising/lowering to come back to the nominal semi-major axis.
- An additional drift of 20 seconds may be needed to account for the fact that the Sentinel-2 satellites may not be in the same orbital plane. The whole 60 days are available for that 20 seconds MLST drift, that is achieved by off-setting the inclination 0.0125 deg, followed by a manoeuvre to come back to the nominal inclination.

TEST transfer		
Semi-major axis	2 x 3.3 km	3.4 m/s
Inclination	2 x 12.5 mdeg	1.6 m/s
TOTAL		**5 m/s**

Orbit transfer manoeuvres

Sentinel-2 corrects its inclination in order to keep its ground-track dead-band at high latitudes, therefore TEST satellite will have to correct the inclination too. The inclination drift is -0.038 deg/year, so if the mission lifetime is 5 years, the total inclination to be corrected would be 0.19 deg.

TEST out of plane manoeuvres		
Inclination	0.19 deg	**24.7 m/s**

For the in-plane manoeuvres 5 years of mission shall be computed considering the nominal launch year (2029) and up to the worst case of a 5-year delay launch. In the TEST case, the worst case corresponds to a launch delay of 4 years.

TEST in-plane manoeuvers										
Year	2029	2030	2031	2032	2033	2034	2035	2036	2037	2038
Delta-V	0.107	0.089	0.103	0.335	1.020	1.747	1.644	1.241	0.631	0.283
TOTAL					**6.3 m/s**					

Orbit transfer manoeuvres

It is assumed that no additional delta-v is required for the formation flying maintenance as the analyses performed show that the TEST satellite will always remain in the assigned control box (i.e. 60 to 80 seconds apart from Sentinel-2)

Orbit transfer manoeuvres

It is assumed that as result of the DRAMA analysis, the number of CAMs during the mission lifetime is 11. A 100% margin on the number of manoeuvres is required so the delta-v will be computed on 22 manoeuvres. For each of them a change of 200 meters in altitude followed by another change of 200 to come back to the nominal altitude shall be considered. That is for each CAM the total variation of the semi-major axis will be 200 meters.

TEST CAMs		
Semi-major axis	11 x 0.2 km	1.2 m/s
TOTAL with 100% margin		**2.4 m/s**

Attitude control based on thrusters

It will be assumed the satellite attitude is controlled only by reaction wheels and magnetorquers, even in safe mode.

Controlled re-entry

The first phase of the controlled re-entry is to lower the perigee to the lowest altitude the satellite can still maintain its attitude within the required boundaries. It is assumed that this altitude is 250 km.
For this phase the manoeuvre plan shall be a trade-off between the gravity losses resulting from the need long of manoeuvres and the numbers of manoeuvres done. In this example a

strategy of lowering the perigee by 105 km on each manoeuvre is assumed. For simplicity a constant 200s Isp and a linear thrust reduction of 1 N per manoeuvres are assumed.

Under these assumptions the computed gravity losses due to the long duration of the manoeuvres are in the order of 1.8% to 2.5% when compared with impulsive manoeuvres.

TEST de-orbit manoeuvres									
Burn	thrust	Duration	Init Apo	Init Per	Init Mass	Final apo	Final Per	Final Mass	DeltaV
1	27 N	877 s	796.5	778	825	794.5	670	813	28.9
2	26 N	890 s	794.5	670	813	792.5	565	801	28.7
3	25 N	930 s	792.5	565	801	790	460	789	29.3
4	24 N	975 s	790	460	789	787.5	355	777	29.8
5	23 N	1023 s	787.5	355	777	785	250	765	30.5
TOTAL									**147.2 m/s**

The second phase of the controlled de-orbit is the last burn. The perigee needs to be lowered from 250km to 50km in a single manoeuvre. At EOL it is considered:
- Total thrust EOL is 22N
- Satellite dry mass is 730 kg
- Isp 200 s.

When considering the manoeuvre is not impulsive, the computed gravity loss due to the long duration is nearly 13% with respect to an impulsive manoeuvre.

TEST Last burn			
Thrust	22 N		
Isp	200 s		
Initial Apo	785 km	**Final Apo**	757 km
Initial Per	250 km	**Final Per**	50 km
Initial mass	765 kg	**Final mass**	736 kg
Duration	2254 s		
Delta V	65.9 m/s		
Total delta V (15% margin)		**75.8 m/s**	

Propellant computation

The computation starts with the propellant required to perform the de-orbiting manoeuvre.

$$Prop_{EOL} = \left(m_{dry} + Prop_{RES} + Prop_{UNC}\right)\left(e^{\frac{\Delta V_{EOL}}{V_{e_{EOL}}}} - 1\right)$$

Considering a $I_{SP_{EOL}}$ = 200 s, a thruster efficiency of 0.985, and starting from the initial mass of 736 kg, the propellant mass needed for EOL operations is equal to 90 kg.

The next step is to compute the propellant needed for the mission operations.

$$Prop_{OP} = \left(m_{dry} + Prop_{RES} + Prop_{UNC} + Prop_{EOL}\right)\left(e^{\frac{\Delta V_{OP}}{V_{e\_OP}}} - 1\right)$$

In this case considering a $I_{SP_{OP}}$= 210 s, a thruster efficiency of 0.985, and starting from the initial mass of 826 kg the propellant needed results to be equal to 16.6 kg.

The final step is to compute the propellant needed for the beginning of life operations.

$$Prop_{BOL} = \left(m_{dry} + Prop_{RES} + Prop_{UNC} + Prop_{EOL} + Prop_{OP}\right)\left(e^{\frac{\Delta V_{BOL}}{V_{e\_BOL}}} - 1\right)$$

In this case considering a $I_{SP_{BOL}}$= 220 s, a thruster efficiency of 0.985, and starting from the initial mass of 842.6 kg the propellant needed results to be equal to 14.9 kg.

Phase	ΔV contribution	Total ΔV [m/s]	Propellant [kg]	Margins	I_{sp} [s]
Beginning of life	Launcher injection errors correction	*32.5*	*14.9*	*None*	220
	Orbit acquisition				
Mission operations	Orbit transfer manoeuvres	*38.4*	*16.6*	*None*	210
	In-plane and out of plane orbit control manoeuvres				
	Constellation/Formation Maintenance				
	Collision Avoidance Manoeuvres (CAM)			*100% on the number of CAM*	
FDIR	Attitude control based on thrusters	*0*	*0*	*100% on propellant*	
End of life	~~Clearance from operational orbit~~	*N/A*	*N/A*	*None*	200
	~~Uncontrolled re-entry~~	*N/A*	*N/A*		
	Controlled re-entry	*223*	*90*	*15% on the final burn only*	
	Propellant Residuals	*N/A*	*2*	*1% of max tank load*	N/A
	Propellant uncertainties and gauging	*N/A*	*4*	*2% of max tank load*	N/A
TOTAL		**293.9 m/s**	**127.3 kg**		

ANNEX B – EXAMPLES OF DRAG COEFFICIENTS FROM PAST MISSIONS

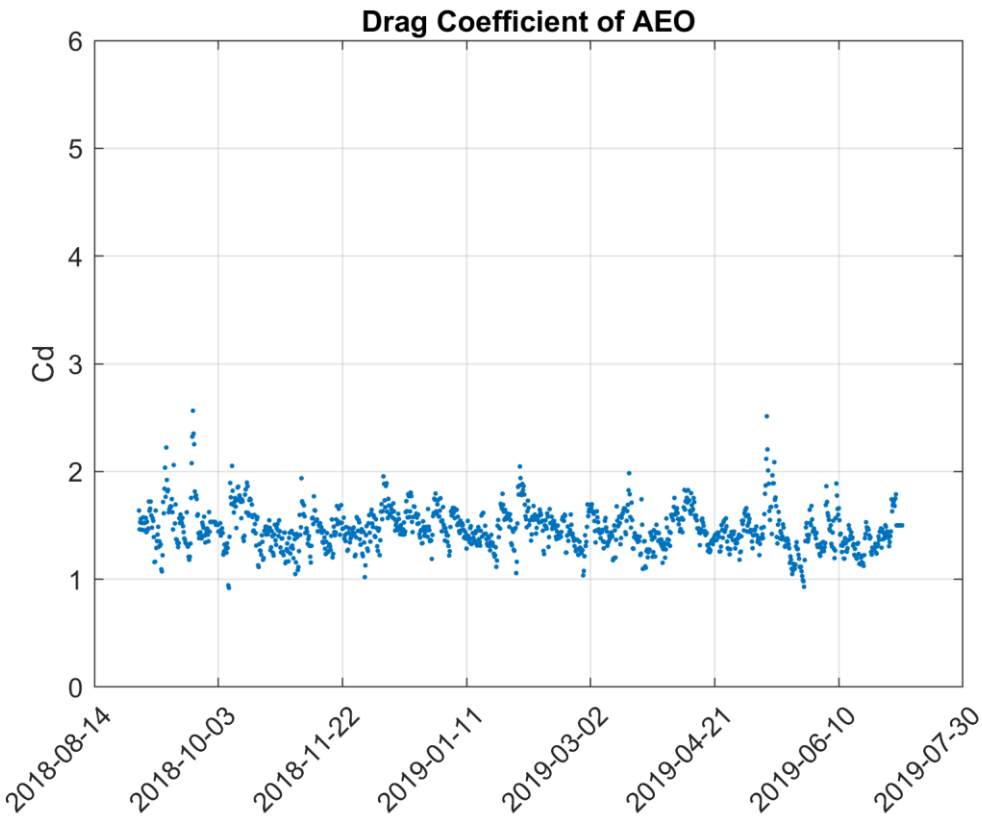

Figure 1 Drag Coefficient of Aeolus

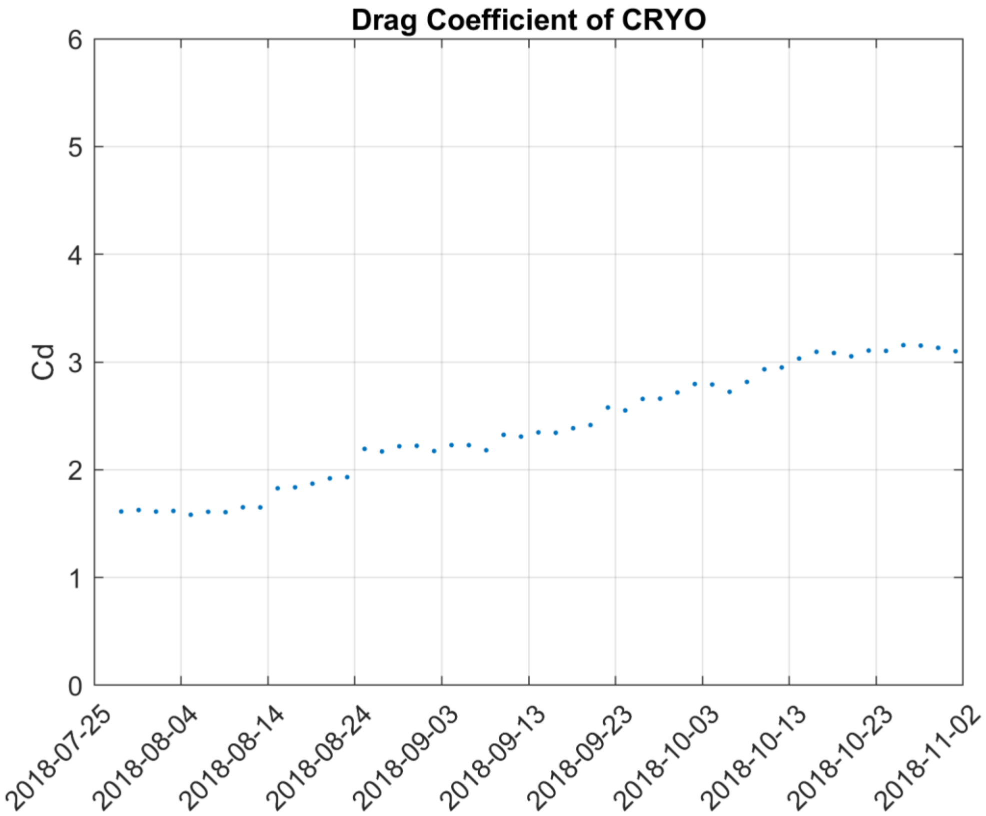

Figure 2 Drag Coefficient of Cryosat-2

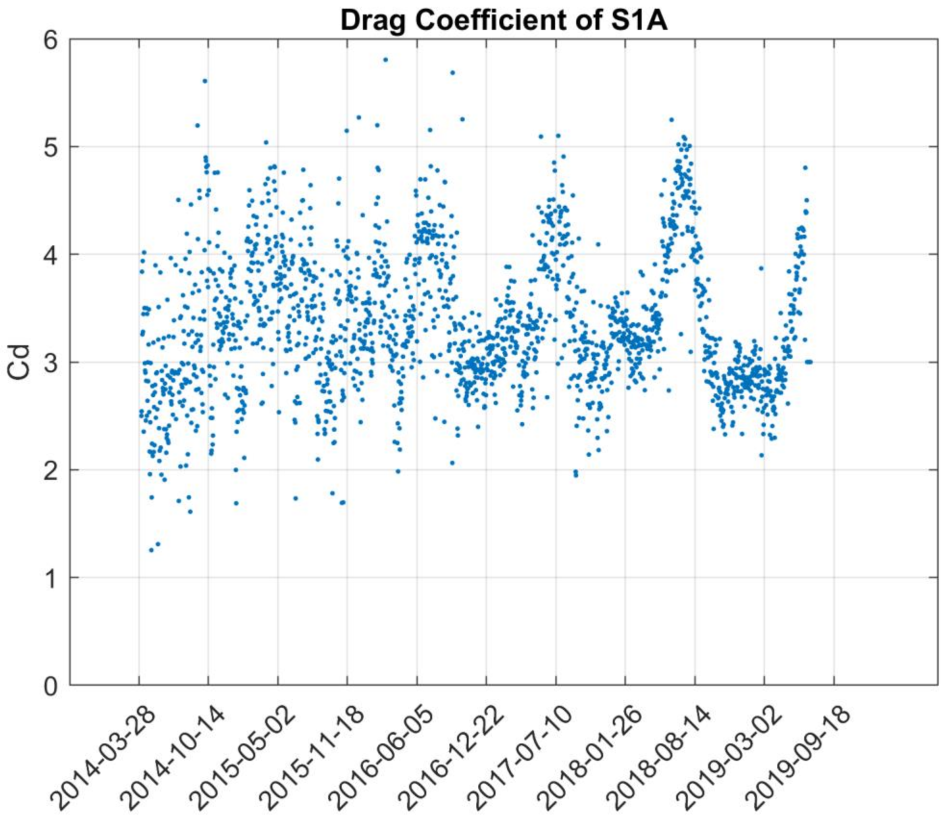

Figure 3 Drag Coefficient of Sentinel-1A

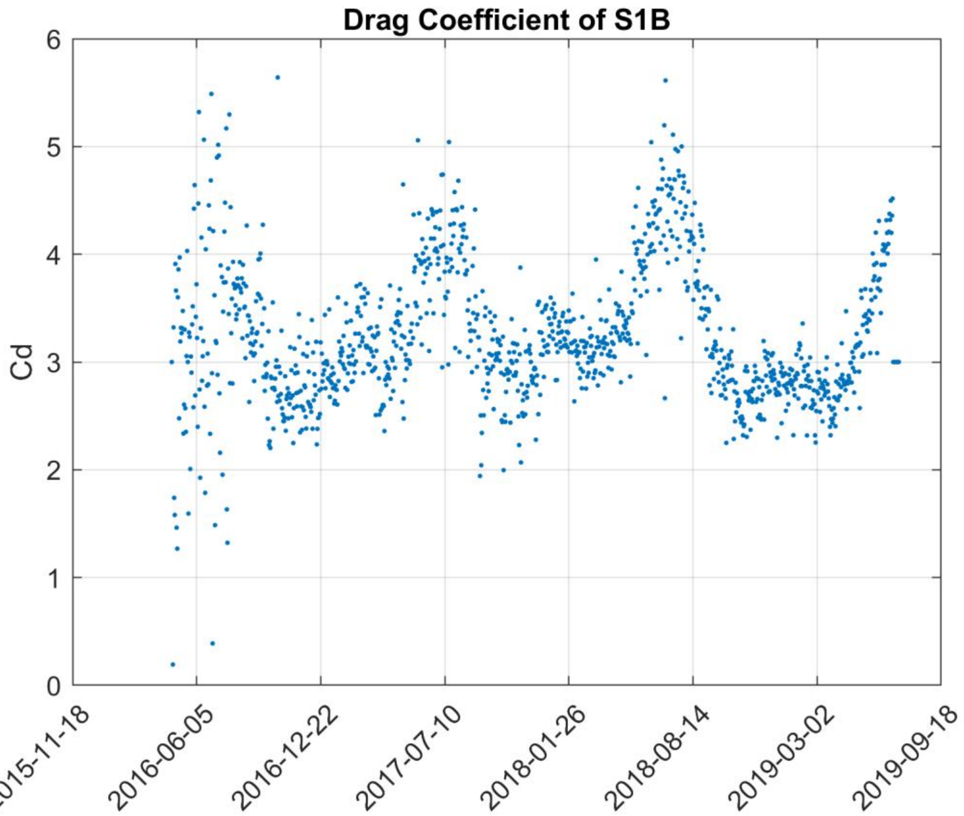

Figure 4 Drag Coefficient of Sentinel-1B

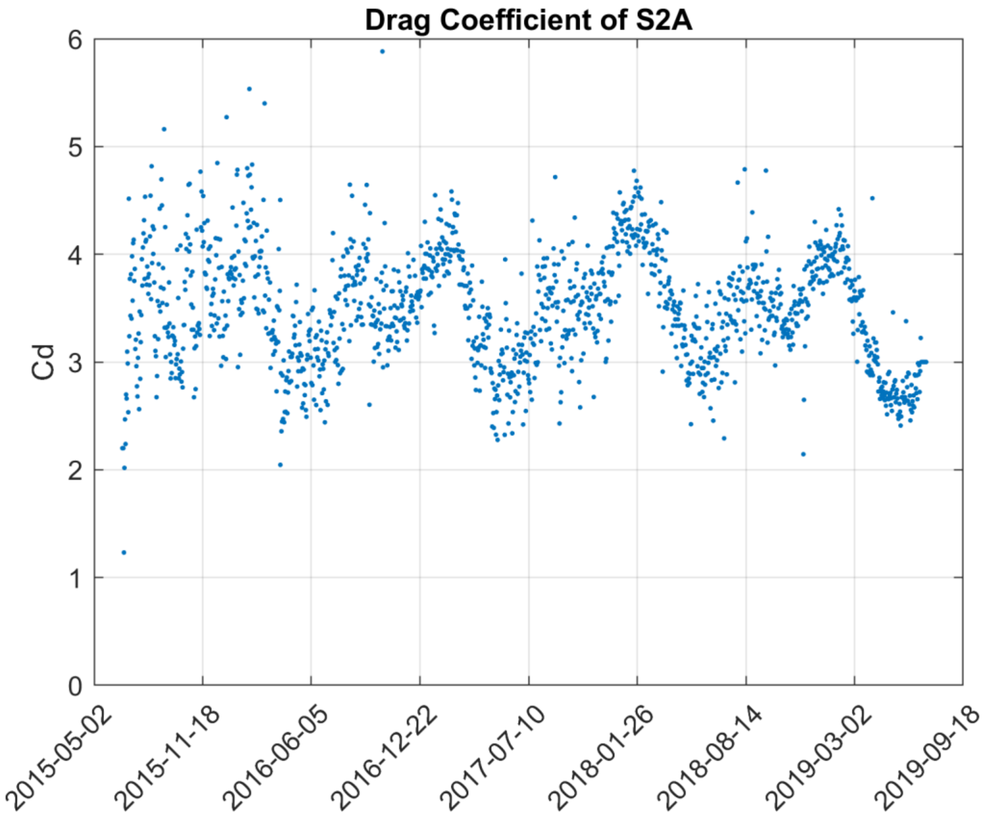

Figure 5 Drag Coefficient of Sentinel-2A

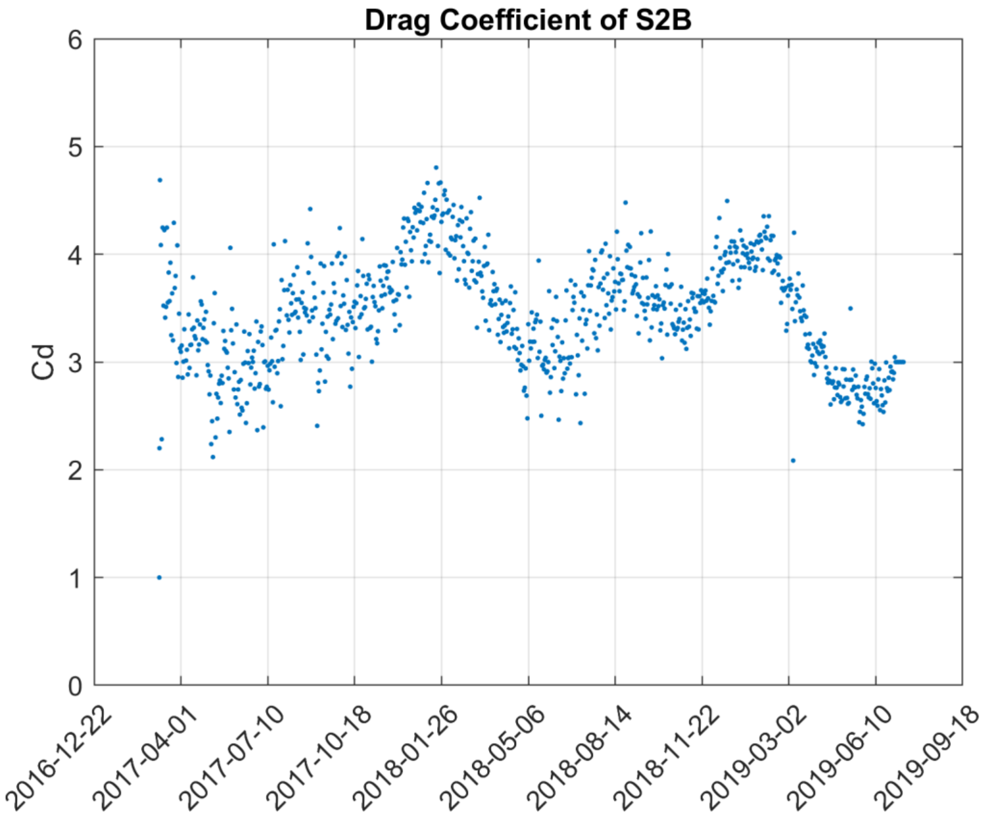

Figure 6 Drag Coefficient of Sentinel-2B

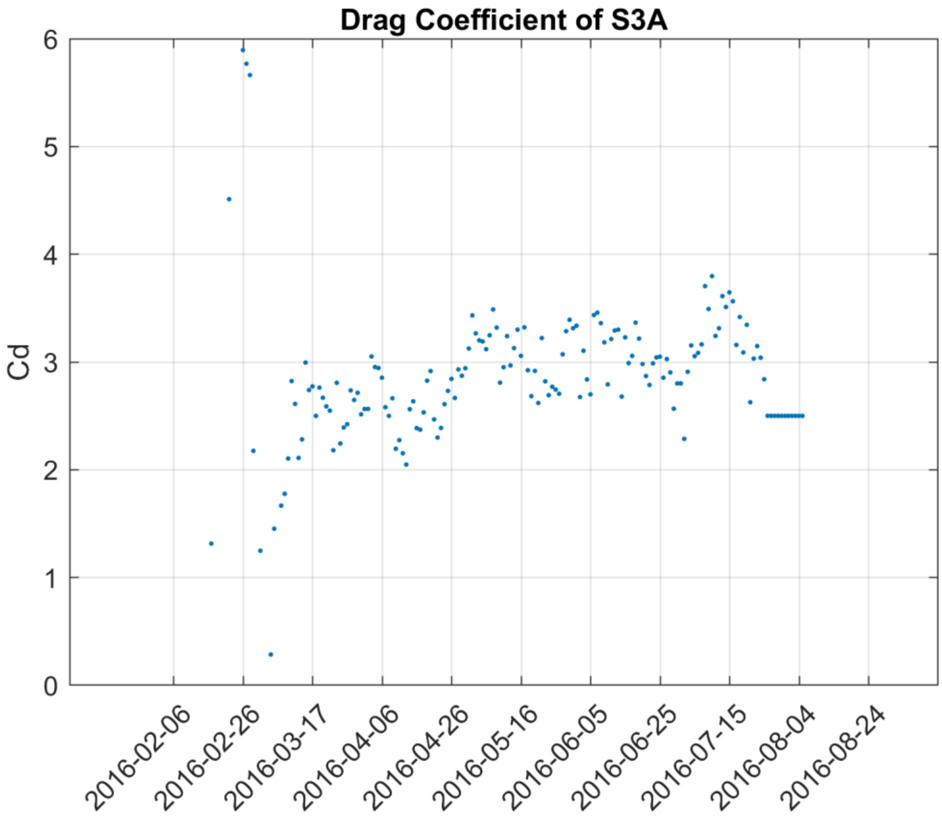

Figure 7 Drag Coefficient of Sentinel-3A

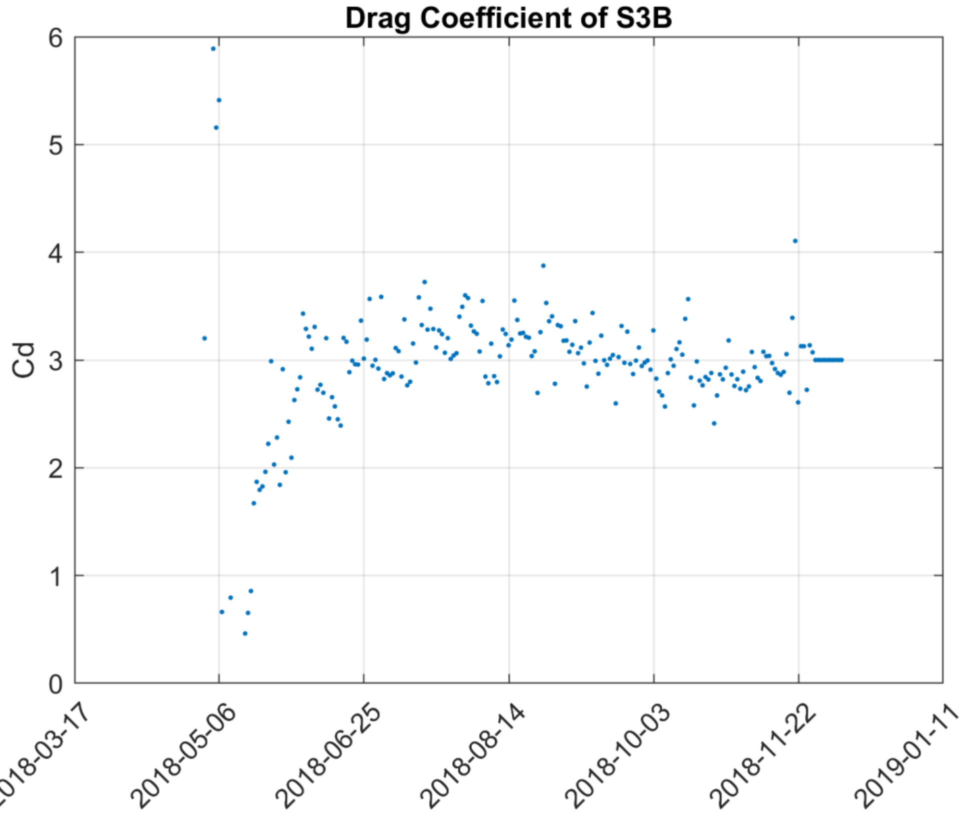

Figure 8 Drag Coefficient of Sentinel-3B

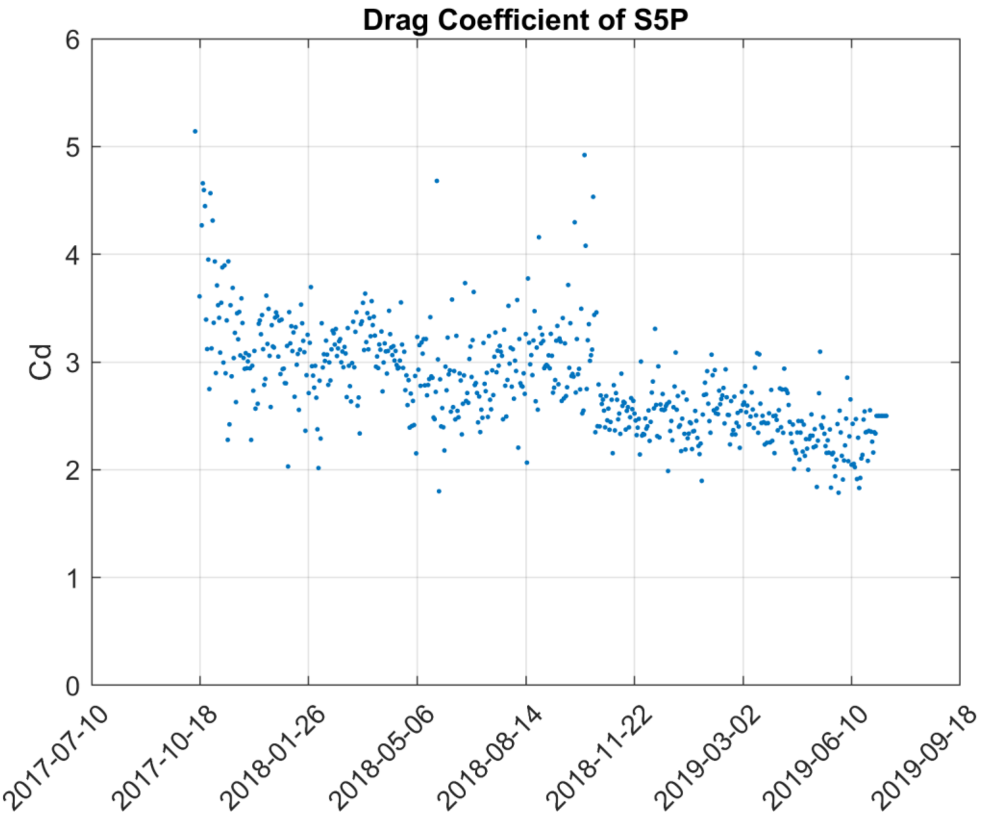

Figure 9 Drag Coefficient of Sentinel-5P

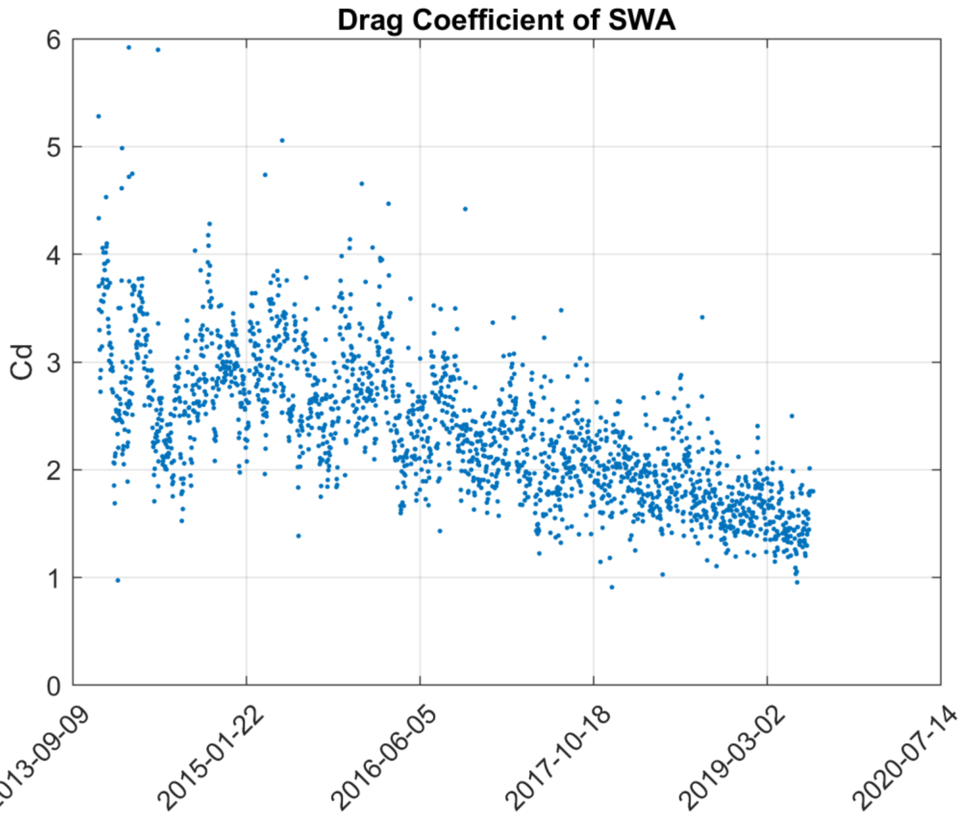

Figure 10 Drag Coefficient of Swarm-A

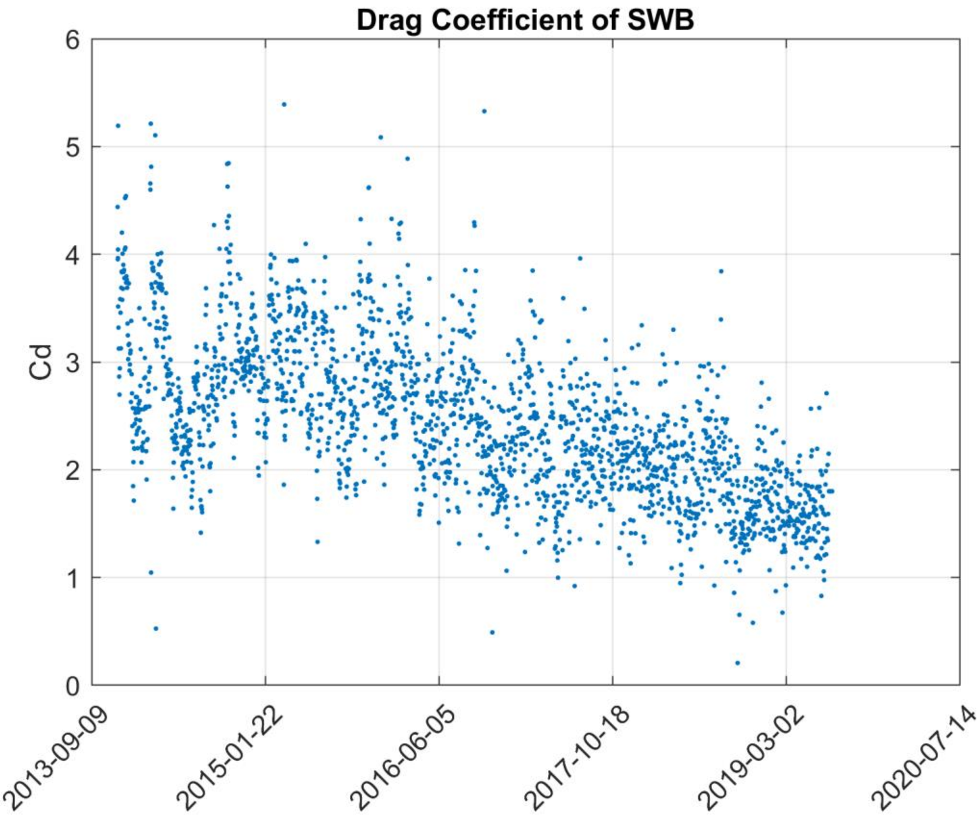

Figure 11 Drag Coefficient of Swarm-B

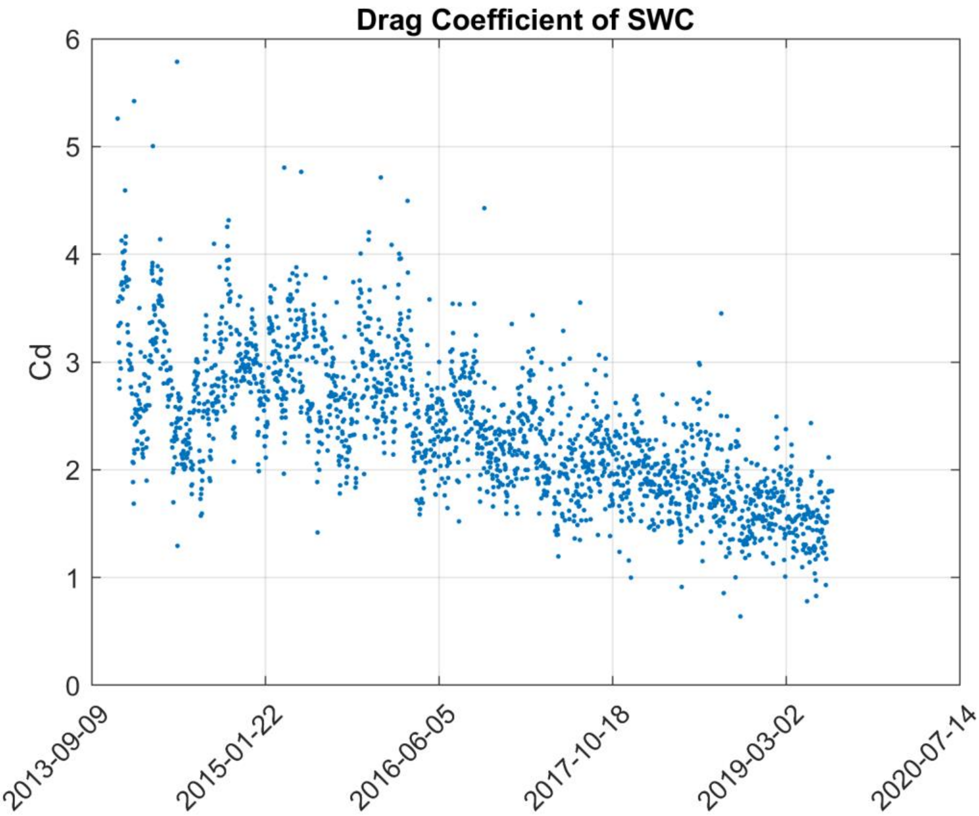

Figure 12 Drag Coefficient of Swarm-C