MISSION DESIGN AND NAVIGATION ANALYSIS OF THE ESA EXOMARS PROGRAM

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Abstract: *This paper presents the mission design and the navigation analysis for the 2016 and 2018 missions of the ExoMars program. Both missions are conducted jointly by the European Space Agency ESA and the Russian Space Agency Roscosmos, in order to investigate the Martian environment and to demonstrate new technologies for the future Mars sample return mission. The significantly different mission objectives are reflected in equally significant differences in the mission design.*

Keywords: *ExoMars, Mission Design, Navigation, Planetary Protection.*

1. Introduction

European Space Agency (ESA) has established the ExoMars program to investigate the Martian environment and to demonstrate new technology for a future Mars exploration. Two missions, the 2016 and 2018 mission, are pursued as part of a broad joint undertaking between ESA and the Russian Space Agency (Roscosmos).

The ExoMars 2016 mission comprises a large Mars orbiter and a landing craft. The orbiter is called "Trace Gas Orbiter" (TGO) and carries scientific instrumentation for the detection of trace gases in the Martian atmosphere and for other atmospheric and surface science. The landing craft is called "Entry, Descent and Landing Demonstrator Module" (EDM). Its mission is to demonstrate and evaluate European Mars landing technologies.

The spacecraft composite will be launched in January 2016 by a Russian-provided Proton M/Breeze M launcher and will arrive at Mars approximately 9 months later in mid-October of 2016. Prior to arrival at Mars, the EDM will be released from the TGO and will enter the Mars atmosphere from a hyperbolic arrival trajectory. Following separation of the EDM, the TGO will go into orbit around Mars to perform a primary science phase of one Martian year followed by a Mars proximity communications phase of about 4 years. The Mars proximity communications will support the 2018 Rovers mission as well as any other international assets on the surface of Mars.

The configuration of the EDM will be developed keeping in mind the scalability to future larger landers. Engineering sensors will be incorporated into the design to assess the performance of the system throughout its Entry, Descent and Landing (EDL) phase. The EDM will have a heat shield diameter of about 2.4 m and will support the Mars atmospheric entry from incoming hyperbolic trajectory. The system will be designed to survive the possibility of a severe dust storm since it will arrive at a period of high probability of encountering a Mars global dust storm. After entry the system will deploy a single stage disk gap band parachute and will complete its landing by using a closed-loop Guidance, Navigation and Control (GNC) system based on a radar Doppler altimeter sensor and on-board inertial measurement units that will guide a liquid propulsion system by the actuation of thrusters to be operated in pulsed on-off mode.

The EDM is expected to survive on the surface of Mars for a short time (about 8 sols) by using the excess energy capacity of its primary batteries. A set of scientific sensors will be embarked as a demonstration of surface science within the mass and electrical (including radio-frequency) resources available in the EDM without adding additional systems for solar power generation or for thermal control, such as radioisotope heater units.

The 2018 mission will deliver a surface platform and a mobile rover to the Martian surface. The rover's task is to perform surface science including the search for traces of extant or former life. The rover and the surface platform will be delivered with a Carrier Module (CM) and a Descent Module (DM) to a Mars surface. The CM will be provided by ESA with some contributions from Roscosmos and the DM will be provided by Roscosmos with some contributions by ESA.

The spacecraft composite of the 2018 mission stacks at the time of the launch consists of the DM and the CM. Inside the DM are a static surface platform and rover which will be deployed on the Mars surface. The surface mission duration shall be at least 180 sols for the rover and one Mars year for the static surface station. The CM carries the DM to Mars, performs fine targeting and attitude operations and is jettisoned shortly before atmospheric entry. The CM is not foreseen to operate after separation but care must be taken to avoid any form of re-contact between CM and DM after separation.

In this paper, the mission design and the navigation analysis for ExoMars 2016 and 2018 mission are presented in parallel. The difference and the relationship of those missions are clearly explained.

2. Mission Design

2.1. ExoMars 2016 mission

The design of the 2016 mission is optimized to deliver the TGO with a defined dry mass into a low circular Mars orbit with an inclination of 74 deg. The TGO will carry the EDM on the interplanetary transfer and deploy it during the Mars approach for a landing in Meridiani Planum (1.82°S, 6.15°W). The spacecraft composite will be launched and placed into an Earth escape trajectory by a Proton M/Breeze M launcher provided by Russia. A large Deep Space Maneuver (DSM) is performed during the 9-month transfer.

The release of the EDM will take place about 3 days prior to the critical Mars Orbit Insertion (MOI) maneuver by the ExoMars TGO. The sequence of maneuvers following the separation will be designed to maximize the chance of receiving the UHF radio beacon signals from the EDM during its EDL. Subsequent passes of the TGO over the landing site of the EDM are not foreseen in the current baseline. Other spacecraft in orbit around Mars may serve as data relay to provide the capability of additional passes over the landing site for EDL data up-load sessions.

Table 1. Mission sequence of ExoMars 2016

Mission Phase	Epoch
Launch	2016/01/07 - 01/27
Deep Space Maneuver	Launch + $\approx$ 130 days
Trajectory Correction Maneuver	EDM entry - 30, 5 days
EDM Separation	EDM entry - 3 days
TGO Orbit Retargeting Maneuver	EDM entry - 2.5 days
Mars Orbit Insertion	2016/10/19
EDM Mars Atmospheric Entry	2016/10/19
TGO Inclination Change Maneuver	MOI + 6 days
Apoapsis Lowering Maneuver	MOI + 8 days
Aerobraking	$\approx$ 340 days
Science Phase	1 Martian year
Data Relay Phase	

After the capture, the TGO will be in a 4 sols elliptical orbit around Mars. Subsequently, the TGO will begin a series of maneuvers to change the orbit inclination to 74 degrees and reduce the apoares using on-board fuel reserves, down to a 1 sol orbit. Further reductions of the apoares will be performed using aerobraking followed by a final circularization maneuver to arrive at the science and communications orbit with an altitude in the range of 350 km to 420 km [1]. The science operations phase is expected to begin at the earliest in June of 2017 (depending on the actual duration of the aerobraking phase) and last for a period of one Martian year. The science instruments on-board will remotely sense the presence, quantity and potential sources of methane in the Martian atmosphere. Near the end of the science operations phase, the rover of the 2018 mission should arrive at Mars (January 2019) so that the emphasis on the TGO operations may shift to provide orbital science as well as Mars proximity data relay function for the rover. The TGO will be designed for consumables that will allow further Mars proximity data relay support and science operations until the end of 2022. The mission sequence for ExoMars 2016 mission is summarized in Tab. 1.

The 2016 mission launch period optimization takes into account the entire series of maneuvers from launcher separation until the start of the aerobraking. The transfer resulting from the launch on each date of the 21-day launch period is optimized individually; the launch period is placed such that the available delivered mass is maximized for the worst case.

The Type II transfer of the ExoMars 2016 baseline mission incorporates a large DSM in the way to Mars to reduce the arrival velocity at Mars which also serves to reduce the magnitude of the MOI, thus reducing the gravity losses during the long MOI burn ($\approx$ 2 hours). The MOI is optimized to maximize the mass delivered into the desired capture orbit as well as decreasing the entry velocities and heat loading for the EDM. The design of a launch period based on such a transfer has to take into account that the intermediate DSM provides a higher flexibility to select the Earth escape velocity than a direct transfer, in which typically for given departure and arrival dates the hyperbolic velocities at both planets are prescribed. In order to ease the mission planning and the operations it has been decided to fix the arrival date to Mars across the launch period. This has a minor effect on the ΔV budget that can be accommodated without problems within the propellant budget allocated for the mission.

The launch period optimization process has been performed with a fixed S/C separated mass constrained to 4332 kg. As result of this optimization process the optimum 21-days launch period, hyperbolic velocity at Earth escape and Mars arrival date have been identified. The following is a summary of the assumptions considered for the computation of the launch period:

- The launch period is 21 days long.
- The launch period based on Earth-Mars Type II transfer in the 2016 opportunity with fixed Mars arrival date.
- The composite mass after separation from the launcher is fixed to 4332 kg across the launch period. This leaves significant launcher margins with respect to the Proton M/Breeze M performance for the chosen escape velocity
- 600 kg EDM released 3 days before reaching the Entry Interface Point (EIP)
- The Orbit Re-targeting Maneuver (ORM) simultaneously targets the pre-MOI periares altitude of the TGO trajectory and also advances the periares time in order to optimize conditions for EDM relay. This maneuver is simulated as part of the trajectory optimization.
- Orbiter inserted into an approximately 4-sols Mars orbit
- MOI numerically computed with thrust direction against the velocity vector, and a target osculating periares altitude at the end of the burn of 250 km. The start time of MOI is variable in order to allow an asymmetric MOI burn and some degree of adjustment of the post-MOI line of apsides.
- The optimization goal is to maximize the mass of the orbiter after the ICM which follows approximately 6 sols after completion of MOI.

Table 2 provides the launch period characteristics as a function of the launch date. The modulus of $\boldsymbol{V}_{inf}$, declination (DEC) and right ascension (RA) are provided for both departure and arrival condition in EME2000. The ΔVs of DSM, MOI and ICM are summarized in Fig. 1. The ΔV shown in the figure are the deterministic values obtained in the transfer computation without any additional margin. The cost function in the optimization process is the TGO mass after the ICM. The variation of DSM, MOI and ICM ΔV and the Mars arrival velocity is smooth across the launch period as can be seen in Fig. 1. The DSM ΔV presents a minimum at launch day 17, while the arrival velocity and the MOI ΔV increase with the launch day. On the other hand, the ICM ΔV decreases with the launch day.

The interplanetary trajectory of ExoMars 2016 mission is shown in Fig. 2. In the figure, both Launch Period Open (LPO) and Launch Peroid Close (LPC) trajectories are presented, however the paths are nearly identical.

2.2. ExoMars 2018 mission

Compared to the 2016 mission, the 2018 interplanetary transfer is simpler. No major maneuver is required during the interplanetary cruise. Deployment of the descent module takes place from

Table 2. ExoMars 2016 Launch period characteristics as a function of the launch day

		Earth departure			Mars arrival		
Day	Date	Vinf [km/s]	DEC [deg]	RA [deg]	Vinf [km/s]	DEC [deg]	RA [deg]
1	2016-01-07	2.73	3.41	241.12	3.22	-24.39	200.21
2	2016-01-08	2.73	3.67	241.57	3.24	-24.74	200.24
3	2016-01-09	2.73	3.97	241.99	3.25	-25.08	200.28
4	2016-01-10	2.73	4.30	242.38	3.26	-25.43	200.33
5	2016-01-11	2.73	4.65	242.74	3.27	-25.78	200.39
6	2016-01-12	2.73	5.04	243.07	3.29	-26.12	200.46
7	2016-01-13	2.73	5.45	243.37	3.30	-26.47	200.54
8	2016-01-14	2.73	5.89	243.64	3.31	-26.82	200.65
9	2016-01-15	2.73	6.34	243.88	3.32	-27.16	200.77
10	2016-01-16	2.73	6.82	244.10	3.33	-27.49	200.91
11	2016-01-17	2.73	7.31	244.30	3.35	-27.83	201.06
12	2016-01-18	2.73	7.83	244.49	3.36	-28.16	201.23
13	2016-01-19	2.73	8.36	244.65	3.37	-28.49	201.41
14	2016-01-20	2.73	8.90	244.80	3.38	-28.82	201.60
15	2016-01-21	2.73	9.47	244.94	3.39	-29.14	201.81
16	2016-01-22	2.73	10.06	245.06	3.40	-29.47	202.02
17	2016-01-23	2.73	10.67	245.17	3.41	-29.80	202.25
18	2016-01-24	2.73	11.30	245.25	3.43	-30.12	202.48
19	2016-01-25	2.73	11.96	245.31	3.44	-30.44	202.71
20	2016-01-26	2.73	12.63	245.35	3.45	-30.76	202.96
21	2016-01-27	2.73	13.32	245.35	3.46	-31.07	203.20

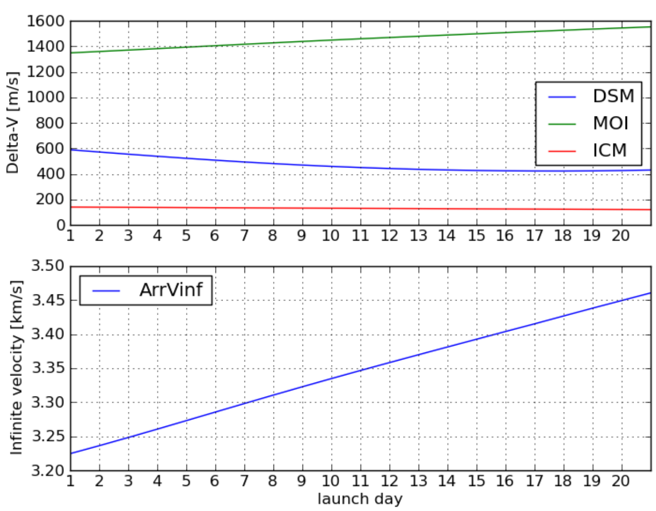


Figure 1. ExoMars 2016 DSM, MOI, ICM and Mars hyperbolic velocity as function of launch day

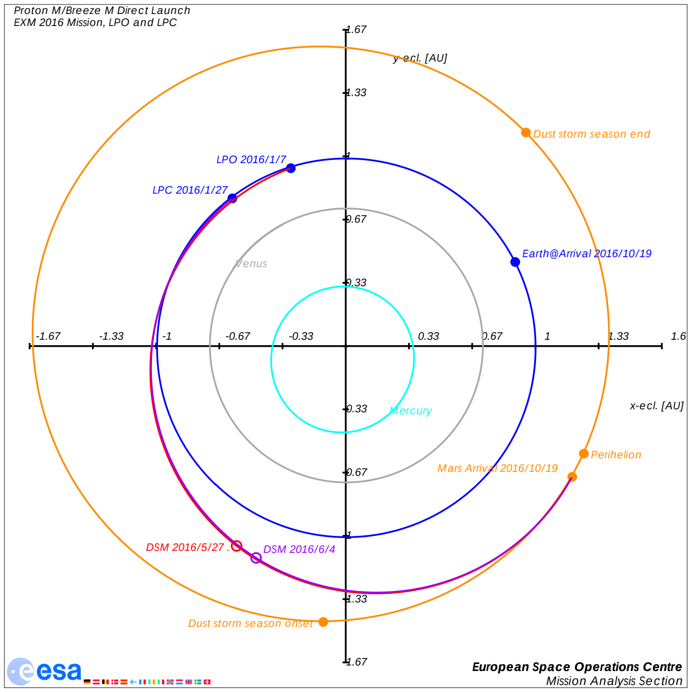


Figure 2. ExoMars 2016 Interplanetary Trajectory (Ecliptic)

hyperbolic approach. The launch period also spans 21 days and the delivery mass is also maximized. The candidates of landing sites are summarized in Fig. 3 together with the landing sites of historical Mars rovers and landers. The candidates of the landing sites are located within a latitude range of 25 deg North to 5 deg South [2]. The selection of the landing site has minor impact for the interplanetary mission design, however there is major impact for the navigation and the EDL of the DM.

A global dust storm is considered carefully for the interplanetary mission design since it has a large effect to the system design of the DM. Although the mechanisms leading to a global dust storm are not yet fully understood, it is clear that thermal effects play a major role. To gain a better understanding of past observations, a statistical analysis had been performed in the course of the ESA ExoMars project[3]. The results are expressed as a function of the solar longitude L_S. Considering the observations of the Mars dust storm made by spacecrafts and Earth-based observatories, the zone from L_S = 180 deg to L_S = 324 deg is excluded from the arrival date of the ExoMars 2018 mission. While an L_S value of 324 deg is assumed as safe for surface operations, there appears to remain some scope for a further mitigation of the likelihood of encountering a global dust storm, by delaying landing further, to 340 or even 350 or 355 deg.

The most common type of transfer for Mars missions is the short transfer, where the trajectory roughly resembles a half-ellipse. The obtained transfer for the 2018 opportunity departs Earth in May 2018 and arrives at Mars on 2019/1/15, fulfilling the constraint on the arrival L_S that states that it arrival shall not be in the global dust storm season. All trajectory design is subject to the following assumptions:

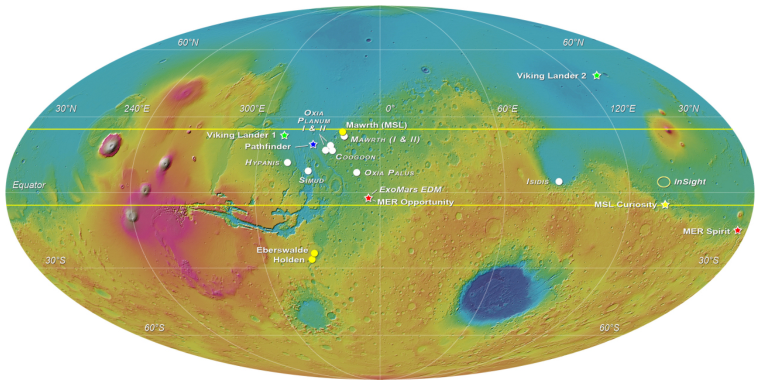


Figure 3. Candidate landing sites for the ExoMars program (Credit : ESA-Roscosmos/LSSWG/E. Hauber)

- Consistency with a Proton M / Breeze M launch
- Application of a launch period that spans 21 consecutive days
- Absence of deep space maneuver (DSM).

The 21-day launch period and associated mission characteristics are listed in Tab. 3. The hyperbolic escape velocity does not exceed 2.94 km/s. The hyperbolic arrival velocity depends on the launch date but the difference between the maximum and the minimum is small. The reference frame for the escape and arrival right ascension and declination is the EME2000 inertial frame. The interplanetary trajectory of LPO and LPC are described in Fig. 4.

3. Navigation analysis

Navigation analyses are performed for both the ExoMars 2016 and 2018 missions. The main challenge of the interplanetary navigation is to deliver entry craft (2016: EDM, 2018: descent module) to the defined atmospheric entry interface points sufficiently accurately to ensure that landing on the surface takes place within the required uncertainty ellipse. The driving parameter is the entry corridor, expressed via the entry flight path angle dispersion. Another important output of the navigation analysis is the expected size of Trajectory Correction Maneuvers (TCMs). 99-percentile of the TCM ΔV magnitude is evaluated for all maneuvers.

The results of this navigation analysis are based on covariance analysis. In this study, the ESOC in-house software INTNAV is used to analyze the interplanetary navigation. The orbit determination processing is simulated with a Square Root Information Filter (SRIF) incorporating measurements. For each TCM, the orbit determination covariance matrices are updated via Monte Carlo simulation assuming a linear guidance law and modeling the maneuver execution errors.

3.1. ExoMars 2016 mission

The navigation analysis for ExoMars 2016 mission is performed to investigate the EDM delivery accuracy at the Entry Interface Point (EIP). Since the trajectory of the ExoMars 2016 performs a DSM during its interplanetary cruise, the impact probability of the upper stage of the launcher is

Table 3. ExoMars 2018 Launch period characteristics as a function of the launch day

Day	Earth departure				Mars arrival		
	Date	Vinf [km/s]	DEC [deg]	RA [deg]	Vinf [km/s]	DEC [deg]	RA [deg]
1	2018-05-07	2.94	-12.3	338.0	3.46	-1.6	244.0
2	2018-05-08	2.88	-12.5	335.6	3.45	-1.8	244.3
3	2018-05-09	2.87	-13.0	335.7	3.44	-1.7	244.4
4	2018-05-10	2.85	-13.5	335.4	3.43	-1.5	244.5
5	2018-05-11	2.84	-14.0	335.1	3.42	-1.3	244.5
6	2018-05-12	2.83	-5.0	330.9	3.49	-6.2	246.9
7	2018-05-13	2.82	-6.0	330.6	3.48	-5.8	246.9
8	2018-05-14	2.80	-7.5	330.4	3.46	-5.1	246.7
9	2018-05-15	2.79	-8.8	330.1	3.44	-4.5	246.6
10	2018-05-16	2.78	-9.6	329.6	3.43	-4.2	246.5
11	2018-05-17	2.78	-10.3	329.1	3.42	-4.0	246.6
12	2018-05-18	2.78	-10.8	328.4	3.41	-3.8	246.6
13	2018-05-19	2.78	-11.3	327.8	3.41	-3.6	246.6
14	2018-05-20	2.78	-11.7	327.1	3.40	-3.5	246.7
15	2018-05-21	2.79	-12.1	326.4	3.39	-3.4	246.8
16	2018-05-22	2.80	-12.5	325.8	3.39	-3.3	246.8
17	2018-05-23	2.82	-12.8	325.1	3.39	-3.3	246.9
18	2018-05-24	2.83	-13.2	324.4	3.38	-3.2	247.0
19	2018-05-25	2.85	-13.5	323.7	3.38	-3.1	247.0
20	2018-05-26	2.86	-13.8	323.1	3.38	-3.1	247.1
21	2018-05-27	2.88	-14.0	322.4	3.38	-3.0	247.1

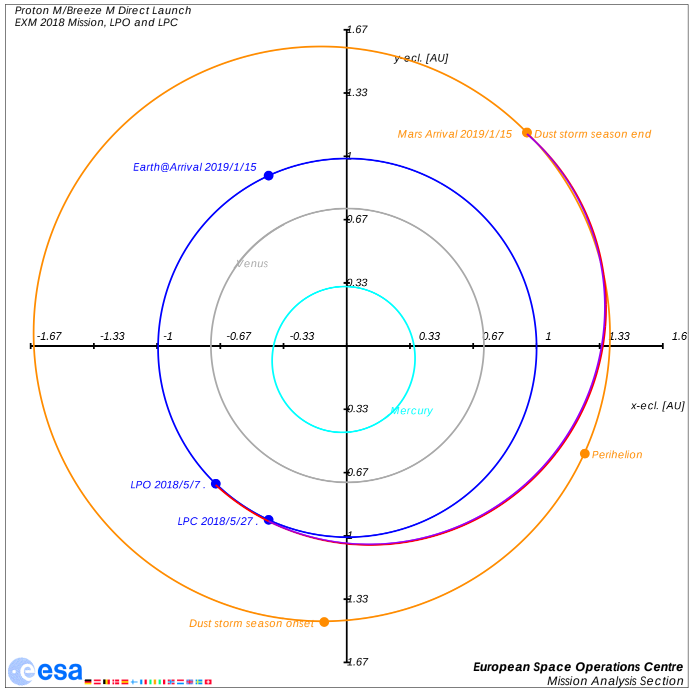


Figure 4. ExoMars 2018 Interplanetary Trajectory (ecliptic)

Table 4. TCM sequence of ExoMars Program

Maneuver	ExoMars 2016	ExoMars 2018
TCM-1 (LIC)	Launch + 7 days	Launch + 7 days
TCM-2	DSM + 14 days	Launch + 30 days
TCM-3	Entry - 30 days	Entry - 30 days
TCM-4	Entry - 5 days	Entry - 8 days
TCM-5	-	Entry - 2 days

less than 10^{-4}. Therefore, the launcher vehicle will target the optimum direction for the reference trajectory. Four TCMs are assumed to be performed during the interplanetary transfer. TCM-1 is planned to be performed at 7 days after the launcher separation in order to remove the launcher dispersion and prepare for the DSM. TCM-2 will perform at 14 days after the DSM to clean-up its execution error, while TCM-3 (Entry - 30 days) and TCM-4 (Entry - 5 days = EDM separation - 2 days) are performed for fine targeting to the EIP. The TCM schedule is summarised in Tab. 4.

3.1.1. Measurements

The assumptions of the measurements are common for both ExoMars 2016 and 2018 mission. Three measurements types (two-way range, Doppler and Delta-DOR) have been considered using the ESA deep space network (MLG: Malargue, CEB: Cebreros, NNO: New Norcia). All measurements are derived from a radiometric link between the ground stations' receiver and the spacecraft.

Table 5. Tracking schedule for ExoMars 2016

Phase	Dates	Range and Doppler	DDOR
post-DSM	DSM to DSM + 15 days	7 pass/week with MLG and CEB	None
Cruise 1	DSM + 15 days to E - 60 days	3 pass/week with MLG	None
Cruise 2	E - 60 days to E - 45 days	7 pass/week with MLG	None
Approach 1	E - 45 days to E - 30 days	7 pass/week with MLG	1 point/week
Approach 2	E-30 days to E - 15 days	7 pass/week with MLG	2 point/week
Approach 3	E-15 days to E - 8 days	7 pass/week with MLG	3 point/week
Approach 4	E-8 days to Entry	7 pass/week with MLG and NNO	4 point/week

Table 6. Measurements assumption

Measurement	error (1 σ)	bias (1 σ)	frequency
Range	4 m	20 m	4 h
Doppler	0.075 mm/s	-	30 min
DDOR	5 cm	-	24 h

For ExoMars 2016 mission, the navigation analysis have been performed from the post-DSM until the EIP. Two cases, LPO and LPC have been regarded. Table 5 shows the tracking schedule used for this navigation analysis. The Delta-DOR schedule is conservative because it considers failures of the measurement operations.

3.1.2. Considered uncertainties

The considered uncertainties of the measurements are summarized in Tab. 6. The provided assumptions are conservative, in particular for the Doppler random error. The 20 m bias in the 2-way range measurements takes into account ranging systems calibration errors and delays due to the Earth troposphere. The Delta-DOR random error assumes ESA Delta-DOR accuracy will improve to NASA level by 2016.

Errors in ground station location take into account errors in the rotational parameters of the Earth. The ground station location error is considered as a bias with respect to the inertial frame. Since Delta-DOR can partially cancel this error by its differencing, different values are considered between range, Doppler and Delta-DOR. Mars ephemeris position uncertainties are also considered as a bias. The errors on the solar radiation pressure model and slewing manoeuvres during the Mars approach are considered in the non-gravitational acceleration (NGA) error as exponentially correlated process noise. Two levels of NGA error have been considered. One is the nominal and the other is the FailOp mode. The FailOp mode value is applied for the last 12 hours before the EDM separation. This assumption implies a balanced system for attitude control. The consider errors are summarized in Tab. 7 and 8.

TCMs are performed in order to meet the EIP conditions. The maneuver execution errors are assumed to have magnitude and direction components. The magnitude error has a fixed and a proportional error. The fixed error is mainly the resolution error and the proportional error is mainly

Table 7. Consider bias

Contents	value (1 sigma)
Ground station location w.r.t. ICRF (RARR, 3 axis)	50 cm
Ground station location w.r.t. ICRF (DDOR, 3 axis)	5 cm
Mars position (3 axis)	500 m

Table 8. Exponentially correlated process noise

Contents	value (1 σ)	Corrected time
Non-gravitational acceleration (Nominal mode)	5.0e-12 km/s^2	1 day
Non-gravitational acceleration (FailOp mode)	1.6e-11 km/s^2	1 day

shutoff error of a thruster. Those errors are defined individually in Table 9. The Data-Cut-Off (DCO) duration is 2 days, which means that all measurements gathered less than 2 days before maneuvers are not taken into account for data processing, verification, command generation and update in the control center.

3.1.3. Navigation results of ExoMars 2016

The dispersion error mapped to the EIP is refered to be as delivery accuracy. Figure 5 shows the evolution of the 3-sigma EFPA uncertainties for both LPO and LPC cases. Since the trajectories are designed to have the same arrival date throughout the launch period, the arrival conditions (the Sun and the Earth geometries) remain similar. Therefore the evolution of the knowledge and dispersion errors shown in Fig. 5 are also similar for both cases. The measurements after the TCM-4 is not used for this analysis, therefore there is no improvement in the knowledge error after then. B-plane dispersion ellipses for LPO and LPC are illustrated in Fig. 6. Each ellipse corresponds to the post TCM-3, TCM-4 and separation dispersion, respectively. It is shown that the EDM separation mechanization error is not negligible. This is because of the long coasting arc (3 days) after the EDM separation. The delivery accuracy of ExoMars 2016 EDM is summarized in Tab. 10. The EDM separation mechanization error is considered. The difference in the EFPA error is caused by the orientation of the B-vector and the B-plane dispersion ellipse. The aiming point of the incoming hyperbola, B-vector, for LPO and LPC varies due to the direction of the hyperbola. The inclination of the incoming hyperbola with respect to the Mars equator differs by about 7 deg, from 26.7 deg for LPO to 33.5 deg for LPC.

Table 9. Maneuver execution errors

TCM uncertainty	value (1 σ)
Delta-V magnitude error (smaller than 15 cm/s)	1.0 mm/s
Delta-V magnitude error (larger than 15 cm/s)	0.667%
Delta-V direction error	0.10 deg

Table 10. ExoMars 2016 mission EDM delivery accuracy

	LPO	LPC
EFPA error (3 σ) [deg]	0.256	0.269
Along track error (3 σ) [km]	22.1	22.7
Timing dispersion (3 σ) [sec]	4.47	4.56

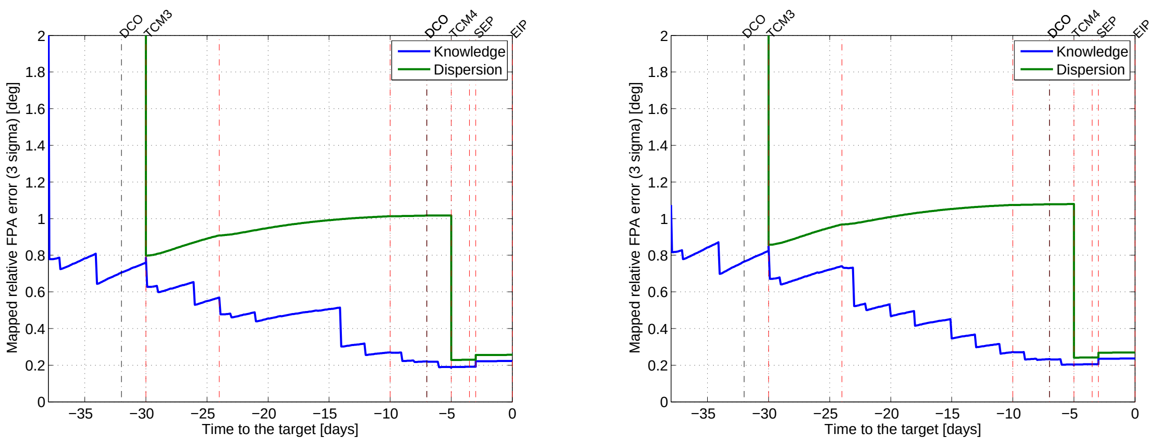


Figure 5. EXM 2016 mission EFPA uncertainty evolution (left: LPO, right: LPC)

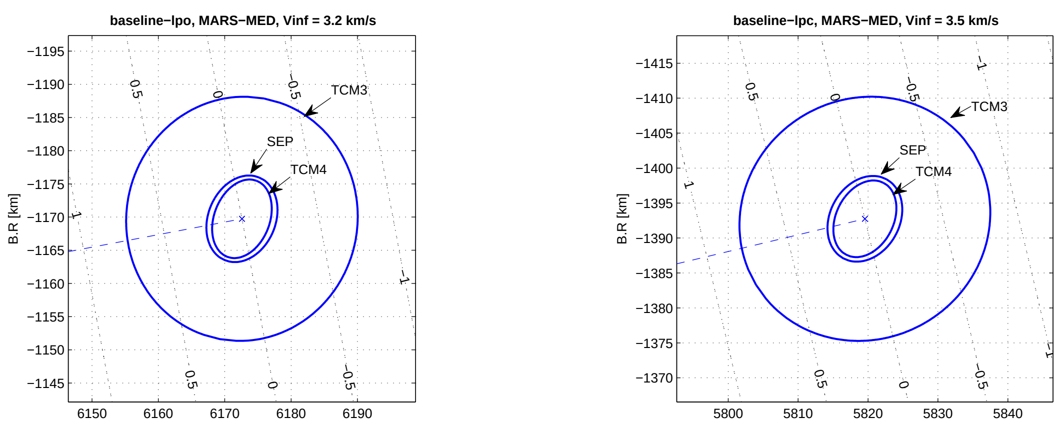


Figure 6. EXM 2016 mission B-plane dispersion ellipse (left: LPO, right: LPC)

Table 11. Sample landing sites for ExoMars 2018 mission

Site	Longitude [deg]	Latitude [deg]
Mawrth Vallis	20 W	25 N
Pathfinder	34 W	20 N
Amazonis Planitia	155 W	15 N
Isidis Planitia	90 E	10 N
Elysium Planitia	142 E	5 N
Gale Crater	137 E	4.5 S

Table 12. Tracking schedule for ExmMars 2018 navigation analysis

Phase	Dates	Range and Doppler	DDOR
Launch	Launch to L + 30 days	7 pass/week with CEB	None
Cruise	L + 30 days to E - 50 days	3 pass/week with CEB	None
Approach 1	E-50 days to E - 30 days	7 pass/week with CEB	2 point/week
Approach 2	E-30 days to E - 8 days	7 pass/week with CEB	3 point/week
Approach 3	E-8 days to Entry	7 pass/week with CEB	4 point/week

3.2. ExoMars 2018 mission

The navigation analysis for ExoMars 2018 mission is performed to investigate the DM delivery accuracy at the EIP. Since the landing site of the rover is not fully determined, the analysis have been performed for six sample landing sites spanning the range of latitude to be regarded (-5 S to 25 N). The sample landing sites are summarised in Tab. 11.

As well as ExoMars 2016 mission, TCMs are performed in order to meet the target Mars arrival conditions. Five TCMs have been assumed in this analysis. TCM-1 (Launch +7 days) is to clean up the launcher dispersion error and the following TCM-2 (Launch +30 days) is to clean-up the TCM1 error. TCM-3 (EIP -30 days), TCM-4 (EIP -8 days) and TCM-5 (EIP -2 days) are performed for fine targeting to the EIP. The last TCM is later than that of ExoMars 2016 mission because ExoMars 2018 mission does not have an orbiter and no need of a re-targeting maneuver. The TCM schedule is summarized in Tab. 4. The TCM execution error is assumed to have a minimum fixed error and a proportional error for the TCM magnitude as well as 2016 mission.

Since the trajectory of the ExoMars 2018 is the direct transfer to Mars, the launcher vehicle will off-target from the optimum direction of the reference trajectory. This is to comply with the planetary protection requirement.

3.2.1. Measurements

For ExoMars 2018 mission, the navigation analysis have been performed from the launcher separation until the EIP. Two cases, LPO and LPC, have been regarded. Table 12 shows the tracking schedule used for this navigation analysis. The tracking schedule is similar to that of the 2016 mission. The same DCO duration of two days is assumed.

Table 13. Exponentially correlated process noise

Contents	value (1 σ)	Corrected time
Non-gravitational acceleration	1.0e-11 km/s^2	1 day

Table 14. ExoMars 2018 mission spacecraft delivery accuracy (LPC)

Landing site Latitude [deg]	Mawrth 25	Pathfinder 20	Amazonis 15	Isidis 10	Elysium 5	Gale -4.5
EFPA error (3 σ) [deg]	24.44	20.44	19.27	18.82	16.56	13.02
Along track error (3 σ) [km]	5.42	4.85	3.93	4.26	3.82	3.14
Timing dispersion (3 σ) [sec]	0.29	0.24	0.23	0.22	0.20	0.15

3.2.2. Considered uncertainties

The considered uncertainties of the measurements, biases and maneuver execution errors are consistent with the assumptions of ExoMars 2016 mission in Tab. 6, 7 and 9. The errors on the solar radiation pressure model and slewing manoeuvres during the Mars approach are considered in the NGA error as exponentially correlated process noise, as well as the ExoMars 2016 mission. However, the NGA level is different because of the different configuration of the spacecraft. The value is summarized in Tab. 13.

3.2.3. Navigation results of ExoMars 2018 mission

The dispersion errors mapped to the EIP for the six sample landing sites are summarized in Tab. 14. The results show the difference in the delivery accuracy for each landing site. It is clear that there is large variation in the EFPA dispersion due to the landing site latitude. There is a linear relationship between the EFPA dispersion and the landing latitude. Since the B-plane dispersion ellipses are similar for the all six sites, the difference comes from the orientation of the dispersion ellipse with respect to the B-vector direction. Figure 7 shows the difference of the B-vector orientation for the landing sites, however Fig. 8 illustrates that the orientation of the B-plane dispersion ellipses with respect to the B.T axis is similar.

To assess the options for improvement of the EFPA dispersion, parametric analyses have been performed for the following parameters.

- NGA level : this depends on the spacecraft design, especially the thrusters layout
- DCO duration : this depends on the ground segment data processing
- Delta-DOR frequency, noise : this depends on the ground station schedule and the ability.

The case Mawrth LPO is selected as a baseline of this analysis because it is the worst case for the six landing sites.

The results are summarised in Tab. 15. The largest contribution are the NGA level and the Delta-DOR noise. Those parameters improve the EFPA dispersion about 15 %. The DCO duration reduction improves the EFPA dispersion with about 11 % because the reduction allows to consider an

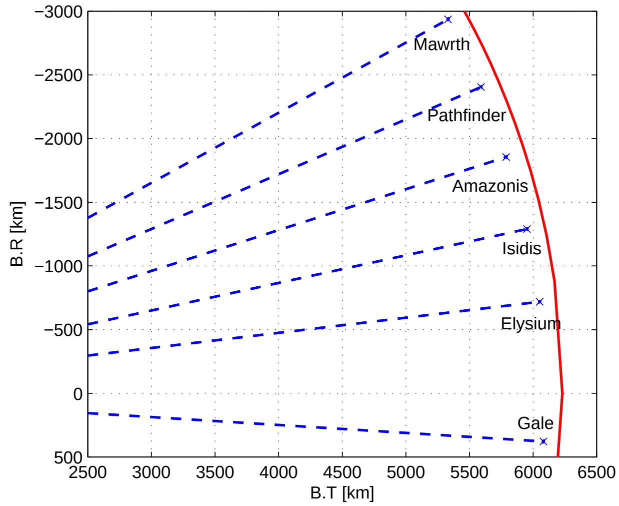
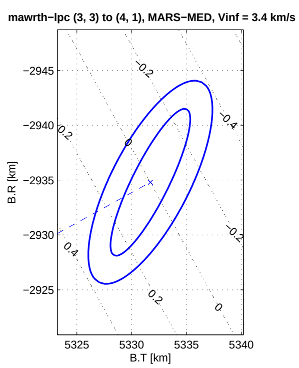
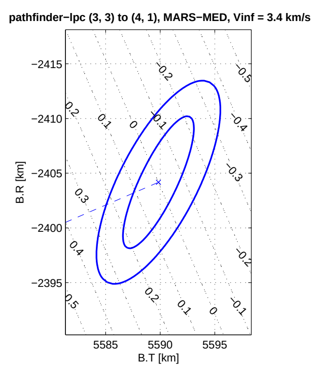


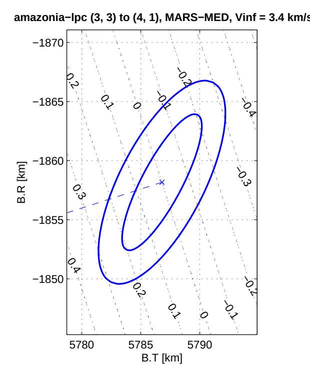
Figure 7. B-vector orientations of the six sample landing sites



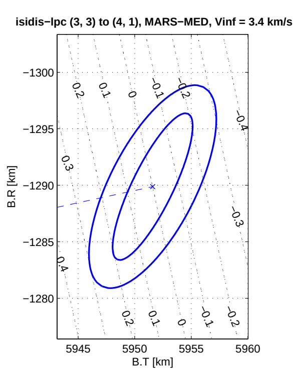
(a) Mawrth (25 N)



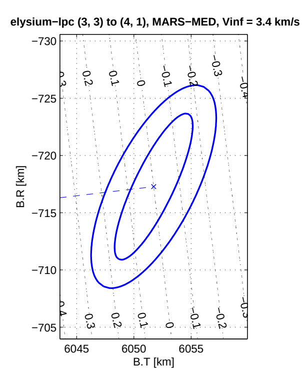
(b) Pathfinder (20 N)



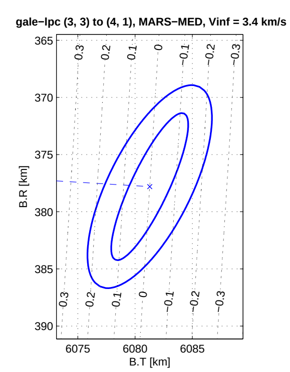
(c) Amazonis (15 N)



(d) Isidis (10 N)



(e) Elysium (5 N)

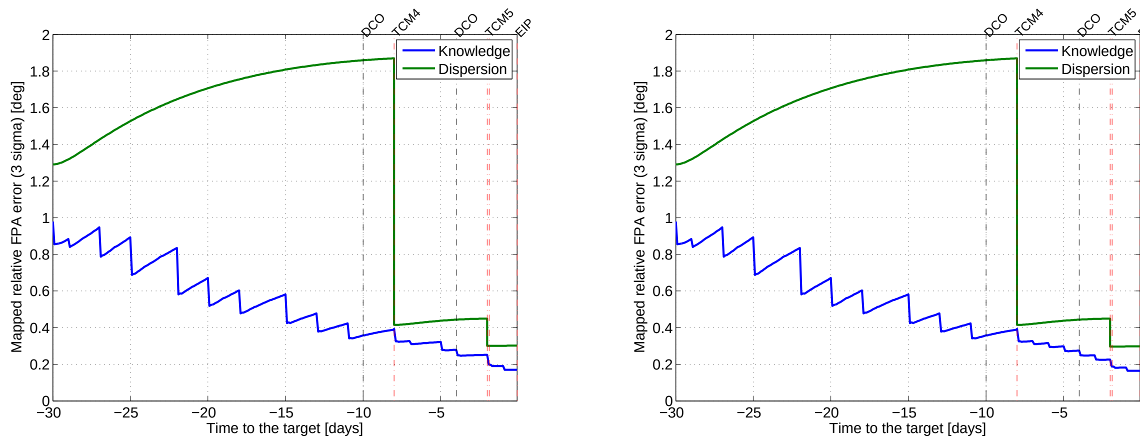


(f) Gale (4.5 S)

Figure 8. EXM 2018 The B-plane ellipse for six sample landing sites

Table 15. Sensitivity analysis for Mawrth LPO case

	Baseline	NGA	DCO	DDOR freq	DDOR noise
NGA level [km/s^2]	1.0e-11	5.0e-12	1.0e-11	1.0e-11	1.0e-11
Data-cut-off [days]	2.0	2.0	1.0	2.0	2.0
DDOR frequency [per week]	4	4	4	7	4
DDOR noise [cm]	5.0	5.0	5.0	5.0	2.0
EFPA error (3 σ) [deg]	24.3	19.4	21.3	23.9	19.6
Along track (3 σ) [km]	5.39	4.34	4.80	5.32	4.41
Timing dispersion (3 σ) [sec]	0.301	0.246	0.268	0.297	0.249



(a) DDOR 4 point/week

(b) DDOR 7 point/week

Figure 9. EFPA error evolution with DDOR frequency of 4 and 7 per week

extra Delta DOR measurement for the TCM-5. It is interesting that there is almost no improvement with adding extra Delta DOR measurements. This is because the added Delta-DOR measurements are not considered in the TCM-5 due to the DCO. Figure 9 shows the mapped dispersion and knowledge error with and without extra Delta-DOR measurements. The knowledge error evolution shows that indeed there is improvement due to the Delta-DOR, but the last Delta DOR measurement considered in the TCM-5 is the same for both cases. With current assumption, the higher Delta-DOR frequency is only meaningful together with the DCO reduction.

3.2.4. Off-targeting for planetary protection

The probability of the launch vehicle upper stage entering the Mars atmosphere is assessed to ensure compliance with planetary protection requirements[4]. If necessary, Earth escape must be off-targeted; the trajectory is then re-targeted towards the correct arrival point via a Launcher Injection Correction (LIC) maneuver (= TCM-1). For the 2016 mission, the LIC only corrects the launcher injection dispersion and there is no need for deterministic retargeting, as the presence of a large DSM during interplanetary transfer already ensures that the upper stage will miss Mars. Conversely, for the 2018 mission, which has no DSM, Earth escape off-targeting is a must and an LIC inclusive of deterministic re-targeting is mandatory. This constitutes a major contribution to the navigation delta-v budget for the 2018 mission.

The following two different planetary protection requirements are considered in this section.

- The probability of impact on Mars of the launcher upper state shall be $\leq$ 1.0e-4 for the first 50 years after launch.
- The probability of non-nominal impact on Mars by the spacecraft shall be $\leq$ 1.0e-2.

The impact probability P is evaluated by integrating the dispersion ellipse on the B-plane over an impact disk of Mars. The radius of the impact disk is calculated using the incoming infinite velocity and the reference radius of Mars. The reference radius is 3516.0 km, which is the Mars radius plus 120 km for the atmosphere. For the LPO of the ExoMars 2018 mission, the radius of the impact disk is 6136 km. Assuming that the dispersion ellipse is independent from the B-plane aiming point $\boldsymbol{b}$ and given by the previous navigation analysis, the optimal biased targeting points $\boldsymbol{b}_i$ are solved by the following formula.

$$\begin{aligned} \text{Minimize} &: \sum \Delta V_{det,i} \\ \text{Optimize} &: \boldsymbol{b}_i \\ \text{subject to} &: P_i \leq P_i^u \qquad\qquad i = 1,\ldots,n \end{aligned} \tag{1}$$

where $\Delta V_{det,i}$ is the deterministic delta-V to correct the $i-1$th aiming point $\boldsymbol{b}_{i-1}$ to ith aiming point $\boldsymbol{b}_i$, P_i^u is the the impact probability which has to be satisfied until the ith TCM epoch, and n is the total number of the maneuvers to correct the aiming point.

For ExoMars 2018 mission, two maneuvers TCM-1 (Launch + 7 days) and TCM-3 (Entry - 30 days) are considered in this off-targeting calculation to comply with the impact probability P_i^u of 1.0e-5 and 1.0e-2, respectively. The impact probability of 1.0e-5, not 1.0e-4 is assumed to be on safe side. The second planetary protection requirement is defined as the impact probability after each TCM multiplied by the probability that the following maneuver does not occur. In this analysis, the possibility of the TCM execution failure after TCM-3 is assumed to be less than 1.0e-2. In absence of the Proton M/Breeze M launcher dispersion information, the covariance of upper stage separation is taken from the Soyuz/Fregat launch of Mars Express for a preliminary analysis. In this analysis, the sample landing site Mawrth Vallis with the LPO trajectory is selected to be an reference case to evaluate the off-targeting impact to the navigation delta-V budget.

The biased aiming point with off-target for Mawrth LPO is shown in Fig. 10. The 3σ B-plane dispersion ellipse of the launcher dispersion is off-targeted from the final target for the landing site.

The navigation analysis has been performed considering the off-targeting in order to evaluate the navigation delta-V budget. Table 16 summarizes the TCM delta-V statistics. The TCM-1 and TCM-3 combine the deterministic re-targeting and the stochastic trajectory correction. The ΔV 99% is calculated by 50,000-sample Monte Carlo maneuver analysis considering the orbit determination and maneuver execution errors. The TCM-1 (LIC) is the largest maneuver and the following maneuvers are negligible size. The total delta-V cost satisfies the navigation delta-V budget.

Table 16. ExoMars 2018 Summary of navigation delta-V

Manoeuvre	Timing	P	ΔV_{det} [m/s]	ΔV 99% [m/s]
TCM-1 (LIC)	L+7 days	1.00e-05	6.320	14.496
TCM-2	L + 30 days	-	-	0.176
TCM-3	E- 30 days	1.00e-02	0.389	0.739
TCM-4	E - 8 days	-	-	0.071
TCM-5	E - 2 days	-	-	0.083
Total	-	-	6.709	15.565

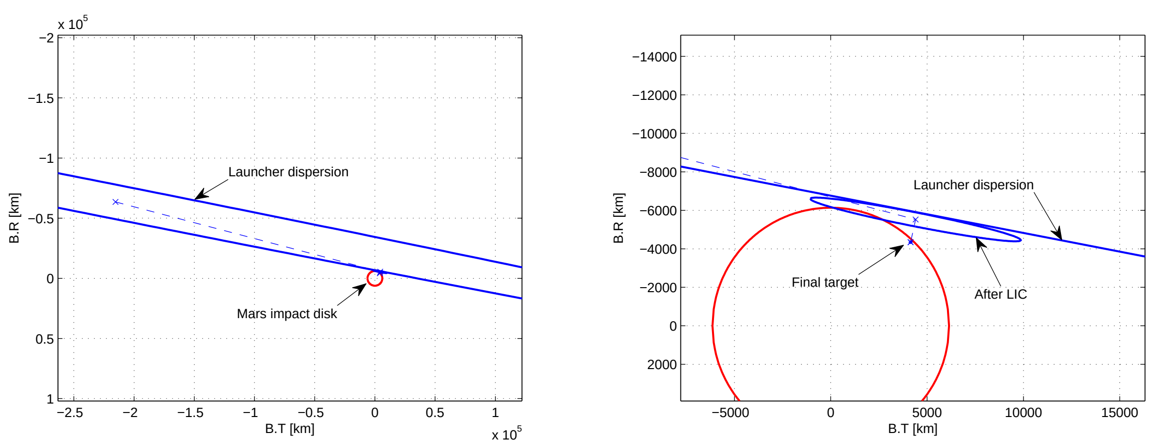


Figure 10. EXM 2018 mission B-plane error ellipse with escape off-targeting to comply with planetary protection requirements

4. Conclusions

This paper has summarized the mission design and the navigation analysis of ESA ExoMars program. Both ExoMars 2016 and 2018 mission have been described. The 21-day launch period is decided in order to satisfy the Proton M/Breeze M launcher and Mars arrival condition constraint of the spacecraft composite. The navigation analysis has been performed in order to investigate the delivery accuracy of the EDL Demonstration Module for 2016 mission and the Descent Module for 2018 mission. The off-targeting of the launch injection aiming point is planned to comply with the planetary protection requirement for 2018 mission.

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