Atmospheric Drag, Occultation 'N' Ionospheric Scintillation (ADONIS) Mission proposal

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The Atmospheric Drag, Occultation 'N' Ionospheric Scintillation mission (ADONIS) is a space weather mission that studies the dynamics of the terrestrial thermosphere and ionosphere over a full solar cycle in Low Earth Orbit (LEO). The objectives are to investigate satellite drag with in-situ measurements, and the ionospheric electron density profiles with radio occultation and scintillation measurements. With a constellation of two spacecraft it is possible to provide near real-time data (NRT) about ionospheric conditions over the northern polar region where current measurements are currently insufficient. The mission shall also provide global high-resolution data to improve ionospheric models. The low-cost constellation can be launched using a single Vega rocket and most of the instruments are already space-proven which allows rapid development and reliability.

1 Introduction

The Sun is continuously emitting particles and electromagnetic radiation into interplanetary space. In addition to the constantly flowing solar wind, Coronal Mass Ejections (CMEs), Corotating Interaction Regions (CIRs) and flares are typical examples of solar activity which have significant impact also on near-Earth space and the Earth's atmosphere. When particles and electromagnetic energy from the Sun reach Earth, both can affect its atmosphere by heating and ionisation. This coupling between Sun and Earth is responsible for forming the ionosphere. The Sun follows an eleven year activity cycle correlated with the occurence rate of typical solar events.

Variations in the ionised particle density in the ionosphere cause satellites signals and other radio waves to be bent. At times of enhanced solar activity dramatic changes in ionospheric and thermospheric properties are observed, often followed by a fallout of communication systems as well as Global Navigation Satellite Systems (GNSS). The nature of this signal refraction has to be determined accurately, since it can result in errors to positioning and can render High-Frequency (HF) radios unusable. During especially energetic events the error on navigational systems can be above the accepted limits set out by the United States' Federal Aviation Authority (FAA) [1]. Also systems such as power lines and telecommunication lines are at risk from ionospheric currents (see [2]), and can affect safety-critical or emergency systems. Additionally, thermal expansion of the atmosphere due to the enhanced solar activity increases the drag felt by satellites. Common examples are the re-entry of the Skylab mission [3] or the fast decay of the International Space Station's orbit requiring frequent altitude boosts.

Studies of the ionosphere with sounding from the ground (ionosondes, radars), satellites and sounding rockets have been performed over decades, however the effects in the thermosphere and ionosphere due to space weather are still not well understood and modelled on a global scale.

Numerous missions have been carried out to study the atmosphere and the ionosphere. Kosmos and CHAMP applied radio occultation measurements, and there was also a microsatellite mapping ionospheric scintillations, called STPSAT1. Drag measurements have been performed by the GOCE satellite, at altitudes below 300 km. The QB50 mission, which is going to use a constellation of 50 cubesats, plans to investigate the drag in the lower thermosphere over a period of three months.

To improve our understanding of the dynamical behaviour of the ionosphere and thermosphere due to changes in the solar activity, we propose a satellite mission that shall provide necessary in-situ measurements of atmospheric parameters important for the drag on satellites at different altitudes. The ADONIS mission also aims at providing measurements important for the refraction of radio signals through the ionosphere. ADONIS is unique in the sense that it applies both techniques (radio occultation and scintillation measurements) simultaneously, and by having two satellites on different orbits, an unprecedented coverage is obtained.

Our proposal is organised as follows: In Section 2 a mission overview is given, Section 3 and 4 describes the mission and the spacecraft designs. The development and cost are presented in Section 5, and finally, conclusions are given in Section 6.

2 Mission overview

2.1 Mission statement

ADONIS is proposed here to study the dynamics of the terrestrial ionosphere and thermosphere for the duration of a full solar cycle. ADONIS shall determine the key parameters in the ionosphere and the thermosphere in relation to satellite drag and signal propagation. The long mission lifetime shall facilitate investigation of the effects of the variability of solar conditions on Earth's atmosphere.

2.2 Objectives

***Obj. 1*: Study the dynamics of the thermosphere and its effects on satellite drag in-situ, in the Low Earth Orbit (LEO) region at 300–800 km.** Drag models of LEO satellites show deviations up to 20% from the actual behaviour [4]. This leads to satellite operators overestimating the fuel required, and also to less accurate orbit predictions. Current modelling for satellite re-entry is not suited for a precise determination of the re-entry position, which is essential to ensure that satellites are de-orbited safely.

***Obj. 2*: Measure the ionospheric response to space weather events in order to derive electron density maps.** Ionosondes and Incoherent Scatter Radars (ISRs) measure most of the ionospheric data from the ground. Both have their limitations such as sparse global coverage and limited vertical distribution. The ADONIS mission shall improve the global coverage and the provision of Total Electron Content (NRT) data for the northern polar region.

***Obj. 3*: Provide measurements relevant to satellite drag and to the ionospheric response to space weather events over a full solar cycle.** A long mission lifetime allows the observation of a large number of similar solar events and the ionospheric and thermospheric response to the electromagnetic flux and particle precipitation. This yields a comprehensive dataset for statistical studies and improved modelling of drag as well as the ionospheric and thermospheric response to Space Weather Events (SWE).

2.3 Requirements

The general requirements for the space mission derived from the objectives are the following: The ADONIS mission shall measure the acceleration on the spacecraft, the atmospheric composition, and the spacecraft temperature in order to provide key parameters for better understanding of atmospheric drag in the high atmosphere (*Obj. 1*). In order to provide the latter affecting telecommunications and navigation, global electron density profiles as well as the plasma parameters shall be determined (*Obj. 2*). The mission shall last at least 11 years (*Obj. 3*). In more detail, the following requirements are identified:

***Req. 1*: The mission shall determine in-situ the drag acceleration as well as the plasma and neutral density, temperature, velocity and the spacecraft surface temperature, which are relevant to model spacecraft drag at altitudes of 300–800 km.**

The drag experienced by a spacecraft is described by:

$$a = \frac{1}{2m}\rho v^2 A c_d, \quad (1)$$

$$c_d = c_d\left(T_0, T_S, n_p, m_p\right), \quad (2)$$

where a is the acceleration of the spacecraft, c_d the drag coefficient, m the mass of the spacecraft, A the spacecraft cross section, v the velocity with respect to the atmosphere, n_p the particle average number density, m_p the average particle mass, ρ the particle average mass density and T_0, T_S the temperatures of the atmosphere (neutrals) and the spacecraft respectively. The dependence of drag on these parameters is one of the main problems in building an accurate drag model.

The mission gives better coverage in latitude, longitude, altitude and local time and longer duration of the measurements than the planned QB50 [5] and GOCE missions [6], which can also measure drag.

***Req. 2*: The mission shall determine the acceleration with an accuracy of 10^{-8} ms^{-2} and a cadence of 1 Hz.** Using the NRLMSIS-00 model [7], the expected average drag acceleration is determined as 10^{-6} ms^{-2}. In order to cover small changes of drag, an accuracy of two orders of magnitude higher than the expected average acceleration is required. ADONIS shall measure the acceleration of the spacecraft with the cadence 1 Hz.

***Req. 3*: The mission shall provide a global daily coverage with a resolution higher than 15° longitude and at different altitudes in order to cope with ionospheric dynamics.** *Obj. 1* and *2* require a low longitudinal separation with a short repetition time. Drag measurements require low passes whereas ionospheric measurements are more complete from higher altitudes.

***Req. 4*: The mission shall provide Near Real-Time (NRT) coverage of the northern polar region.** The strongest and most recurrent ionospheric space weather effects occur at high latitudes, but ground-based measurements are sparse in these regions. A uniform spatial coverage in NRT above the northern polar region is required in addition to *Req. 3* to meet *Obj. 2*.

***Req. 5*: The mission data shall enable the derivation of global electron density maps with altitude resolution comparable to ground based system.** The mission shall use radio occultation to derive the electron density profile. Ionospheric scintillation shall also be used, to gain information about transient ionospheric phenomena such as polar cap patches and how they affect signal propagation.

In order to reach the mission goals and to improve on current ISR experiments e.g. EISCAT, the Common Program (CP) 'manda' on a UHF/VHF system is considered as a baseline. This experiment has a (vertical) spatial and temporal resolution of ¡450 m and 6 s respectively. The range for this experiment is also limited to $58.5 - 513$ km [8].

***Req. 6*: The mission shall determine in-situ the bulk velocity and ion/electron temperatures with a higher cadence than ground based systems.** Bulk velocity measurements provide information about the plasma flow in the ionosphere. This augments ground-based radar measurements, which provide line-of-sight ion bulk velocity either away from or towards the beam.

***Req. 7*: The mission shall determine the magnetic field in-situ with higher cadence than 1 s and resolution of 0.5 nT for a dynamic range of ±80,000 nT.** This is required to get a complete picture of magnetic field changes due to solar wind influences causing ionospheric perturbations. The magnetic field in the ionosphere can vary between magnitudes of 25000 nT at the equator and 65000 nT at the poles. Small perturbations of the order of nT are caused by ionospheric currents and plasma waves, these perturbations have timescales of a few seconds [9], therefore a resolution of 0.5 nT is required at sub-second cadence.

***Req. 8*: The mission shall operate for a full solar cycle.** In order to satisfy *Obj. 3* the mission has a lifetime of 11 years.

3 Mission design

In order to meet the requirements mapped out in Section 2, the ADONIS mission is designed as a constellation of two identical spacecraft, A-DONIS and B-DONIS, orbiting in LEO in almost perpenticular orbital planes, covering altitudes from 300–800 km with constantly moving apogee. The constellation is possible to be launched with a single launch vehicle.

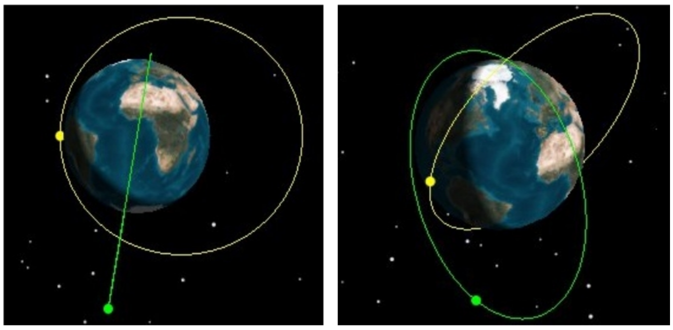

Figure 1: The orbits of A-DONIS (green) and B-DONIS (yellow) in their final configuration (eccentricity exagerated).

3.1 Orbit

The final satellite configuration consists of two different elliptical orbital planes, both with a 80° inclination with the apogee at 800 km and perigee at 300 km. The need for a nearly polar orbit rises from the fact, that a main goal of the mission are the in-situ measurements with high spatial resolution around the whole globe. The orbit was intentionally chosen to not be sun-synchronous to enable the observation of the same location at different local time. The apogee altitude was limited due to the total ionisation dose, which would be accumulated over the 11 year lifetime. The perigee altitude was chosen to be low, since the presence of higher density layers (resulting to higher drag accelerations) would allow the accelerometer to perform accurate measurements over a wider range of altitudes.

Initially, the two satellites are positioned in the same orbital plane. An impulse change is imparted on A-DONIS ($\Delta V = 0.14$ km/s) and on B-DONIS ($\Delta V = 0.09$ km/s) in order to achieve a circular (800 km) and elliptical (300×500 km) orbit, respectively. The precession rates of the Right Ascension of the Ascending Node (RAAN) is thus different, with a relative precession rate of 0.255°/day. By lowering the inclination of the orbits from (the theoretically ideal for coverage) 90° down to 80°, it is possible to take advantage of this drift and control its precession rate. The difference in RAAN becomes equal to 90°, 340 days after launch. The same ΔV as for the initial orbits are applied to A-DONIS and B-DONIS respectively (apogee kick burn) in order to achieve two identical orbits (300×800 km), with a 90° difference in the two orbital planes (RAAN) and 90° difference in the argument of perigee. By this configuration build-up, the mission saves propellant and lowers the cost, since the plane change is carried out by the gravitational perturbations rather than an on board propulsion system or an additional launch. The final configuration provides the required spatial and temporal coverage, as shown in Fig. 2.

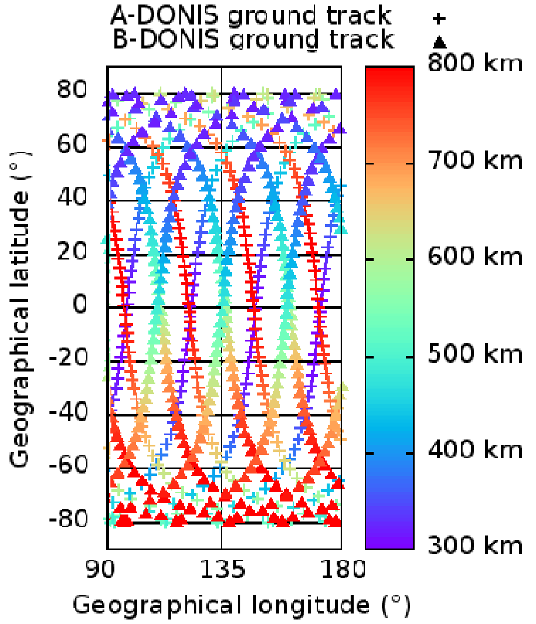

Figure 2: Coverage of the ADONIS mission. Longitude shown from 80 to 180°. The coverage is identical for other longitudes.

3.2 Instrumentation

The payload is selected to fulfil the mission requirements. All instruments, except for the 3D particle analyser (3DPA), have heritage from previous missions, thus the instrumentation is space proven, has Technical Readiness Level (TRL) between 7 and 9 and development costs are saved. The list of instrument ranges and resolutions is given in Tab. 1, and further technical details in Tab. 4.

The **Ion and Neutral Mass Spectrometer (INMS)** contributes to *Req. 1, Req. 5* (see Section 2.3). It measures the mass spectrum of low-mass ionised and neutral species. Neutral compounds are important for drag measurement, and the change in total composition is important for ionospheric monitoring. The instrument has been designed for use on the QB50, the range of measurement is adjusted to the mission requirements (see Tab. 4).

The **3D Particle Analyzer (3DPA)** is chosen to meet *Req. 5*. It provides the velocity and temperature moments of ions and electrons calculated from their spatial distribution with a high time resolution of 100 ms and provides a 4π sr field of view even on a three axis stabilised spacecraft. The instrument is not yet proven in a relevant environment, however has TRL 7 and is going to be tested on the ICI-4 sounding rocket in December 2013 [10]. The 3DPA consists of two cylindrical detectors with 168 entrance windows. Other detectors generally rely on the spacecraft spin to obtain three components of velocity.

The **Langmuir probes (m-NLP)** determine the electron density and contribute to *Req. 5*. They are also providing in-situ data comparable to the occultation and scintillation measurements. The Langmuir probes on the ADONIS satellites are identical to those of the QB50 satellites, but the range of the measurements is adjusted to the mission needs [11].

Radio occultation measurements shall satisfy *Req. 4*. With the **Integrated GPS Occultation Receiver (IGOR+) instrument**, A-DONIS and B-DONIS receive GNSS signals which get Doppler-shifted while gaining frequency-dependent (dispersive) ionospheric delay. By measuring this shift in frequencies, the Total Electron Content (TEC) of the ionosphere along the ray path is derived with a relative accuracy of 10^9electrons m^{-2} i.e. 0.001 TEC Unit (TECU). IGOR is operational on the COSMIC/Formosat-3 constellation. Its 0.5 m 3D root-mean-square (rms) position accuracy and 0.2 mm/s orbital velocity accuracy is provided by the instrument itself [12]. TEC values derived from the measurements on the required orbit extend the high-resolution ionospheric sampling of regions with sparse coverage (arctic regions, oceans and southern hemisphere). The development of engineering model of TriG receiver[13] (GALILEO, GPS and GLONASS) was finished, but its 50 W power requirement is too high for trade-off providing 500 additional Radio Occultations daily.

The **Computerised Ionospheric Tomography Receiver in Space (CITRIS)** is designed to map local scintillations in the ionosphere by simultaneous measurements of Doppler- and phase shifts of radio beacons transmitted by ground stations and 10 more LEO spacecraft. Hence, ray paths both vertically and horizontally through the ionosphere enable the derivation of TEC and plasma inhomogenities with relative accuracy of 0.003 TECU (1 TECU absolute) and 10 Hz sampling to determine S4 scintillation.

The measurement principle is similar to that of GNSS, but the ratio of the frequencies used is higher (in case of the ground stations 5.1, other LEO satellites 2.6, GPS only 1.3) [14]. Thus, ionospheric tomography by LEO spacecraft is able to provide a higher resolution for mapping scintillation (a few up to hundreds of meters). This is achieved by summing TEC over shorter ray paths.

Scintillation is at the moment not monitored by any operational spacecraft. At scales of meters to tens and hundreds of meters, the ionosphere changes in minutes, thus rapid sampling is important. Near real-time TEC data in the northern polar region shall be directly downlinked to Svalbard station, providing ionospheric monitoring products.

Thermistors measure the surface temperature of the spacecraft, which is one of several parameters influencing the drag coefficent. Thus this measurement is needed to fulfil *Req. 1*. The solar panels are already

Requirements	Range	Sensitivity	Instrument name
Particle composition	0–50 amu	0.4 amu, 1 Hz NRT	Ion & Neutral Mass Spectrometer (INMS)
On-board temperature	-170 to 160 °C	1 °C , 1 Hz	Thermistors
S/C acceleration	0–5 g	10^{-8}m/s^2, 1 Hz	Italian Spring Accelerometer (ISA)
Plasma velocity, temp.	10 eV–30 keV	5 eV, 1 Hz NRT	3D Particle Analyser (3DPA)
Plasma density	10^9–10^{12} m^{-3}	10^9 m^{-3}, 1 Hz NRT	Langmuir Probe (m-NLP)
Total Electron Content	3–5 TECU	0.001 TECU, 10 Hz	RF occultation instr. IGOR+
Total Electron Content	1–3 TECU	0.003 TECU, 10 Hz S4	Radio tomography receiver (CITRIS)
Magnetic field	±80 µT	0.5 nT, 10 Hz NRT	Flux Gate Magnetometer (FGM)

Table 1: Instrument range and sensitivity. NRT: Near Real-Time.

equipped with thermistors. Additionally there are thermistors on the front and back surface.

The **Fluxgate Magnetometer's (FGM)** heritage is from the Cluster and THEMIS missions [15]. It measures the changes of the magnetic field components with the range and resolution specified in Tab. 1. Such resolution is unusual, but it is technically feasible [16]. The sensor is positioned at the tip of a short 1.0 m boom.

The **Italian Spring accelerometer (ISA)** is scheduled to fly on BepiColombo. In order to achieve the desired accuracy of 10^{-8} ms^{-2} with a sampling rate of 1 Hz, customisation of the ISA is necessary to satisfy *Req. 2* [17]. The ISA is very sensitive to temperature changes and so it is covered by a thermal sytem to keep the temperature stable [18].

3.3 Launcher

The satellites are launched with an Arianespace Vega launcher. Vega has a liftoff mass of 137 tonnes and is able to carry up to 1.5 tonnes of payload to an 800 km circular orbit. The ADONIS mission has a total payload mass of 1.1 tonnes. which leaves a margin of 400 kg for the dual launch adapter, saving costs by using the most cost-efficient rocket available to put the satellites into the required orbit. The launch is expected in 2019 at the Guiana Space Centre, Kourou, French Guiana.

3.4 Ground segment

The ground segment of ADONIS consists of a ground station (GS), Mission Operations Centre (MOC), Science Operations Centre (SOC) and Space Weather Services. An overview of the mission ground segment is shown in Fig. 4.

The **Ground Station** selected for the mission is the Svalbard SG-3 ground station, owned and operated by the Norwegian company Kongsberg Satellite Services AS (KSAT). It uses S-Band for downlinking housekeeping telemetry and uploading telecommands (downlink scheduling, etc) to the space segment. Science and operational TM data is received using X-band.

The ground station location was picked due its position close to the North Pole, mainly for good ground coverage of satellite passes and secondly to have short time delay for downlinking the near real-time space weather data in the North Pole region.

The number of ground passes changes due to the precession of the satellite orbital planes around the North pole, which has a period of about two-thirds of a year. It was calculated that in the best case scenario (when the intersection of the orbital planes is closest to the ground station) the ground station has a coverage of $> 95\%$ of orbits of both satellites.

In the worst case scenario (intersection furthest away) there is a maximum of four consecutive uncovered orbits for one of the satellites and an average of 13.5 of 15.15 orbits per day (85%). The other satellite still remains covered at the maximum $> 95\%$. This relationship alternates between the two satellites, depending on the phase of the precession.

The average ground pass time was calculated to 10.5 minutes/orbit and worst-case (usable) ground pass was calculated to 4 minutes/orbit, resulting in a tracking requirement of about 4 h/day for both satellites together.

The order of downlinking operational and science TM data is based on a combination of priority schemes and schedules. NRT data always has the highest priority (unless explicitly decided otherwise by the operations team).

The **Mission Operations Centre (MOC)** is responsible for monitoring and maintaining of the flight critical systems of the space segment; performing orbital maintenance maneuvers,providing interfaces to as well from the Science Operations Centre (SOC) for science data and scheduling and data to the space weather services. Due to the use of the commericial SSTL satellite platform, standard SSTL ground station systems for ground control software and hardware is used. The **Science Operations Centre (SOC)** is responsible for scheduling the science measurements, downlink schedules, and priorities, as well as providing support to the MOC for instrument calibrations and maintenance. The on-board sci-

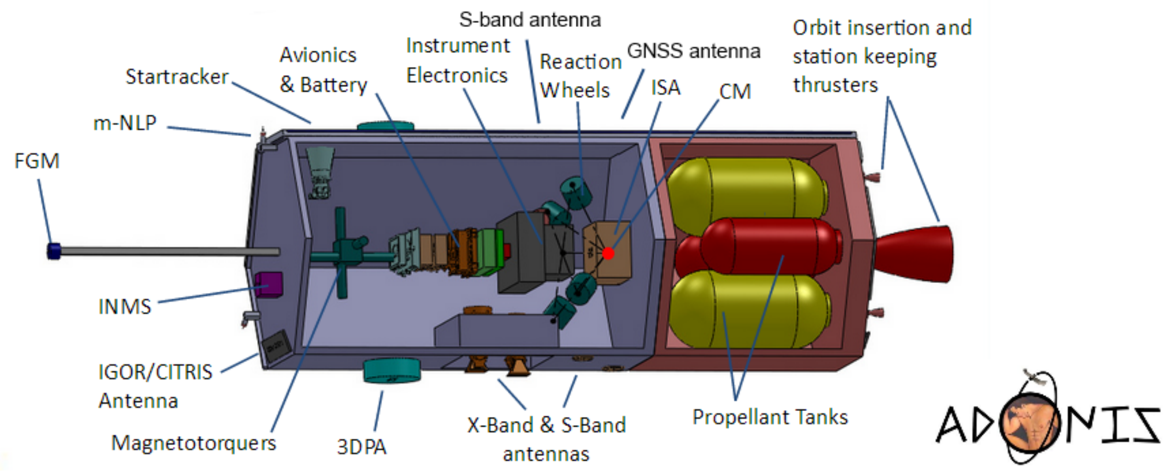

Figure 3: A-DONIS satellite preliminary layout.

ence instruments generate about 720 Mbit per orbit. In the worst case scenario (4 missed orbits/passes), 3600 Mbit of data is produced. The maximum downlink capacity for one pass is 4500 Mbit, giving a downlink margin of 720 Mbit.

The total generated scientific data during the mission is roughly 40 Tbit.

3.5 Disposal

ESA requires the removal of expired space systems. The requirement for the LEO region is the removal and disposal not later than 25 years after the end of the mission. The mission is structured so that the orbital decay is part of the scientific phase, and allows to investigate the drag at 300 km with different perigee velocities (circularisation phase) as well as to study of the drag below 300 km until the re-entry of the satellites (spiralisation phase). The mission lifetime of ADONIS shall end with a controlled re-entry using thrusters over unpopulated areas.

4 Spacecraft design

The ADONIS mission uses two identical spacecraft based on the commercial SSTL-300 platform (list price 23 M€), with a customised structure to meet the scientific requirements for both the drag (*Req. 1*–*Req. 3*) and the ionospheric measurements (*Req. 3*-*Req. 6*). To satisfy *Req. 1*, the spacecraft has a minimised frontal area ($0.8\,\mathrm{m}^2$) and a simple hexagonal shape with only one 1 m-boom deployed in the ram direction. The mass for each spacecraft is 560 kg including payload and fuel for orbit insertion, station keeping and maneuvers satisfying *Req. 7*. Fig. 3 shows the position of the instruments and substructures. The instruments are arranged in such a way that they do not interfere with each other. The main structure is a lightweight aluminium skin frame with aluminium skinned honeycomb panels. Avionics, instrument and propulsion components have to be aligned in the spacecraft so that the centre of pressure is ahead of the centre of mass. Customisation of the satellite platform also increases the lifetime to cover a full solar cycle. The preliminary design of the satellite is shown in Fig. 3.

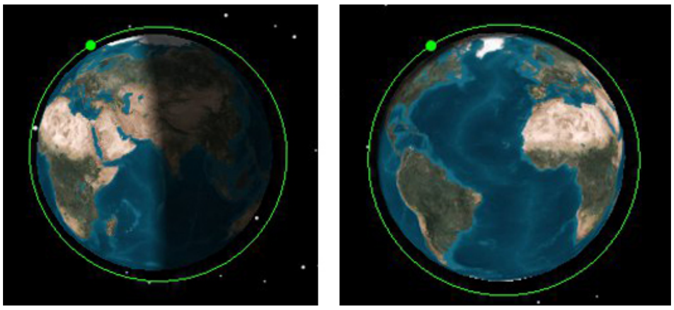

Figure 5: Cold (left) and hot (right) case scenario.

4.1 Power subsystem

The power subsystem design depends on the chosen orbit, the nadir pointing attitude and the drag measurements which are carried out. The orbital plane of the satellite slowly precesses with respect to the incoming solar radiation, which means that during their entire lifetime, each spacecraft has the incoming solar radiation on all its surfaces except the bottom.

The spacecraft experience significant drag in orbit, for these reasons only body-mounted solar cells on the top and lateral panels are selected instead of deployable ones. The solar panel area is designed for the worst and cold case scenario (see Fig. 5 left), where the solar panels are not exposed to sunlight for half the orbit (T = 48 min).

Spectrolab 29.5% NeXt Triple Junction (XTJ) GaInP2/GaAs/Ge are selected as solar cells providing a high specific power $P_0 = 398\,\mathrm{W/m}^2$, with a yearly degradation of 2.75%. Considering the losses in the efficiencies (assembly 5%, shadowing 5%, temperature 15%) and the total life degradation (30%)

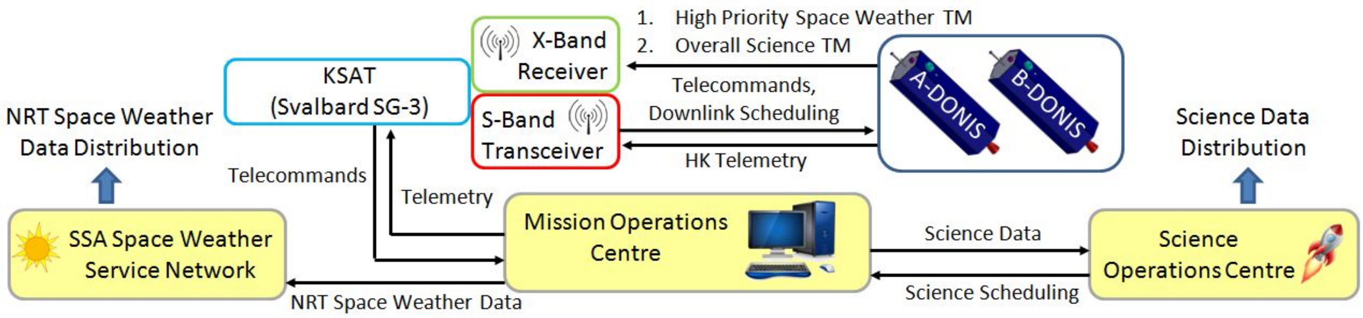

Figure 4: Ground segment overview.

the total designed exposed area is $A_{SP} = 2.5\,\mathrm{m}^2$. One solar panel on the top ($0.56 \times 2.6\,\mathrm{m}^2$) and two on the top lateral panels ($0.56 \times 1.8\,\mathrm{m}^2$) provide the necessary power, in the worst case scenario, at the end of life.

Lithium Ion Batteries are chosen to cover the power requirements during the eclipse periods, with a capacity of 38 Ah, maximum current of 38 A and nominal voltage of 3.6 V. One single battery provides 90 W with a mass below 1 kg (including a margin of 20%). The depth of discharge is about 75%.

4.2 Thermal design

The thermal design of the satellites for the hot and cold scenarios are presented in Fig. 5. Thermal insulating material (MultiLayer Insulation – MLI) is designed to cover all the exposed areas of the lateral panels of the spacecraft, beneath the solar panels, in order to thermally decouple them from the satellite body.

The radiators are designed based on the radiative thermal exchange between the spacecraft, the incoming solar flux, and the Earth's radiation flux along the orbit. The following optical properties are chosen for the radiators: emmissivity $\epsilon = 0.8$ and absorption coefficient $\alpha = 0.1$. Due to the varying orientation of the spacecraft with respect to the solar radiation expected during the satellite's lifetime, multiple radiators covered with louvers are used. The design of the radiators is done for the hot case scenario (see Fig. 5 right) during which the satellite is always exposed to the solar radiation (quasi dawn-dusk orbit). The radiators' dimensions are determined based on a trade-off between the excess of power produced by the lateral solar panels and the satellite total length (length of the lateral solar panels and lateral radiators) shown in Fig. 3. The area of each radiator is $0.4\,\mathrm{m}^2$ including a 20% margin. The solar panels, since decoupled from the spacecraft body, reach an equilibrium temperature during the orbit. The loss of efficiency due to off-nominal operative temperature of the solar panels is taken into account in the solar panels design.

In addition to the passive one, an active thermal control is installed in order to transfer the heat from the internal components and instrumentation toward the radiators. The active thermal control with be mainly necessary during the hot case (quasi dawn-dusk orbit) during which extra power is produced.

During the cold case scenario (quasi noon-midnight orbit), during the periods of eclipse, it is possible to reduce the power dissipation from the radiators by reducing or turning off the heat that reaches the radiator (mainly due to electrical power) or even reducing the total exposed area of the radiators with the louvers. During the worst conditions of the cold case scenario (maximum eclipse of 34 minutes) heaters will provide the necessary heat to the sensitive onboard instrumentation using the batteries (20 W at worst).

4.3 AOCS

The AOCS of the spacecraft is constrained by the accuracy requirements of the payload instruments. From the AOCS point of view *Req. 1* is best met by providing constant cross sectional area with respect to the flight vector. This means that the spacecraft is 3-axis stabilised in order to ensure identical aerodynamic conditions during the drag measurements. In order to stabilise all three rotational axes, the actuators used, four reaction wheels, are installed in pyramidal configuration. The desaturation of the wheels takes place by the usage of 3 magnetotorquers and has to be performed every 36 hours, in the worst case, which includes a constant external torque around the pitch angle. For the attitude determination, the system is equiped with 2-axis Sun sensors, one star tracker and a 3-axis magnetometer, which provides the data for the magnetic coils. The measurement of the angular velocity is carried out by a laser gyro. In order to make the communication with the ground station feasible, the antennas should point down in the radial direction during the passages over GS. For these reasons, the resulting attitude should always be achieved by yaw steering nadir pointing mode, in oder to avoid drag on the side areas of the spacecraft. The centre of mass has been chosen to be behind the centre of volume, to provide aerodynamic stability

S/C	ΔV (m/s) A	B	Prop. (kg) A	B	Margin
Injection	280	185	58	32	25%
Tr. corr.	5	100	2	28	35%
Orbit corr.	500	500	75	75	40%
Avoidances	100	100	15	15	10%
Total	885	885	150	150	10%

Table 2: ΔV and propellant budget for the ADONIS mission (Tr.: transit, corr.: correction).

4.4 Propulsion system

Both satellites have a propulsion system which is used to perform the orbital maneuvers and corrections during the mission's lifetime. Each spacecraft needs to carry out an impulsive maneuver in order to change its orbital elements during the constellation build up procedure. For this reason a 100 N bi-propellant engine using MMH and NTO is installed on board (Isp=300 s). Apart from the orbital injection, it is very important to ensure that the propulsion system will compensate for the drag deceleration, which the spacecraft experiences when flying in LEO. Therefore, four additional propulsion engines have been added to each satellite (10 N, Isp=300 s) using the same propellant as the larger engine. The total ΔV change due to drag over the period of 11 years has been simulated (with overestimated solar flux and geomagnetic activity) and was used to derive the needed propellant mass for the orbital correction. The total ΔV budget and propellant needs are demonstrated in Tab. 2. The propellant needed for the counteracting of the accumulated drag deceleration played an important role in the decision of the perigee altitude, due to the extreme increase in needed propellant mass for lower altitudes.

4.5 OBC & OBDH

Onboard data handling (OBDH) and monitoring functions are provided by two redundant SSTL OBC750 on-board computers (OBC). A real-time operating system is used to support SSTL's standard spacecraft on-board software which controls and monitors all the on-board systems.

A dual-redundant CAN bus provides communication between the subsystems and the OBC. The CAN bus is a resilient high-speed serial platform which runs at 388 kbit/s.

Control algorithm, data gathering of analog sensors and control of actuators is provided by the ADCS (Attitude Determination and Control Subsystem) which is running on the OBC. The ADCS modules interface between the CAN bus and analog sensors and actuators. On-board time is provided by the GPS receivers.

Band	X	S (down)	S (up)
Data rate	105 Mbit/s	38.4 kbit/s	19.2 kbit/s
Frequency	8.5 GHz	2.2 GHz	2.1 GHz
Tx Power	5.0 W	0.5 W	~15 W
Tx Ant.	10 cm horn	8 cm patch	13 m dish
Rx Ant.	13 m dish	13 m dish	8 cm dish
Margin	6.0 dB	15.4 dB	40.6 dB

Table 3: Telecommunication system summary.

The on-board science data storage is provided by a SSTL High Speed Data Recorder (HSDR), with 128 Gbit storage capacity. It also includes the interfaces between the platform and the payloads for the science data, utilising the internal LVDS drivers in the HSDR for redundant 10 Mbit/s SpaceWire links to each instrument [19].

4.6 Telecommunications

The telecommunication system uses a combination of S-band and X-band links. S-band will be used for receiving telecommands and relaying housekeeping and control telemetry to ground. X-band will be used for downlinking the science telemetry. Tab. 3 gives a summary of the typical telecom system of SSTL-300 commercial platform.

The on-board telecom system of ADONIS spacecraft include the following SSTL products: XTx400 X-Band Transmitter, S-Band Uplink Receiver and S-Band Downlink Transmitter. For the S-band systems two opposite facing SSTL S-Band Patch Antennas are used (for a near spherical gain pattern) and for the X-band tranmitter a SSTL Antenna-Pointing-Mechanism is used. All antennas are redundant. The selection of the telecom hardware was driven by the compability with the SSTL platform and its heritage [20].

The link margins were calculated using a slant range of 2200 km, a tracking GS 13 m dish antenna, located close to Longyearbyen in Svalbard with S-Band EIRP of 98 dBm, S-Band G/T of 23 dB/K and X-Band G/T 32 dB/K [21].

5 Development and cost

5.1 Total cost

ADONIS is a two-spacecraft mission which can be expanded with more spacecraft launches in the years following the launch of A-DONIS and B-DONIS. Therefore it is aiming for maximal performance at low cost in order to meet the mission objectives. The total budget allocation for payload, launcher, ground operations and additional infrastrucures is shown in Tab. 5. The expected ground operation

		Size (cm)	Power (W)	Data (bps)	Mass (kg)
Bus	Structure	110x110x100	N/A	N/A	325
	Avionics	35x25x50	40 (61 peak)	N/A	12
	Communication	35x25x50	15 (50 peak)	N/A	10
	Bus Total	N/A	**55 (111 peak)**	N/A	**347**
Payload	INMS	10x10x10	3	2048	3.6
	3DPA	ø25x10	1	58000	1.2
	m-NLP	10x7.5x5	3.5	1900	0.3
	Thermistors	3.3x0.066x0.066	0.01	96	0.036
	CITRIS	40x31x12	12,3	15000	5.4
	IGOR+	21.8x24x14.4	22	20000	6.96
	ISA	3.1x1.7x1.3	12.1	9600	9.78
	FGM	10x10x10	0.8	400	1.8
	Boom	100	N/A	N/A	3
	Payload total	N/A	**55**		**33**
					380 (dry mass)
	Fuel	N/A	N/A	N/A	165
		N/A	**110 (166 peak)**	**107k**	**545 (wet mass)**

Table 4: Mass, size, power and data rate of subsystems. Values given with 20% margin (10% for fuel and avionics).

Item	Cost	Amount	Total cost
Vega launcher	35	1	35
SSTL-300 bus	25	2	50
Customisation	60	1	60
Propulsion	17.5	2	35
Full payload	25	2	50
Ground ops.	45	1	45
Mission cost			275

Table 5: ADONIS mission cost summary (in M€).

cost is 4.05 M€/year. The satellite tracking cost was calculated with 164 €/h of tracking and an additional 59 € per satellite pass with 2·2h/day and 4200 passes/year results in 750 k€/yr. The Mission Control cost is estimated by allocation 600 k€ per 24/7 operation position. With 2–3 operators and additional cost, an overall Mission Control cost of 2000 k€ is expected. Science operations fall with 1000 k€ for 3-4 employees into the budget, while NRT operations require one position and hardware cost of 300 k€/yr.

5.2 Descoping options

During the cost estimation process, the following two descope options could be identified, that still meet most of the mission objectives although being less preferable:

a) Use of a single spacecraft By using one single spacecraft, the overall mission cost would decrease to 165 M€. This decreases the resolution and coverage area by half, and increases the time it takes to make a global map of TEC.

b) Decreasing the mission duration Decreasing the mission duration to 5 years, would result in lower operational costs, therefore summing up the total costs to 251 M€. The disadvantage of this option is the lack of information on the second half of the Solar cylce. Since the cost savings of 14 M€ are negligible compared to the full mission budget, this option appears less cost efficient.

5.3 Mission time line

The next Solar cycle is expected to start in 2019. Due to operational and scientific issues the launch of the ADONIS mission should preferably be at the beginning of the cycle, resulting in a 6-year design and development phase starting from 2013.

5.4 Risks

The ADONIS mission does not show evidence of higher risk than an average Low Earth Orbit mission. Of particular interest are the satellite bus customisation and the plasma analyser which have not yet been space-proven. Here the main risks appear in the interaction with space environment and in combination with other subsystems. An exchange of the proposed bus system to another would increase the overall costs without significant effects in risk prevention. For the plasma analyser and the ISA the use of different standard instrument was considered. Additionally to these particular risks, launch failures and space weather events were identified as risky. Launch failures are part of every space mission but due to the use of the standard launcher Vega not in the range of risk mitigation. Almost all of ADONIS' components are already space proven having thus high TRL, therefore suggesting the identified risk being acceptable.

6 Conclusions

The ADONIS mission presents a cost effective satellite constellation for global monitoring of the ionosphere. The long duration of the mission shall improve our understanding of the space weather impact on satellite drag and telecommunications. Additionally this mission allows for near real-time monitoring with unprecedented altitude resolution of the polar ionosphere, and provides data to improve existing models. This mission could be the first step on building a continuous global ionospheric monitoring system. The mission could further be developed into a network of ionospheric monitoring satellites providing global near real-time data.

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