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A Geostationary Equivalent Polar Observation System

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ABSTRACT

Various potential polar observation systems are evaluated and compared against a set of requirements to develop a system capable of providing geostationary equivalent coverage of the Earth's high-latitude regions. Consideration is given to Non-Keplerian orbits, where it is found that the orbit altitudes are restrictively high, and traditional highly-elliptical Molniya orbits where it is found that no single spacecraft can provide observations 'over the pole' to 55 degrees latitude of equal quality to those produced by geostationary systems. Subsequent analysis of the Taranis concept reveals the ability of these systems to adequately view the target region with a reduction in the required number of spacecraft. Mass budget analysis reveals a possible payload mass of 151 – 379 kg for a single platform launched using the Soyuz launcher to a 16 hour, 90 degree inclination Taranis orbit for an 8.5 year mission. Four spacecraft, capable of observing continuously to 55 degrees latitude using single imagery, can be launched on the Ariane 5, which results in a payload mass range of 193 – 482 kg for an 8.5 year mission.

KEYWORDS: [Polar, Earth observation, Highly-elliptical orbit, Electric propulsion, Taranis]

INTRODUCTION

The considerable environmental change occurring in the Arctic together with the concept of 'Arctic Amplification' and the significant effect this has had on the terrestrial and marine ecosystems is widely accepted [1]. However, the impact of these changes within the cryosphere on northern and mid-latitudes, and the broader global climate remains unclear [2-7]. These uncertainties, along with the potentially global impact accentuate the current disparity between the breadth and depth of observations available for the tropics and mid-latitudes with the lack of geostationary-equivalent observations over the polar regions.

Observations from geostationary orbits (GEO) are critically limited beyond around 55 degrees latitude due to the oblique viewing geometry [8], as illustrated in Figure 1. This gives the View Zenith Angle (VZA) contours from nine spacecraft in GEO, namely GOES-12, -13, -15, Elektro-L, MTSAT-1R, and Meteosat-6, -7, -8, -9, and clearly shows the inability of spacecraft in GEO to sufficiently observe beyond around 55 degrees latitude with sufficient quality.

Spacecraft in low Earth orbits (LEO) are unable to provide the large-scale contextual information available from GEO, therefore a significantly large number would be required to provide continuous or near-continuous imagery. The gap in data for the high-latitude regions can be partially resolved, for certain data products, using composite images from spacecraft in LEO and GEO. However, the time delay in creating these images makes them impractical for now-casting applications such as meteorology. Consequently, there is currently no source of high-resolution temporal data for the polar regions, resulting in a lack of critical data sets such as, for example, the retrieval of atmospheric motion vectors (AMV) and bi-directional reflectance distribution functions (BDRF). It is therefore necessary to significantly improve monitoring to reduce the existing uncertainties in the polar regions.

A geostationary-like polar observing system would radically improve key polar observations, including resolution of the diurnal cycle of phenomena related to, amongst other things, winds, clouds, sea ice, snow cover, and surface temperature of ice, sea and land. Analysis of such phenomena will enable improved weather forecasting and modeling and will significantly improve monitoring of 'Essential Climate Variables' (ECVs) to enhance understanding of climate change and modeling of associated feedback processes.

This paper makes a comparison between proposed polar observation systems and evaluates the concepts against a set of defined requirements. Further analysis is then conducted on a newly developed highly-elliptical orbit concept, termed "Taranis", which uses low-thrust propulsion to maintain a Keplerian orbit away from the natural critical-inclination [9, 10]. Mission analysis is then conducted, and possible payload mass determined, considering various launch vehicles.

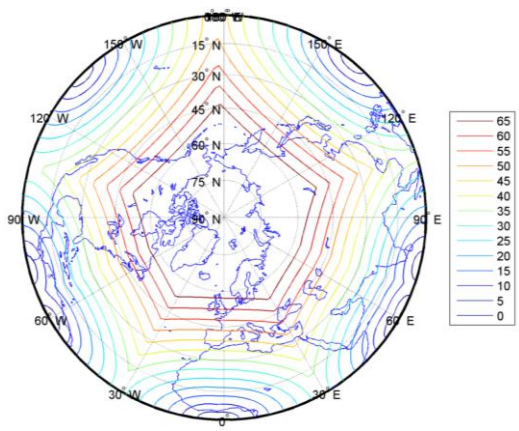


Figure 1 View zenith angle contours of nine spacecraft in geostationary orbits (contours show view angle measured in degrees)

REQUIREMENTS FOR A GEOSTATIONARY EQUIVALENT POLAR OBSERVATION SYSTEM

Requirement 1: *Spacecraft shall be able to continuously observe all longitudes at latitudes between 55 and 90 degrees with a VZA of less than 60 degrees*

This is the minimum level of coverage required by the constellation as at around 55 degrees latitude observations from GEO platforms become less reliable for many data products [8]. The VZA limit also matches that of GEO systems to 55 degrees latitude, ensuring data of equal quality to equatorial regions.

Requirement 2: *The orbit altitude shall be less than 45000 km*

In order to ensure adequate GEO-like spatial resolution from instruments of similar size and scale, the limit placed on the apogee altitude is selected as 45000 km (*25%* higher than GEO altitude), matching the limit currently being used by other polar mission concepts including the Polar Communications and Weather (PCW) mission [11, 12].

POLAR OBSERVATION SYSTEMS

Exploiting the Three-Body Problem

Recent investigations have been conducted into orbits enabled by exploitation of the three-body problem. Specifically to highly non-Keplerian orbits [13], such as the Polesitter mission concept, first introduced in 1980 to allow continuous high-latitude observation [14]. This concept involves placing a spacecraft above one of the Earth's poles at around 2 - 3 million kilometers altitude, by means of continuous acceleration provided by a solar sail, electric propulsion (EP) system or combination of the two, to allow near-hemispheric observations [15, 16]. It has been shown that a payload of order 100 kg can be hosted for approximately four years using EP and up to seven years with both a solar sail and EP system [17].

Similarly, natural and solar sail displaced eight-shaped orbits are families of periodic orbits connected to the L1 and L2 Lagrangian points of the circular restricted three-body problem, and have recently been considered for polar observation [18]. The propulsion requirement for these orbits is minimal as acceleration is only required for station-keeping of the spacecraft. However, as with the Polesitter concept, the spacecraft is several million kilometers from Earth thus neither concept satisfies requirement 2.

Highly-Elliptical Orbits

To counteract the lack of adequate observations of key-high latitude regions, the World Meteorological Organization (WMO) endorsed the use of highly-elliptical orbits (HEO) *'for quasi-permanent monitoring of high-latitudes and polar regions'* [19]. HEOs, such as the Molniya and Tundra orbits, have been used extensively by the Russian Federation for high-latitude communication since 1965[20].

The Molniya orbit has an apogee altitude comparable to the altitude of GEO platforms, a period equal to one-half of a sidereal day, and inclination equal to a 'critical' value. The change in argument of perigee over one orbit revolution is given in Eq. (1)

$$(\Delta\omega)_0^{2\pi} = \frac{3J_2R_E^2(3+5\mathrm{Cos}(2i))}{4a^2(-1+e^2)^2} \qquad (1)$$

where, R_E is the mean radius of the Earth equal to 6371 km, i is the inclination of the orbit, a is the orbit semi-major axis and e is the eccentricity of the orbit. The value of the critical inclination is then derived by setting Eq. (1) equal to zero and solving for the inclination, giving a value of 63.4 degrees, independent of the orbit semi-major axis or eccentricity. These

critically inclined HEOs have been examined in [21] for high-latitude communications.

The highly elliptical geosynchronous Tundra orbit also has an inclination of 63.4 degrees although the semi-major axis is greater than that of the Molniya orbit making the orbital period equal to one solar day. The Tundra orbit has previously been considered as an alternative to GEO [22], however to date the only use has been the Sirius Satellite Radio [23].

Subsequent to [19], an EO system in a HEO was studied for launch by 2017 [11]. The PCW mission proposes two spacecraft in 16 hour Molniya-like orbits in a single orbit plane to provide observations of regions of Canadian interest, with VZAs not exceeding 70 degrees, and to provide broadband services [11, 12, 24]. The Russian Federation has also proposed a more comprehensive arctic observation system known as 'Arktika' which consists of spacecraft in GEO, LEO and HEOs to obtain real-time information such as wind velocity and direction, precipitation and ice conditions for weather forecasting, flight safety and navigation in the high-latitude regions [25].

Although Molniya orbits can provide enhanced high-latitude observation, they cannot comply with requirement 1, using a single spacecraft. A spacecraft at apogee on a Molniya orbit observing to 55 degrees latitude has a peak VZA around 10 degrees higher than when the same location is observed from GEO. Therefore using traditional Molniya orbits coverage would continue to be dependent on composite images, which may be discontinuous in viewing geometry and take time to compile, offering less value for now-casting applications such as meteorological services.

Recent studies have also considered modifications to the Molniya and Tundra orbits for high latitude applications, such as a Polar Tundra orbit with station-keeping performed using impulsive maneuvers [26]. Similarly, in [27], HEOs with inclinations other than the critical value are considered to provide high-latitude communications, although the conclusion of the study is that inclinations greater than 63.43 degrees are only realistic for orbits with 24 hour period.

Taranis Orbits

Taranis is the term used to denote a series of HEOs that can be derived through the application of acceleration, provided by a propulsion system, to allow free selection of the orbit period and inclination. As discussed, from Eq. (1), to negate the secular drift in the argument of perigee, caused by the non-spherical nature of Earth, the inclination of the Molniya orbit is limited to 63.4 degrees. This fixed critical inclination limits the possible applications of HEOs and the remote sensing opportunities available from them. The propulsion system used to enable the Taranis orbit is therefore nulling the dominant perturbation acceleration vector, due to the non-spherical shape of Earth, acting on the spacecraft; accordingly, a low-thrust propulsion system is sufficient as the magnitude of this natural perturbation is of similar magnitude to such systems [28-30]. Thereafter, the orbit inclination and period can be chosen to optimally fulfill the mission objectives [9]. For example, for high-latutude observation an inclination of 90 degrees is considered. Combined with an argument of perigee of 270 degrees, this places the orbit apogee directly above the North pole. The process of obtaining the total acceleration required to achieve this orbit is detailed in [9, 10], where, it is shown that the acceleration should be directed in the radial and transverse, R & T, directions.

Numerical analysis is used to calculate the time spacecraft on 90 degree inclination Taranis orbits of varying periods can view to 55 degrees latitude and thus determine the number of spacecraft required to achieve continuous observation above this latitude limit. It should be noted that visibility analysis is conducted assuming each individual spacecraft can view the entire region of interest, i.e. using non-composite imagery. The results of this process are plotted in Figure 2 alongside the apogee altitude limit, derived from requirement 2, and a limit placed on the semi-latus rectum of 15000 km to minimize the effect of radiation from high energy protons which can be extremely damaging.

A 16 hour orbit is consequently selected as the most beneficial, as this allows a repeat ground-track in two days and allows continuous observation using a minimum of four spacecraft. As the perigee altitude increases, the radiation from high-energy protons decreases due to the corresponding increase in semi-latus rectum. Figure 3 illustrates the results for 5 year missions where it is shown that the total proton flux for the 16 hour orbit with a perigee altitude of 8000 km is almost completely absorbed with approximately 7 mm of aluminum shielding; this is compared with only 2 mm for a perigee altitude of 10000 km. However, as perigee altitude increases, apogee altitude decreases, thus decreasing eccentricity and the time above high-latitude regions. Therefore such orbits tend to require a greater number of spacecraft to provide continuous observation, as seen in Figure 2, where an 11000 km perigee altitude on a 16 hour orbit is shown to require an additional spacecraft in comparison to a 10000 km perigee altitude. A 16 hour Taranis orbit with perigee altitude of 10000 km and apogee altitude of 41740 km is therefore selected as the most beneficial for high

latitude observation when taking into consideration both the radiation dose and apogee altitude constraints.

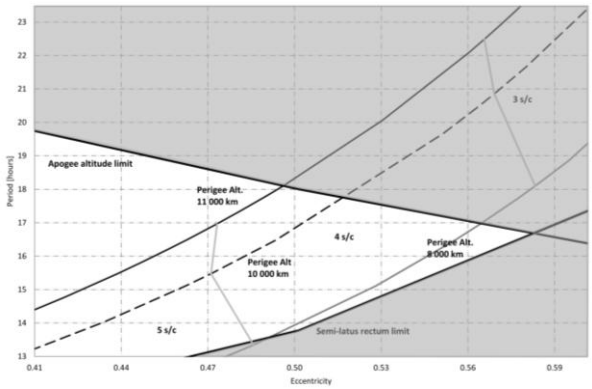


Figure 2 Observation to 55 degrees showing apogee-altitude limit, semi-latus rectum limit and required number of spacecraft.

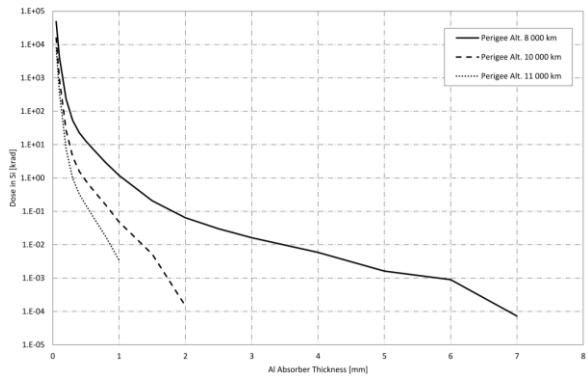


Figure 3 Comparison of trapped proton dose for 16 hour Taranis orbits of varying perigee altitude with mission durations of 5 years, found using SPENVIS.

Lowering the perigee of the orbit will increase the radiation dose from high-energy protons but reduce the number of spacecraft required. Visibility analysis is therefore once again conducted to 55 degrees, in this case simply placing a restriction on the maximum apogee altitude of the orbit to determine the number of spacecraft required; the results of this process for orbits with perigee altitudes of 300, 1000 and 2000 km are shown in Figure 4. It is clear that three spacecraft on a 12 hour orbit can provide continuous observation to 55 degrees latitude.

Subsequent selection of the appropriate perigee altitude for the 12 hour orbit is conducted by considering debris mitigation guidelines. At EoL the spacecraft can either be re-orbited to an orbit with perigee altitude > 2000 km, or de-orbited to an orbit where perigee altitude is ≤ 300 km where it will naturally degrade due to atmospheric drag and re-enter the Earth's atmosphere within 25 years. If a perigee altitude of 2000 km is selected, such that a perigee re-orbit maneuver is avoided, the corresponding apogee altitude for a 12 hour orbit is 38500 km. If at EoL the spacecraft is simply decommissioned in this orbit, the orbit argument of perigee will drift around the orbit plane as the continuous low-thrust previously preventing this is removed; over long periods this may result in the orbit apogee intersecting the GEO ring, thus violating debris mitigation guidelines. This therefore drives the orbit selection towards an orbit with a lower perigee altitude, such that the orbit decay occurs prior to the intersection of apogee with the GEO ring. Numerical simulations reveal that spacecraft on 12 hour orbits with perigee altitudes higher than approximately 300 km will require an EoL de-orbit maneuver to reduce the perigee altitude to this value such that re-entry to the Earth's atmosphere occurs within 25 years. A 12 hour orbit with perigee altitude of 300 km is therefore selected for consideration.

Finally, a third orbit is selected for further evaluation. This is a 10 hour orbit, with a perigee altitude selected as 2000 km, to avoid an EoL maneuver, and apogee altitude of 32400 km.

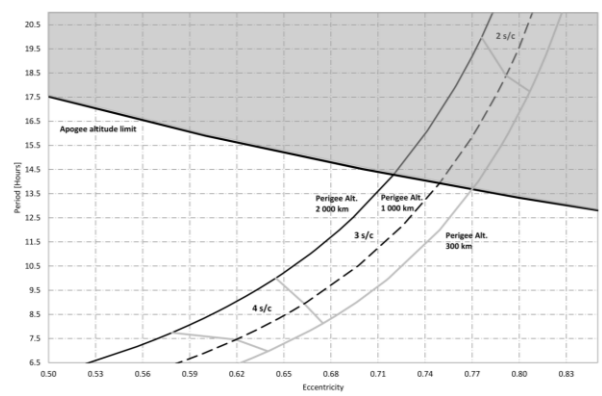


Figure 4 Observation to 55 degrees showing apogee-altitude limit and required number of spacecraft.

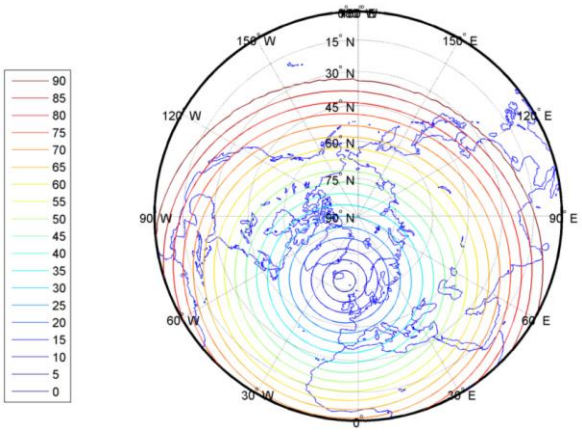
POLAR OBSERVATION SYSTEMS COMPARISON

It is clear that due to the significantly high altitude of spacecraft on Polesitter and eight-shaped orbits, these concepts do not fulfill requirement 2. Comparison is therefore made of the level of coverage available from Molniya and Taranis concepts, with inclinations of 63.43 and 90 degrees respectively, considering both composite and single-image coverage.

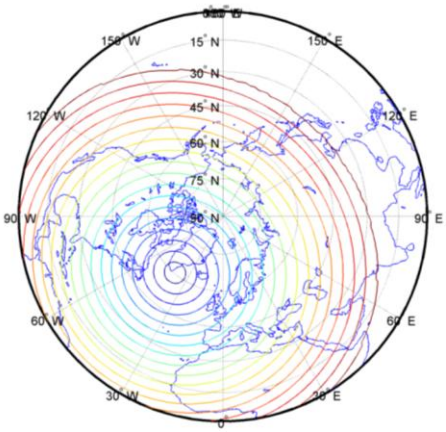
The inability of spacecraft on a 12 hour Molniya orbit to view 'over' the pole to 55 degrees latitude with a VZA sufficient to match that of a spacecraft in GEO is illustrated in Figure 5. This shows the VZA during the best and worst case observations i.e. when the

spacecraft is at apogee and when it is entering the observation window (minus 4 hours from apogee). In the best case the view angle at 55 degrees latitude is around 70 degrees and in the worst case is around 75 degrees, therefore around 10 and 15 degrees higher respectively than that of GEO spacecraft at 55 degrees latitude.

On the other hand, the ability of a 12 hour Taranis orbit to meet requirements 1 and 2 and therefore fully overcome the high-latitude data deficit is shown in Figure 6. At 55 degrees latitude the VZAs are 35 and 55 degrees for the best and worst Taranis cases respectively. Subsequent analysis has revealed that in order to provide continuous coverage beyond 55 degrees latitude, using single imagery, three spacecraft on a single plane of a 12 hour Taranis orbit are required.

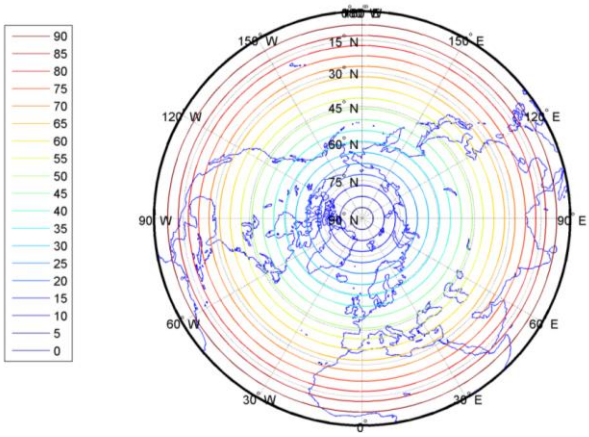


Best Case

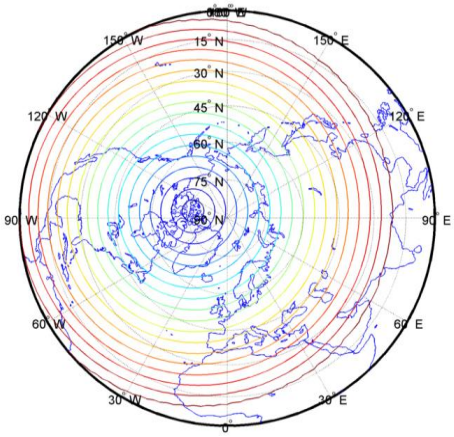


Worst Case

Figure 5 VZA contours of a spacecraft on a 12 hour Molniya orbit. Best case – altitude 40082 km, latitude 64 degrees. Worst case – altitude 36477 km, latitude 61 degrees.



Best Case



Worst Case

Figure 6 VZA contours of a spacecraft on a 12 hour Taranis orbit. Best case – altitude 40170 km, latitude 90 degrees. Worst case – altitude 36750 km, latitude 77 degrees.

If composite images are deemed to be acceptable the Taranis orbit is shown to offer benefits over the Molniya orbit in terms of number of spacecraft required to achieve continuous coverage. A constellation of three spacecraft on a 12 hour Molniya orbit, separated at four hour intervals around the orbit, and on three planes 120 degrees apart can provide continuous coverage to 55 degrees latitude. In this scenario two spacecraft simultaneously image the desired region while the third spacecraft is at perigee. The polar stereographic plot showing the worst case VZAs from two spacecraft, one entering and one leaving the observation region are given in Figure 7, where it is clear that the view angle at 55 degrees latitude is reduced to around 45 degrees. The observation from this constellation is further illustrated by outputs from

satellite visualization software SaVi[1] in Figure 8. This shows the coverage at 4, 8 and 12 hours through the orbit propagation, green regions are those in view of one spacecraft and darker green regions are those in view of two spacecraft. The smaller circles represent the spacecraft at perigee at the time stated.

However, as shown in Figure 9 and Figure 10, two spacecraft separated by 6 hours on a single plane of a 12 hour Taranis orbit can provide continuous observation to 55 degrees latitude. Figure 9 represents the worst case, where one spacecraft is entering and one spacecraft is leaving the observation region and shows that the VZA at 55 degrees latitude is around 37 degrees. The benefit of the Taranis concept over the traditional Molniya orbit is therefore clear.

It should be noted that although the results shown here are for 12 hour orbits, the same conclusions can be drawn for 16 hour orbits. Four spacecraft are required on a single plane of a 16 hour Taranis orbit to provide continuous coverage to 55 degrees latitude, using single imagery. This is not possible using a 16 hour Molniya orbit due to the inadequate VZA. Two spacecraft on a single plane of a 16 hour Taranis orbit can provide continuous coverage using composite images, while four spacecraft are required on four orbit planes of a Molniya orbit.

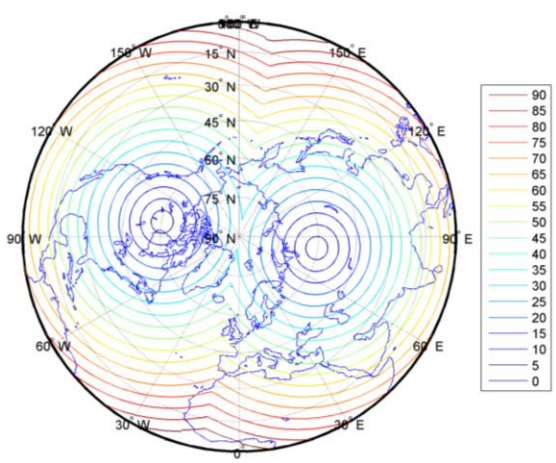


Figure 7 VZA contours of two spacecraft on a Molniya orbit (three spacecraft constellation on three orbit planes). Worst case observation when one spacecraft is entering the observation region and one is leaving.

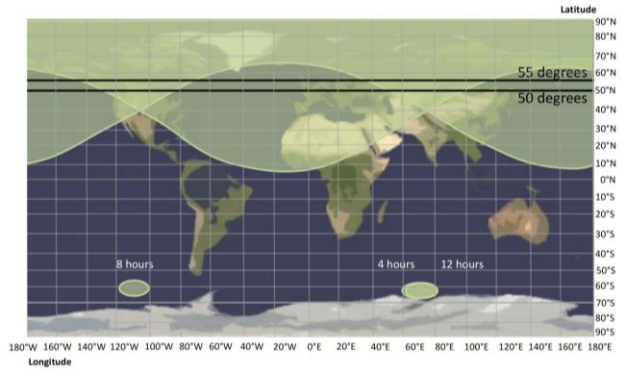


Figure 8 Coverage from three spacecraft on three orbit planes of the Molniya orbit. Green areas represent the regions in view of one spacecraft, with elevation greater than 27 degrees, and darker green regions are in view of more than one spacecraft.

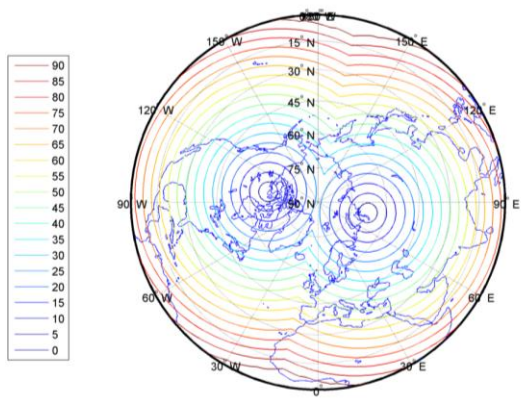
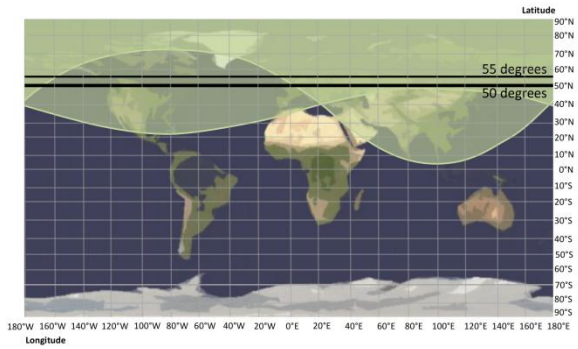
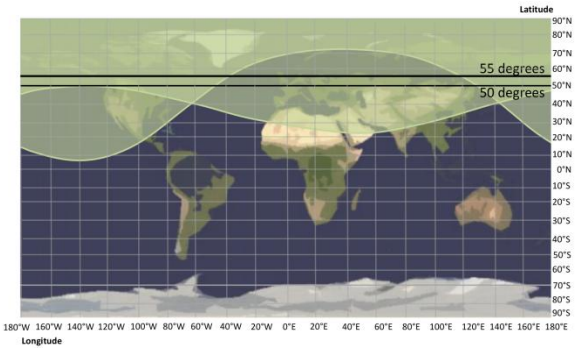


Figure 9 VZA contours of two spacecraft on a Taranis orbit (two spacecraft, single orbit plane).



(a)

[1]http://personal.ee.surrey.ac.uk/Personal/L.Wood/software/SaVi/



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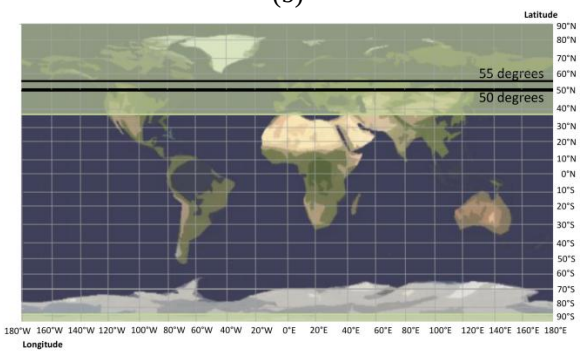


(c)

Figure 10 Coverage from two spacecraft on single plane of 12 hour Taranis orbit at (a) 4 hours (b) 8 hours and (c) 12 hours around orbit. Green areas represent the regions in view of one spacecraft, with elevation greater than 27 degrees, and darker green regions are in view of more than one spacecraft.

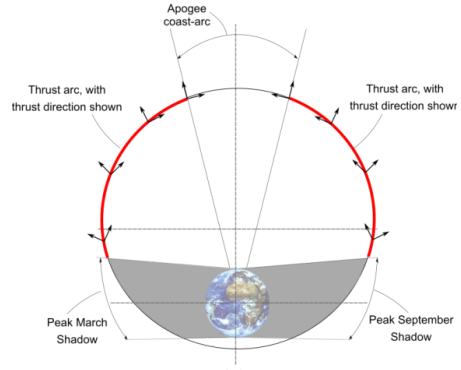
TARANIS MISSION ANALYSIS

Orbit Analysis

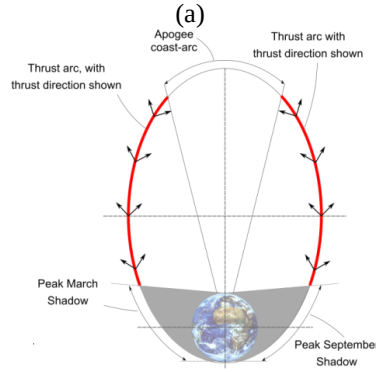
Previous sections of this paper have identified three potential Taranis systems for further consideration, which are capable of observing continuously to 55 degrees latitude with maximum VZAs equal to those from GEO altitude. These are

- four spacecraft on a 16 hour, 10000 x 41740 km altitude, low-radiation orbit;
- three spacecraft on a 12 hour, 300 x 40170 km altitude, high-radiation orbit; and
- three spacecraft on a 10 hour, 2000 x 32400 km altitude orbit, no de-orbit required.

The selected orbit architectures are presented in Figure 11, showing shadow and coast-arc data.



(a)



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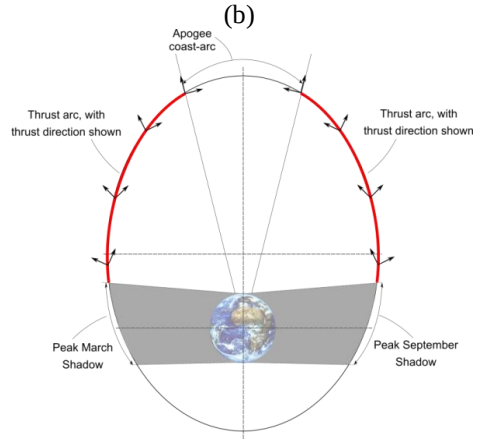

(c)

Figure 11 (a) 10 hour orbit (b) 12 hour orbit (c) 16 hour orbit

In order to mitigate potential interference from the EP system required to enable Taranis orbits with science operations, to provide a power rich environment for the science suite, and to mitigate against battery mass concerns, coast arcs are included about apogee, perigee and through possible Earth shadow regions. Shadow regions are said to occur at 245 and 115 degrees, corresponding to the peak March and September shadows respectively. For the 16 and 12 hour orbits, the minimum apogee coast arc occurring at the beginning-of-life (BoL) is 4 hours, which equate to true anomaly ranges of ±18.5 and ±13.7 degrees about apogee respectively. The length of this coast arc will increase

as the mission progresses such as to maintain a constant orbit averaged acceleration magnitude due to the reduction in spacecraft mass incurred with the consumption of propellant. Similarly, for the 10 hour orbit the BoL apogee coast arc is 3 1/3 hours, equating to a true anomaly range of ±17.95 degrees about apogee. Table 1 presents the summary of the 10, 12 and 16 orbit architectures which includes; the required acceleration to allow coast-arcs at apogee, perigee and through possible Earth shadow regions and the variation in altitude and apparent diameter of Earth through apogee.

Table 1 Summary of mission architectures

	10 hr	12 hr	16 hr
Perigee Altitude [km]	2000	300	10000
Apogee Altitude [km]	32400	40170	41740
Required no. of s/c (no composite images)	3	3	4
Initial acceleration magnitude per R & T direction through thrust arcs [mm s^{-2}]	0.129	0.109	0.012
Altitude range through apogee [km]	29335–32400	36750–40170	39575–41740
Variation in apparent diameter of Earth	12.25 – 11.12° (1.13°)	9.84 – 9.01 ° (0.82°)	9.15 – 8.68 ° (0.47°)

Launch Analysis

As the orbits considered are 'non-standard', launcher user manuals do not detail the mass which can be delivered to these orbits. Analysis is presented to determine the delivery mass using the Ariane 5 ES and Soyuz launchers as representative vehicles.

The spacecraft is delivered to the 12 hour, 300 x 40170 km, orbit via a circular 300 km intermediate orbit. The mass that can be delivered to the final orbit is found by subtracting the dry mass of the upper stage (4.5 tons for the Ariane 5[31]), and the propellant mass required to insert the payload into the target orbit from the initial mass (23 tons for the Ariane 5). The propellant mass is calculated using

$$m_{prop} = m_0 \left(1 - Exp\left[-\frac{\Delta V_{insert}}{I_{sp} g_0}\right]\right) \qquad (2)$$

where, m_0 is the initial mass, I_{sp} is the specific impulse, equal to 325 s (for the Ariane 5), g_0 is standard gravity and ΔV_{insert} is the difference between the velocity at perigee on the target orbit and the velocity at the intermediate orbit, found using

$$V = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}. \qquad (3)$$

For the Ariane 5, excluding a de-orbit manoeuvre for the upper stage, this results in a ΔV_{insert} equal to 2492 m s^{-1}, a propellant mass of 12476 kg and a delivered spacecraft mass to the target orbit of 6042 kg. The same process is conducted for the 10 and 16 hour orbits, resulting in a delivered spacecraft mass of 5869 and 4095 kg respectively to the target orbit. These results are summarized in Table 2. Similarly, considering launch using the Soyuz rocket with a Fregat upper stage, from Kourou, [32] the payload is delivered to the target orbit via a 90 degree circular intermediate orbit with an altitude of 400 km; due to the Soyuz launch profile. The resulting payload delivered to the 10 hour orbit is 1226 kg, to the 12 hour orbit is 1250 kg and to the 16 hour orbit is 859 kg. These results are summarized in Table 3.

Table 2 Summary of launch mass allocation using the Ariane 5.

	10 hr	12 hr	16 hr
Dry mass of upper stage [kg]	4500		
Specific impulse of upper stage [s]	325		
Δ*V* required to attain target orbit [m s^{-1}]	2539	2492	3137
Required launch vehicle propellant mass [kg]	12631	12476	14405
Delivered mass to target orbit [kg]	5869	6024	4095

Table 3 Summary of launch mass allocation using the Soyuz.

	10 hr	12 hr	16 hr
Dry mass of upper stage [kg]	950		
Specific impulse of upper stage [s]	331		
Δ*V* required to attain target orbit [m s^{-1}]	2499	2464	3100
Required launch vehicle propellant mass [kg]	2524	2500	2891
Delivered mass to target orbit [kg]	1226	1250	859

Mass Budget Analysis

Feasible payload masses are determined for each system, based on the delivered mass to the target orbits considering multiple spacecraft per launch (three spacecraft for 10 and 12 hour orbits and four spacecraft

on the 16 hour orbit) from the Ariane 5, and single spacecraft launch from the Soyuz.

The mass of the required propellant over the total thrust duration, Δt is found using

$$m_{prop} = \frac{T}{I_{sp} g_0} \Delta t \tag{4}$$

where T is the constant thrust required. The mass of the propellant tanks is estimated as 10% of the total propellant mass [33],

$$m_{tank} = \frac{m_{prop}}{10}. \tag{5}$$

The associated structure mass to support the tank can be estimated as [33],

$$m_{support} = \frac{m_{tank}}{10} \tag{6}$$

Finally, the mass of the EP system m_{EP}, is a function of the peak power demand of the system

$$m_{EP} = k_{EP} P_{max} \tag{7}$$

Where k_{EP} is the specific performance of the EP system and P_{max} is the peak power demand of the EP system.

Multiple Spacecraft Launch

Payload mass analysis is firstly conducted assuming that all spacecraft required to provide continuous observation are launched together on one Ariane 5 vehicle, that is to say three spacecraft for the 10 and 12 hour orbits and four spacecraft for the 16 hour orbit. The corresponding wet mass of each spacecraft is given in Table 4 in addition to the selection of the EP system, based on the thrust magnitude. The mass breakdown for each spacecraft on the 10, 12 and 16 hour orbits are given in Table 5. Following deduction of the propellant, EP system, support and tank mass, the available instrument mass is estimated as 20, 30, 50% of the remaining mass [34]. The propellant mass for the 12 hour orbit includes the propellant to maintain the orbit due to atmospheric drag effects and for the 16 hour orbit the propellant mass includes that required for an EoL de-orbit maneuver to reduce the apogee altitude below that of GEO. The subsequent payload mass values, for mission durations between 1 and 10 years, are presented in Figure 12, where it is clear that for a 8.5 year mission significant payload allocations, 81 – 203 kg for the 10 hour orbit, 32 – 81 kg for the 12 hour orbit, and 170 – 426 kg for the 16 hour orbit, are achievable with launch on the Ariane 5 ES launcher.

Table 4 Thruster data for multiple spacecraft launch on Ariane 5

	10 hr	12 hr	16 hr
Total wet mass, m_{total} [kg]	1956	2008	1024
Instantaneous thrust magnitude per R & T direction through thrust arcs [mN]	177.86	218.87	12.18
Thruster type	T6	T6	T5
Max. power per thruster, P_{max} [kW]	4.5	4.5	0.3
Specific impulse, I_{sp} [s]	4600	4600	3500
Specific performance, k_{EP} [kg kW^{-1}]	55	55	55

Table 5 Spacecraft mass breakdown for multiple spacecraft launch on Ariane 5

	10 hr	12 hr	16 hr
Thruster mass, m_{EP} [kg]	495	495	33
Propellant mass range (1-10 years), m_{prop} [kg]	133 – 992	175 – 1250	30 – 133
Tank mass range (1-10 years), m_{tank} [kg]	13 – 99	17 – 125	3 – 13
Support mass range (1-10 years), $m_{support}$ [kg]	13 – 99	17 – 125	3 – 13

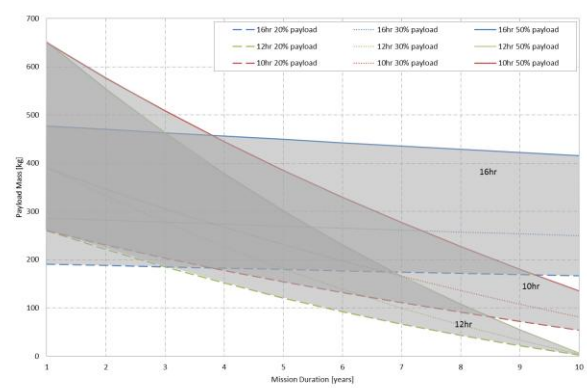

Figure 12 Spacecraft mass allocation over a range of mission lifetimes for 10, 12 and 16 hour orbits for multiple spacecraft launch from the Ariane 5 ES launcher.

Single Spacecraft Launch

Consideration is also given to launch of a single spacecraft on the Soyuz vehicle, where the total wet mass of each spacecraft is given as the delivered mass to the target orbit, as stated in Table 6. The payload masses are again calculated and plotted as a function of mission lifetime in Figure 13, where it is evident that significant payload masses can be obtained with 58 – 145 kg for the 10 hour orbit and 142 – 356 kg for the 16 hour orbit, for 8.5 year missions. However, the possible payload capacity is significantly lower for the 12 hour

orbit, with only 26 – 66 kg available for a mission of 8.5 years.

Table 6 Spacecraft mass breakdown for single spacecraft launch on Soyuz Fregat

	10 hr	12 hr	16 hr
Total wet mass, m_{total} [kg]	1226	1250	859
Instantaneous thrust magnitude per R & T direction through thrust arc [mN]	111.46	136.63	10.30
Thruster type	Astrium RIT-XT	Astrium RIT-XT	T5
Max. power per thruster, P_{max} [kW]	2.4	2.4	0.3
Specific impulse, I_{sp} [s]	4500	4500	3500
Thruster mass, m_{EP} [kg]	264	264	33
Propellant mass range (1-10 years), m_{prop} [kg]	85 – 632	111 – 789	26 – 111
Tank mass range (1-10 years), m_{tank} [kg]	9 – 63	11 – 79	3 – 11
Support mass range (1-10 years), $m_{support}$ [kg]	9 – 63	11 – 79	3 – 11

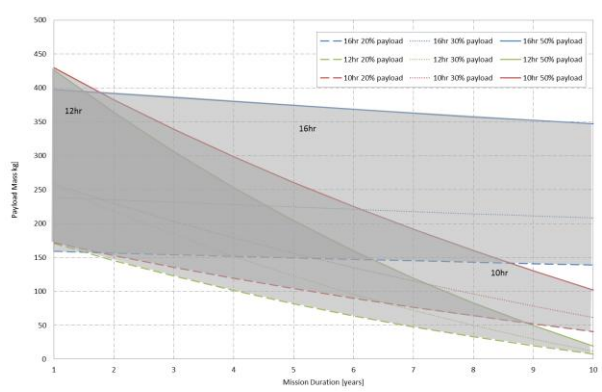

Figure 13 Spacecraft mass allocation over a range of mission lifetimes for 10, 12 and 16 hour orbits for single spacecraft launch from the Soyuz launcher.

Varying Inclination

In order to reduce the orbital inclination, and hence the required acceleration, visibility analysis has been conducted to quantify the level of coverage from spacecraft on 12 and 16 hour Taranis orbits with varying inclinations. Throughout this analysis, the maximum VZAs obtained from GEO are maintained. The level of coverage achievable from a single spacecraft on 12 hour Taranis orbits of varying inclination is shown in Figure 14. It is clear that three spacecraft on a 12 hour orbit with a minimum inclination of 76 degrees can provide complete coverage to 55 degrees latitude. This allows a 30% reduction in initial acceleration magnitude from 0.154 mm s^{-1}, for an inclination of 90 degrees, to 0.109 mm s^{-1}, for an inclination of 76 degrees. It should be noted that the curves shown in Figure 14 and Figure 15 are not smooth due to the fidelity of the data used to produce them.

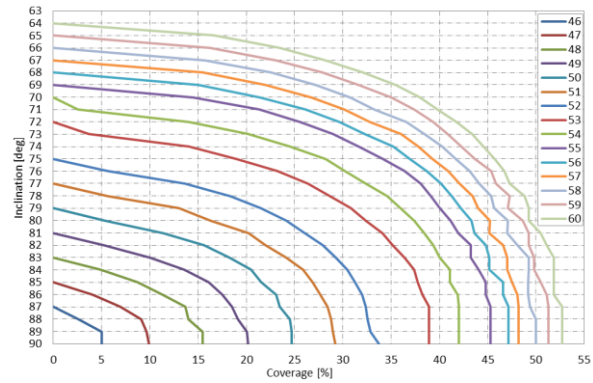

Figure 14 Coverage from one spacecraft on 12 hour Taranis orbits of varying inclination – contours are lines of latitude

Figure 15 presents the percentage coverage from a single spacecraft on 16 hour Taranis orbits of varying inclination. In this case complete coverage to 55 degrees latitude can be achieved using four spacecraft with a minimum inclination of 84 degrees. This gives a marginal reduction in the required acceleration magnitude 0.017 mm s^{-1}, for a 90 degree orbit, to 0.016 mm s^{-1} for an inclination of 84 degrees.

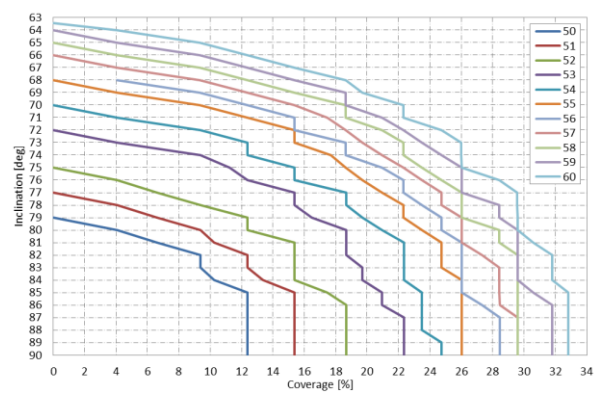

Figure 15 Coverage from one spacecraft on 16 hour Taranis orbits of varying inclination – contours are lines of latitude

POSSIBLE TARANIS INSTRUMENTATION

One of the key advantages of a Taranis orbit is the high level of electrical power available to the instrument payload. Due to the duration of the orbital maintenance thrusting required by the EP system, the total power generation would be in the lower kW range. In addition, this power would only be used by the EP system during orbital thrust-arcs, and not during the most useful apogee portion of the orbit, when directly over the polar-regions.

Based on an available mass of around 100 – 250 kg, a number of compelling instrument packages could be envisioned to exploit the unique features of the Taranis orbit. The most likely categories would be meteorological multi-band imaging, space weather monitoring, and communications.

Meteorological imaging can aid in the acquisition of a number of critical lower atmospheric parameters, (e.g. cloud-motion wind vectors), and form a vital input to weather-forecasting systems. Such contextual and co-temporal imaging would give, for the first time, GEO like data at the poles. The INSAT-2/3 multi-band imager is a good example of what a low-mass atmospheric imager from GEO like altitudes can achieve. Within the constraints of 55 kg and 50 W, the imager uses a gimbaled scan mirror to sweep the FOV in VIS (Visible), VNIR (Visible Near Infrared), and SWIR (Shortwave Infrared) bands. The SWIR band in particular is useful for snow-cover and snow-cloud discrimination and aerosol measurement. In addition, INSAT 3 carries a 90 kg/100 W infrared sounder to measure the temperature and humidity profiles thus obtaining three-dimensional representations of the atmosphere. In terms of capability versus mass/power, these are the types of metrological sensing instruments that could be ideal for a Taranis orbit with minimal changes.

Space weather, particularly how significant events from the Sun impact the Earth's upper atmosphere, is a key area of increasing interest, not only to satellite health, but also to aviation. As more commercial aviation routes fly over the Arctic, monitoring of the changes of radiation in these regions becomes more important. Space weather instrumentation generally have low mass/power requirements (often <10 kg/10 W), and as such a suite of instruments that require GEO-like altitudes (e.g. Auroral imaging, UV imaging, ionosondes, high-energy particle detectors etc.) would be ideal for exploiting Taranis orbits. In addition, the changing altitude of the orbit while traversing from perigee to apogee allows for unique cross-sectional measurements to be taken.

As the polar sea and air traffic increases with time, reliable communication becomes not only desired but often required by law. Indeed, the ESA ArticCOM study results revealed a severe lack of communication capacity for all Arctic regions [35]. Specific communication payloads are customizable to the available payload mass and power limits, and depending on the levels and type of service needing to be provided, would operate well in high-altitude polar orbits. Even limited bandwidth communications for emergency Search And Rescue (SAR) purposes (e.g. Cospas-Sarsat network, Emergency Position-Indicating Radio Beacon EPIRB, etc.) would be of great benefit to the Arctic region as at present no reliable communications exist in these areas.

Based on a set of scientific requirements derived in consultation with the user community, and given in Table 7, the most likely instrument is a visible/infra-red imager based on heritage from the Flexible Combined Imager (FCI) under development for the Meteosat Third Generation Platforms. Analysis of the FCI revealed that a mass of 250 kg is achievable by reducing the spectral channels available and using a lightened structure. This is therefore the instrument the subsequent mass analysis is based on.

Table 7 Scientific requirements

Number	Requirement
SR-1	Satellite-derived Atmospheric Wind Vectors from cloud-feature tracking available over the full polar disk with at least hourly temporal resolution.
SR-2	Satellite-derived surface albedo of ice and snow surfaces with at least monthly resolution.
SR-3	Satellite-derived sea surface temperature, land surface temperature and ice surface temperature with at least hourly resolution under clear skies.
SR-4	Satellite-derived aerosol optical depth and aerosol class above all surfaces (water, ice land and snow).
SR-5	Satellite-derived surface solar irradiance derived from reflectance imagery with at least 15 minute sampling (to match the accuracy available from SEVIRI).
SR-6	Observation of mid/upper tropospheric humidity at "water vapor" thermal wavelengths across the target region, suitable also for supporting humidity-based AMVs.
SR-7	Simultaneous image acquisition with significant overlap of coverage around the times of handover between prime platforms, to support rigorous inter-calibration and applications benefiting from "dual view".

DOWN-SELECTION OF TARANIS MISSION CONCEPTS

Following the initial Strawman mass budget, presented previously, a small number of mission concepts are considered for more detailed analysis. From Figure 12 and Figure 13 it is clear that the 12 hour orbit offers no

benefit over the 10 hour orbit in terms of available payload mass and EoL spacecraft disposal, thus the 12 hour orbit is not considered further. Furthermore, due to the assumption that launch will be conducted using a single spacecraft on a Soyuz vehicle and multiple spacecraft on an Ariane 5, launch costs will always be the same for the 10 and 16 hour orbits. Therefore due to the higher available payload mass for the same launch cost, the 16 hour orbit is considered to be the most beneficial.

Although details have been given of the possible reduction in acceleration that can be gained for lower orbital inclinations, analysis has revealed that the resulting propellant saving over the mission duration is relatively small. Therefore orbital inclinations of 90 degrees are considered herein.

Two mission concepts are selected for further consideration, one using single imagery and one using composite imagery. These are

- four spacecraft on a 16 hour, 10000 x 41740 km, 90 degree inclination orbit – single image coverage; and
- two spacecraft on a 16 hour, 10000 x 41740 km, 90 degree inclination orbit – composite coverage.

Thruster Lifetime Assessment

Analysis is conducted to ensure that the required operating time of the thrusters on-board the spacecraft, in the concepts considered, is within the design lifetime. For the T5 thruster the design limit is around 15000 hours [30]. Three thrusters are required on the Taranis spacecraft, one on each tangential face and one on the radial face. The radial thruster is therefore operational for the total thrust arc duration, while each tangential thruster is only operational for half of this time. Throughout the mission, the length of the apogee coast arc varies; the size of the coast arc for each mission considered and the required operating time for a 10 year mission are given in Table 8.

Determining the thruster operating time, for each launch option, using the thrust levels previously stated in Table 4 and Table 6 results in operating times significantly higher than the 15000 hours maximum of the T5 thruster. Consequently consideration is given to increasing the thrust level to 20 mN, the maximum value of the T5 thruster. This will reduce the size of the thrust arc and therefore the operating time of the EP system. The resulting values of coast arc width and the corresponding required thruster operating times are also given in Table 8.

Increasing the maximum thrust for the single spacecraft Soyuz launch reduces the thruster operating time significantly below the design limit. However, the operating time for the radial thruster for the multiple spacecraft Ariane 5 launch case remains above the operating limit by around 5000 hours for a 10 year mission. It is therefore likely that in this case an additional radial thruster would be required to ensure operation throughout the entire mission.

A small increase in payload mass can also be gained by increasing the thrust level, due to the reduction in required propellant, as shown in Figure 16 and Figure 17. For example, for an 8.5 year mission with single spacecraft launch from the Soyuz the payload mass range is 151 – 379 kg (20 – 50% of mass after deductions for fuel, tanks and thrusters), this is compared with 142 – 356 kg for the previous smaller maximum thrust value. Similarly, for launch of multiple spacecraft on the Ariane 5 the payload mass range is 193 – 482 kg compared to 171 – 427 kg previously.

Table 8 Thruster operating times for 16 hour orbit and various launch options

Launch	Thrust level per thruster (mN)	Initial Coast arc width (deg)	Coast arc width 10 years (deg)	R time (hrs)	T time (hrs)
Single s/c Soyuz	10.3	±18.49	±23.16	35100	17500
Single s/c Soyuz	20	±25.76	±31.88	4200	2100
Four s/c Ariane 5	12.18	±18.14	±22.87	35400	17700
Four s/c Ariane 5	20	±33.94	±36.70	20550	10280

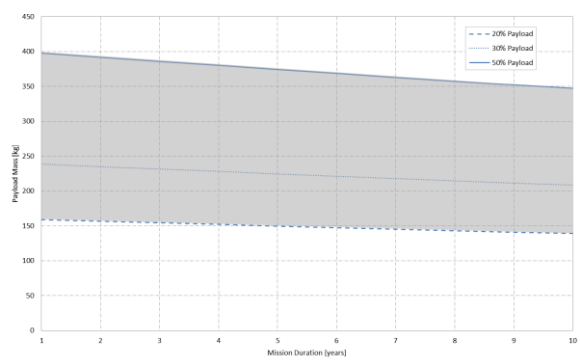

Figure 16 Payload mass allocation over a range of mission durations for the 16 hour, 90 degree inclination, orbit for single spacecraft launch on the Soyuz launcher – maximum thrust of 20 mN

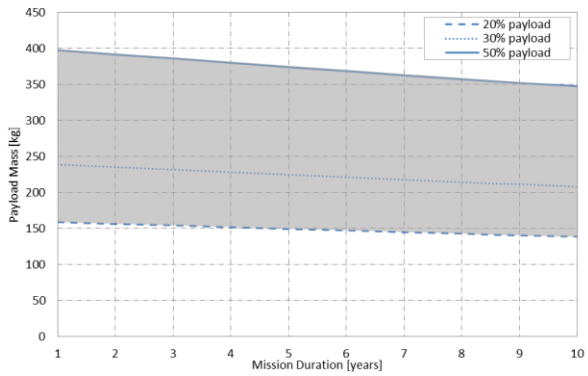

Figure 17 Payload mass allocation over a range of mission durations for the 16 hour, 90 degree inclination, orbit for multiple spacecraft launch on the Ariane 5 launcher – maximum thrust of 20 mN

Top-Level System Analysis

The subsystem mass breakdown for a Taranis spacecraft, based on the down-selected concepts, is given in Table 9. It should be noted that at this stage this is a high-level analysis designed to give a mass estimate for each sub-system.

As stated previously, three T5 thrusters are required to enable the Taranis orbit; one on the radial face and one on each of the tangential faces. In order to ensure redundancy in each direction a thruster is added in each direction, giving a total of five thrusters. From [36], the mass of the propulsion system is assumed to equal 8% of the spacecraft dry mass.

The propellant mass required for an 8.5 year mission including EoL de-orbit is found from the previous analysis, and the mass of the propellant tanks is assumed to equal 10% of the propellant mass.

The payload mass is based on the 250 kg required for the FCI with a reduction in the spectral channels of the instrument and lightened structure.

From [36], the mass of the thermal control; telemetry, tracking and command; on-board processing; and attitude determination and control subsystems are assumed to equal 4, 5, 4 and 7% of the spacecraft dry mass. From [36], the structural mass accounts for a moderately large proportion of the total spacecraft mass at 24% of the dry mass. It should be noted that these estimates are an average of the range suggested in [36].

Electrical power for the Taranis spacecraft will be generated by solar photovoltaic arrays sized according to the assumed power requirements of the other subsystems. This is based on the maximum power the spacecraft will require at a single point. This was found to be approximately 3.3 kW, with the main demands from the propulsion system (maximum of two T5 thrusters firing) of 1.7 kW, communications system assumed to be of the order of 1.2 kW, and the FCI of 320 W. This results in a solar array of 12 m^2 and mass of 132 kg.

This results in a total wet mass of 869 kg, 10 kg higher than the 859 kg capacity of the Soyuz vehicle to the 16 hour Taranis orbit. The design of a Taranis platform launched using the Soyuz vehicle would therefore be challenging and would require a reduction in mass to become feasible.

This top-level system analysis is also conducted for a platform launched using the Ariane 5 vehicle (four spacecraft launch) and the results given in Table 9. In this instance, the total mass is found to be 979 kg, i.e. 45 kg less than the mass which can be launched to the 16 hour orbit using the Ariane 5. Multiple spacecraft launch using the Ariane 5 is therefore less challenging than the single spacecraft Soyuz launch case. This being said, the cost of launching the Ariane 5 is greater than that of the Soyuz. Further details on launch cost are therefore required before a decision could be made on the most beneficial system.

Table 9 Subsystem mass breakdown for a Taranis spacecraft on a 16 hour orbit

Component	Single Soyuz launch Mass (kg)	Multiple Ariane 5 launch Mass (kg)
Propulsion	63	73
Propellant	70	112
Propellant tanks	7	11
Payload	250	250
Thermal control	32	36
Telemetry, tracking & command	39	46
On-board processing	32	36
Attitude determination & control	55	64
Structure & mechanisms	189	219
Power	132	132
TOTAL	**869**	**979**

SUMMARY

This paper has outlined a set of requirements for selection of a polar observation system and measured proposed systems against these criteria. This process revealed the inability of Non-Keplerian and traditional HEO systems to fully overcome the high-latitude data-deficit. Subsequent analysis of the Taranis HEO

concept was then presented. Further analysis was conducted of three systems:

- four spacecraft on a 16 hour, 10000 x 41740 km, low-radiation orbit;
- three spacecraft on a 12 hour, 300 x 40170 km, high-radiation orbit; and
- three spacecraft on a 10 hour, 2000 x 32400 km orbit

Each of these polar observation systems is capable of observing continuously to 55 degrees latitude, with maximum VZAs equal to those produced from GEO.

Down-selection of the Taranis mission concepts was conducted and two concepts were chosen for further investigation; one using single imagery and one using composite imagery. These are

- four spacecraft on a 16 hour, 10000 x 41740 km, 84 degree inclination orbit (single imagery); and
- two spacecraft on a 16 hour, 10000 x 41740 km, 90 degree inclination orbit (composite imagery).

The required thruster operating times and top-level system breakdown were investigated giving consideration to two launch options; single spacecraft launch to enable the two spacecraft composite coverage system, and four spacecraft launch to complete the single imagery system. It was found that by increasing the maximum thrust to 20 mN the thrust arc is adequately reduced to ensure the thruster operating time, for the single spacecraft Soyuz launch, is below the designed limit of the T5 thruster. However for multiple spacecraft launch using the Ariane 5, the radial thruster operating time remains greater than the T5 limit. It is expected that an additional radial thruster would therefore have to be added to ensure operation throughout the mission.

A general overview of the spacecraft systems revealed that it would be challenging to launch a Taranis spacecraft using the Soyuz vehicle. This is due to the restricted mass which can be delivered to the 16 hour orbit. This would therefore require an aggressive platform design in order to become feasible. On the other hand, the design of a platform launched, along with three other platforms, using the Ariane 5 would be more flexible and would allow for significantly greater margins.

CONCLUSION

Analysis of Taranis platforms for polar observation has been conducted. Considering multiple spacecraft per launch (three spacecraft for 10 and 12 hour orbits and four spacecraft on a 16 hour orbit), reasonable payload allocations are possible. For 8.5 year missions with launch on the Ariane 5 launcher payload masses of 81 – 203 kg for the 10 hour orbit, 32 – 81 kg for the 12 hour orbit, and 170 – 426 kg for the 16 hour orbit are possible. Similar payload mass values are achievable considering single spacecraft launch using the Soyuz vehicle for the 10 and 16 hour orbits of 58 – 145 kg and 142 – 356 kg respectively. However for the 12 hour orbit, the capacity for payload is significantly reduced.

The process of down-selecting the Taranis concepts revealed two potential mission options, both on the 16 hour Taranis orbit with an inclination of 90 degrees. These are two spacecraft providing composite coverage with single spacecraft launch using the Soyuz and four spacecraft providing single image coverage with all spacecraft launched on an Ariane 5 vehicle. Performing mass budget analysis for each with maximum thrust values of 20 mN resulted in payload mass ranges of 151 – 379 kg and 193 – 482 kg for platforms launched from the Soyuz and Ariane 5 vehicles respectively for an 8.5 year mission.

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