DARIS, A FLEET OF PASSIVE FORMATION FLYING SMALL SATELLITES FOR LOW FREQUENCY RADIO ASTRONOMY

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ABSTRACT

DARIS (Distributed Aperture Array for Radio Astronomy In Space) is a mission to conduct radio astronomy in the low frequency region from 1-10MHz. This region has not yet been explored, as the Earth's ionosphere is opaque to those frequencies, and so a space based observatory is the only solution. DARIS will undertake an extragalactic survey of the low frequency sky, and can also detect some transient radio events such as solar or planetary bursts.

To achieve these scientific objectives, DARIS comprises a space-based array, forming a very large effective aperture, as required for such a long wavelength survey. Each station in the array (each required to be a small satellite to ensure several nodes can be flown) carries three orthogonal dipole antennas, each 5m in length. The more station nodes in the array, the more sensitive the antenna. The entire fleet remains within a 100km diameter cloud.

A very large data volume is generated by each node, as the antennas have to capture all radio signals, after which the data can be correlated to find the astronomical signal in the noise. As the astronomical signals also have a noise-like nature, no compression is possible on the data captured by the nodes. The data volume is too high to transfer directly to Earth, and will need to be correlated in space. Distributed correlation between the nodes is technically challenging, and therefore a mothership acts as the central correlator and then downlinks the correlated data (lower volume) to Earth.

Another consideration when taking low frequency radio astronomy measurements is the level of interference (RFI) coming from the Earth. The array needs to be located either in the Shielded Zone of the Moon (SZM) or far enough from Earth that the RFI is can be neglected. Landing on the Moon in the SZM is a possibility, providing stable baselines, however this would be a very complicated mission. Other possibilities are either orbiting the moon and taking measurements while flying in the SZM, or in a solar orbit more than 5 million km from Earth.

A Soyuz launcher will allow 6 small satellites to be carried, with the mothership also carrying a set of the astronomical antennae and so forming a seventh node.

The mothership will also act as the carrier craft to transport all the spacecraft to their final orbit, providing the significant Delta-V required, and has a wet mass of 900kg, including 320kg of propellant.

The other 6 nodes are small satellites (less than 100kg), and cannot carry much propellant. However, the nodes must remain in formation, requiring formation flying (defined as maintaining

position to within a particular fraction of the wavelength being observed, which in this case is long). Therefore, significant effort was invested to find orbits where the 7 satellites remain formation flying with little or no orbital maintenance. The position must not change by more than 3m within an integration period, which determines the maximum relative range rates.

1. THE DARIS MISSION

The Science Case

The frequency band below 30 MHz is one of the last unexplored bands in radio astronomy. Due to a combination of ionospheric scintillation below ~30MHz, its opaqueness below ~10MHz, and man-made radio frequency interference (RFI), earth-bound radio astronomy observations are either severely limited in sensitivity and spatial resolution or entirely impossible. There are however, a number of interesting physical processes that naturally take place at these low frequencies [2,5,6], but for which the detailed study has been hampered by the limitations mentioned above. A space-based low-frequency radio array [1,3,4] would suffer significantly less from these limitations and hence would open up the last, virtually unexplored frequency domain in the electromagnetic spectrum.

The low-frequency band is well suited for studying the early cosmos at high hydrogen redshifts, the so-called dark ages, extragalactic surveys, (extra) solar planetary bursts, and high energy particle physics. In addition, space research such as space weather tomography, are also areas of scientific interest.

The extension of science into new regimes in the electromagnetic spectrum has in the past led to new insights and unexpected discoveries; different frequencies correspond to different energies and hence to different physical processes. In that respect the extension into the low-frequency regime below ~30 MHz will undoubtedly lead to new scientific discoveries in many area's of astronomy.

The DARIS space-based radio telescope concept is based on a three-dimensional satellite constellation operating as a coherent large aperture synthesis array. Initially, this distributed aperture array will consist of eight individual antenna nodes (satellites) in passive formation flight, and will be completely scalable.

With such an array, sensitive and high resolution (~1') extra-galactic surveys can be conducted. With eight antennas observing at 10 MHz for one year, DARIS will reach a sensitivity level of 65 mJy and a resolution of ~1', which gives it the potential to detect half a million sources. A second interesting science case is detection of radio transient events, Jupiter like flares, and Crab like pulses. The proposed antenna constellation will be sensitive enough to detect the Jupiter-like flares and the Crab-like pulses without being confusion limited. Radio imaging of the Sun is a third science case. Tracking solar explosions such as Coronal Mass Ejections (CMEs) as they propagate in the Solar wind, improve space weather forecasting and provide early warning of magnetic storms. As the DARIS cluster proposes three dipoles per satellite node, full Stokes polarimetry will be possible. For sources beyond our solar system we expect linearly and circularly polarised signals. Scaling up the array, in free space or on the far-side of the Moon, would open up an even larger scientific domain, including cosmology studies into the dark ages, and studies on weak radio transients and extra-solar planets searches.

Science Requirements

As mentioned above, the imaging principle of the DARIS constellation is aperture synthesis. The received signals of each of the satellites are transported to a central satellite where they are correlated and integrated. These integrated correlation matrices are sent to Earth for further processing. Sky images can then be created by applying beam-forming techniques or Fourier transforming techniques to the observed correlation data or by related techniques. Main system requirements related to the imaging process are discussed next.

A maximum baseline (pair-wise satellite distances) of 100 km is needed to obtain sufficient spatial resolution (~1' at 10 MHz) for the science cases under consideration, especially for the extragalactic surveys. More than 100 km is not meaningful as the scattering of the interstellar medium will limit the spatial resolution. Also, the number of independent baselines, or spatial sample points, must be larger than the number of astronomical sources which ultimately can be detected. This is achieved by combining multiple observations and by combining sky maps of different frequencies (bandwidth synthesis).
It should be emphasized that a low-frequency array in space does not need to fix the relative satellite distances to within a fraction of a wavelength. On the contrary, it is advantageous that the satellites have a relative drift, provided that they remain inside the cluster diameter of 100 km. There is however a requirement that the satellites do not move more that 1/10 of a wavelength in one interferometer-correlation integration time period of one second.
A full 10 or 30 MHz instantaneous bandwidth would be nice to have, but even with a roughly 10% frequency duty cycle, or 1 MHz bandwidth, there is exciting new science to be done, as discussed above. A channel frequency resolution of 1 kHz was chosen so that the signal processing (correlation) inside the cluster can be considered narrow-band. This means that time delays can be represented by phase rotations. This allows the creation of an all-sky image from one correlation matrix or by combining a set of correlation matrices consisting of multiple temporal snapshots. Using broad-band processing techniques, such as applying time-delays, would limit the observable filed of view. Narrow-band processing in space is favourable as it shifts signal processing load from space to Earth.
The integration time will be one second. At the deployment locations under consideration, the relative drift of the satellites will remain within a fraction of a wavelength within this time period. If the stability of the baselines allow, for example at Sun-Earth leading/trailing orbits, the integration time can increase up to 1000 seconds. One of the advantages of increasing the integration time is that the down-link data transport rate to Earth is reduced.

2. DISTRIBUTED ANTENNA PRINCIPLE

Radio astronomers calculate the fourier transform of the measured coherence function to make maps of the sky. The coherence function (ζ_{ij}) is the cross correlation product between two antenna signals x_i (t) and $x_j(t)$ and is given as $E(x_i(t)\ x^*(t-\tau_{ij}))$, where E(.) is the expectation operator and τ_{ij} is the light travel time between observation of the same plane wave at the two spatial positions labelled i and j. The superscript (*) indicates conjugation. The number of cross-correlation products increase as $O(N^2)$ for N antennas and the expectation operator is applied over a period of integration time τ_{int}. There are two ways to implement such a system. One is using the traditional correlator model XF i.e. cross correlation first and fourier transform later and the more recent FX correlator [7] which measures the cross-power spectrum between two antenna signals. While XF architecture is beneficial because bandwidth can be traded for spectral resolution, FX architecture reduces processing requirements [8] and offers scalability when the number of antennas is large. We choose to use the FX correlator architecture for DARIS system.
DARIS cluster will comprise of 8 satellite nodes along with a mother ship, totalling to 9 observational satellite nodes. A centralized-correlator-centralized-downlink is envisaged for the DARIS mission. All the 8 satellite nodes will observe the sky in frequency range of 1-10MHz and will communicate the chosen instantaneous bandwidth of 1 MHz to the mother ship. The mother ship will host a centralized correlator to make cross correlation products and will downlink the result to earth based ground stations.

DARIS Nodes

Each of the observational satellite node will comprise of three dipoles, offering three separate data paths to the node level signal processing unit. Since the total observational bandwidth is too low, Direct Digital Conversion (DDC) is employed. Thus entire observational bandwidth of 10MHz input signal from each dipole is conditioned and directly sampled using a 16bit analog to digital converter at the nyquist rate of 20 MHz. A Poly phase Filter Bank (PFB) [9] is used to selective choose the desired 1 MHz bandwidth [10]. Furthermore RFI mitigation techniques can be employed to eliminate interference and reducing the number of bits to 1. Thus the output from each signal processing data path will be (Nyquist rate)×(# of bits)×(instantaneous bandwidth), i.e. 2×1×1 MHz=2 Mbps. For three data paths, the total output from each satellite is 6 Mbps, is sent to the intra satellite communication layer for transport to the centralized mother-ship.

DARIS Mother-ship

The DARIS mother-ship will receive 3×8=24 signal paths from the entire DARIS cluster of 8 satellites. In addition, since the mothership will also be an observation station the total number of data paths for processing will be 3×8+3=27. Each of the 27 signal paths will have a data rate of 2 Mbps as mentioned in the previous section. The 1 MHz bandwidth is then divided into narrow bands of 1KHz spectral resolution using a Poly-phase Filter Bank (PFB), to meet the narrow band criterion. The total number of bins at the output of the PFB is 1MHz/1Khz = 1000. All the 1000 bins from each of the 27 data paths are fed into a single correlator unit. The correlator makes cross correlation products between each pair of the 27 data paths and integrates and averages over the integration time of 1 sec. The output from the correlator is given by (nyquist rate)×(# of signals)×(# of signals)×(# of bins)×(# of bits)/(integration time), i.e. 27×27×1000×1/1 = 1.458 Mbps. The total data of 1.458 Mbps is then down-linked to earth via the downlink communication layer.

3. BOUNDARY CONDITIONS

In order to conduct the science described in the science case above there are of course several solutions that can be realised. However in order to both impose a limit on the design and also ensure the implementability of the DARIS mission, several boundary conditions are implemented.
The first boundary condition is programatics. DARIS is a science mission, and therefore ESA's Cosmic Vision was taken as the reference for the programatic framework for implementing science missions in Europe. Presently the Cosmic Vision programme contains medium and large missions (so called M and L class missions, respectively). L class missions are for very complex, usually collaborative missions with one or more other space agencies (currently studied L class missions are: the Laser Interferometer Space Antenna, LISA, a collaboration with NASA attempting to directly detect gravitational waves for the first time; the International X-Ray Observatory, IXO, a 3.3m diameter X-Ray telescope with a 20m focal length; and the NASA/ESA Laplace mission to the Jupiter system, of which Europe's contribution, is the Jupiter Ganymede Orbiter, JGO). M class missions are for missions with a higher Technology Readiness Level (TRL) and lower risk, requiring also less technology development. DARIS requires not significant technology development to be realised, and so should be considered as an M class mission. M class missions have a maximum cost to ESA of €475 million in 2009 economic conditions. There is also at present a so called mission of opportunity, SPICA, which has a cost much lower than the M class cost cap, and will be realised if an opportunity in the Cosmic Vision Programme allows. DARIS may even fit within such a classification.
Having agreed that DARIS should be considered as an M class mission, this drives the technology that can be considered in the design of the system; of course, more advanced technologies, and/or extensive technology development plans could realise a more extensive mission, but at a higher cost and risk and in a longer time frame. Only high TRL technologies and components are considered, relying on trusted design solutions to produce a concept that can deliver on the science case.
It was also decided at an early stage to focus on European launchers only, as DARIS is intended to be an ESA mission. The Ariane 5 is the largest European launcher, however at approximately €150 million also the most expensive. In order to propose a more affordable mission, the Soyuz was chosen as the launch vehicle for DARIS. (There are other European launch vehicles, however they are too small for DARIS.) Europe will soon operate the Soyuz from its Kourou spaceport, however

the launch performance data for an Earth Escape trajectory is not yet known, and so the user manual has the same performance as a Soyuz launching from Baikonour. However, due to Kourou being closer to the equator, and less difficult drop zone requirements, the performance will be higher. An Arianespace representative stated that the performance should be 1800kg for a C3 of 0.5m2/s2 but it won't be confirmed until an update of the User Manual is issued, and he couldn't say when that would be. For this feasibility study 1800kg will be used, unless or until further information is provided.

4. INTERSATELLITE COMMUNICATIONS

The inter-satellite communication link (ISL) establishes the data link between the satellites and the mothership. Most implemented ISLs take place over long distances using optical communication [13,15]. Optical communication has many advantages when compared to radio communication. These advantages include reduced mass, power, and volume of equipment, higher data rates, no regulatory restrictions as experienced for radio frequency (RF) bands. However, to communicate effectively over the DARIS range of 100 kilometres using optical communication, this would require (a) accurate alignment (b) more power than is probably available (c) no glare from the moon due to solar radiation (d) no solar radiation either, (e) no glare from the Earth, and (f) a lack of starlight. The orientation of the small satellites would not permit anything like the alignment accuracy required to open up the possibility of an optical link going up to 100 kilometres at very high data rates. Consequently, an RF solution is the only practical approach for the ISL.

The ISL is a two way communication link, existing of a downlink of the (astronomical) data towards the central correlator in the mothership and an uplink for housekeeping, control, timing and synchronization. The data rate of the uplink is rather small. For the time being we assume a data rate of 100 kbps for the uplink. As discussed above, the ISL downlink rate is 6 Mbps, as we assume an RFI free environment, such as the shielded zone of the Moon or a long distance from the Earth.

The ISL channel is in principle just free space loss. Multipaths or even Doppler effects can be neglected. Interference from other satellites in the configuration and also other spectrum users in space can be of impact of the channel. However, for now, we assume just the free space loss of the channel. If we assume an ISL average frequency of 2.4 GHz, the free space path loss with 50 kilometre node-to-mothership distance is 134 dB.

For the calculations of the link budget several approaches can be made. One can define a required Eb/N0 and calculate the necessary transmit power to achieve this. For the downlink transmitter on the satellites, a coaxial switch is used in the final stage to send the signals to the antenna which faces the mothership. Each face of the node has a patch antenna. Since it is required to know the exact orientation of the node with respect to the mothership, this information can be used to choose the correct antenna. On the receive side at the mothership, each patch antenna has a separate LNA. After the amplification of the signals, the signals are first detected and the on-board computer will combine the signals of each of the six signal paths. With a 5 W transmit power, 6 Mbps transfer rate, the link can be established.

Each node in the DARIS configuration will send its data to the mothership. Several techniques can be used to assure that the data can be selected by the individual receivers. The possible dimensions for multiple access are time, frequency, code and space. One of most straightforward implementations is frequency division multiplexing (FDM). Each node will transmit its data using PSK-modulation in a narrow band-width channel. The channels are separated by large guard bands to prevent interference between the channels. For the central uplink a separate band in the spectrum is allocated. If each channel is transmitted simultaneously, the overall data rate will be the sum of all the channels.

If the available bandwidth is limited, for example by extending the number of nodes or in case higher data transmissions are needed, a more (bandwidth) efficient modulation is required. One of the promising techniques to overcome this is OFDM (Orthogonal Frequency Division Multiplexing) [12,14,16]. OFDM has been adopted as standard for DVB, DAB and WLAN. The main advantages

of OFDM are it is very well suited to frequency selective channels and it potentially offers a good spectral efficiency. With OFDM, the separation between each channel is equal to the bandwidth of each channel, which is the minimum distance by which the channels can be separated. The individual channels (in this case the signal of a single node) will be modulated using a form of Phased Shift Keying-PSK, Amplitude Shift Keying-ASK, or a combination QAM (Quadrature Amplitude Modulation).
Although OFDM is a more bandwidth efficient technique, its implementation is more complex (eg. the demands on timing synchronization is much higher), Which modulation technique to use depends on various aspects. In Figure 1 the required bandwidth for FDM and OFDM is plotted as function of the number of nodes in the array. The bandwidth of a single satellite is determined by the required data rate, coding rate, windowing and modulation technique. In this example the required bandwidth is taken 6 MHz for the 6 Mbps data stream. For FDM a guard band of 2 MHz is used to prevent interference between the bands .

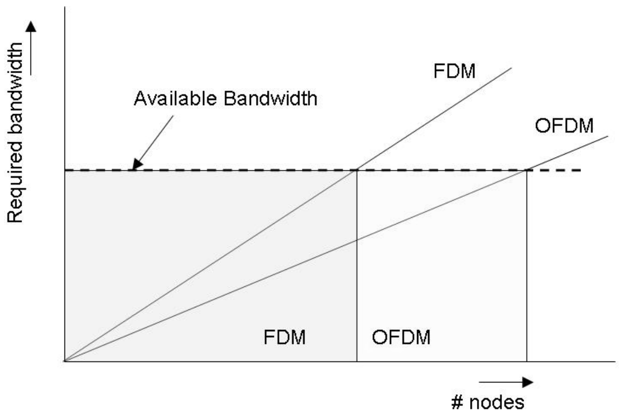

Fig. 1. Example of the required bandwidth for an ISL implementation of FDM and OFDM.

If the available bandwidth for the system is known, as well as the number of nodes, the choice between FDM and OFDM can be made. In the current DARIS design we aim for eight nodes, but with flexibility and upgradeability in mind we are working with OFDM.
The system design of the ISL is the next step. In Fig. 2 a schematic of the proposed uplink ISL is presented.
The data signal consisting of the housekeeping data, etc. is transferred into an OFDM signal. This baseband signal is upconverted to RF. Since the signal must be send to all the nodes and the mothership is in the center of the configuration, all six faces of the mothership should transmit the data. This will be done using a power splitter. A patch antenna transmits the signal towards the receivers.
At the receiver six patch antennas are present at each face of the node. Since the exact orientation of the node with respect to the mothership is known, one of the antennas will receive the signal optimally. Using a coaxial switch (controlled by the CPU) this signal is selected, amplified and finally detected in the node.
In Fig. 3 a schematic of the proposed downlink ISL is presented.

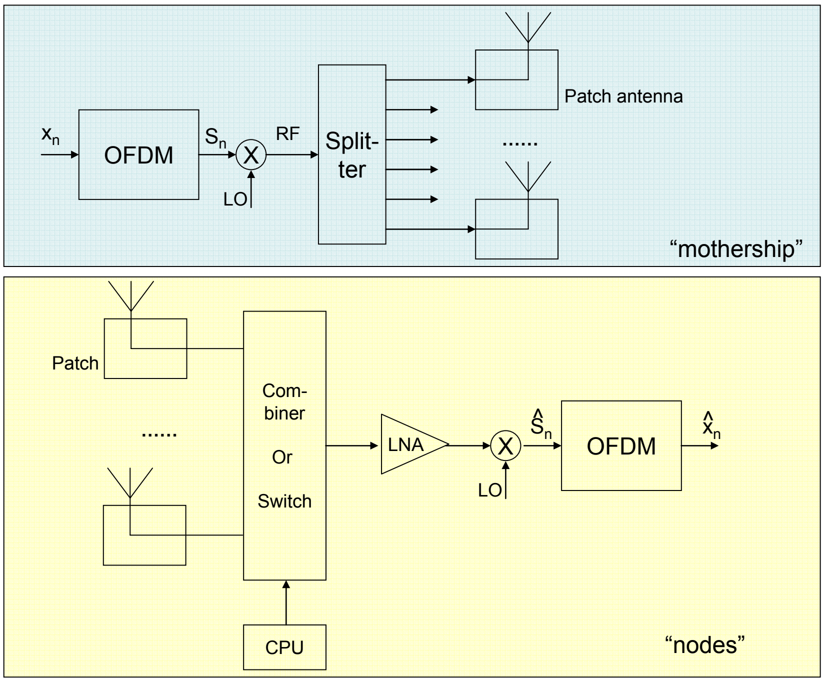

Fig. 2. Schematic diagram of the uplink ISL

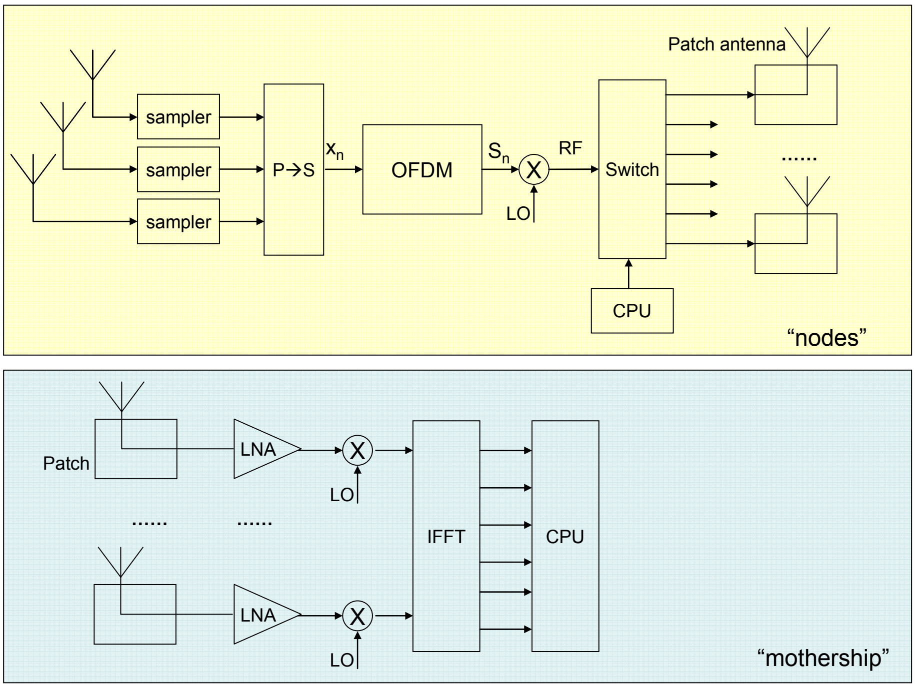

Fig. 3. Schematic diagram of the downlink ISL

The astronomical data from the three antennas is sampled (of course the sampler also consists of a LNA). The digital data is send to an OFDM block to generate an OFDM spectrum. After upconversion to RF, a coaxial switch selects the antenna which faces the mothership optimally. Since this information is known in the on-board computer this is possible. The signal is transmitted to the proper patch antenna.

At the mothership all the six patch antennas can receive signals from nodes. Therefore, a LNA is connected to each patch antenna to reduce noise as much as possible. The downconverter signals of the six signal paths is detected in a OFDM block and send to the CPU for further processing.

5. MISSION ANALYSIS

Potential Deployment Locations

The complexity of keeping nine spacecraft (nodes plus mothership) within a 100 km range with active controls is extremely high. To keep the mass low and reduce the complexity, the option of passive formation flying is chosen without the use of engines or heavy orbital maintenance equipment. In this part, the reference orbit is the orbit of the mothership around the central body and is considered to be the centre of the formation. The node spacecraft follow relative orbits around the mothership.
As discussed above, the RFI from the Earth must be avoided. Two ways of reducing or eliminating this noise is by increasing the distance to the Earth or by shielding from the Earth. The closest possibility for shielding is using the Moon as protection against the noise. Two types of reference orbits are possible to obtain an eclipse behind the Moon with respect to the Earth: an orbit in the Earth-Moon L2-point and an equatorial orbit around the Moon. The orbit around the L2-point requires active formation flying. The equatorial orbit around the Moon gives eclipse times behind the Moon with respect to the Earth to perform the measurements.
By increasing the distance to the Earth, the communication with the Earth becomes more difficult. To perform measurements with sufficient quality a distance of at least 1 million km from the Earth is required. A first possibility is an orbit around the Sun-Earth L2-point, but as for the orbit around the Earth-Moon L2-point, this orbit requires active orbit control for all separate spacecraft. The second possibility is an orbit following the same path as the Earth, but with a change in true anomaly. This results in a leading or trailing orbit. The problem is that the Earth has a very large influence on this type of orbit. The location of the Earth is always the same with respect to the flight direction of the formation. This means that the formation will start drifting away from its location. This can also be explained by the restricted three-body problem, where the formation will continuously orbit on a location with a non-stable gravitational potential, which can be solved by placing the formation in the L4 or L5-point. At these points the formation would remain on the same location, but the distance with the Earth would be around 150 million km. This is too far for the large amount of data to be transferred to the Earth.
The third possibility is an orbit with a slightly different eccentricity than the Earth orbit. Without the presence of the Earth's gravity field, this formation describes an elliptical orbit around the Earth. The motion around the Sun can be seen in Fig. 4. When including the gravity of the Earth, the perfect elliptical shape changes. By keeping the distance sufficiently large, the stability is good enough.

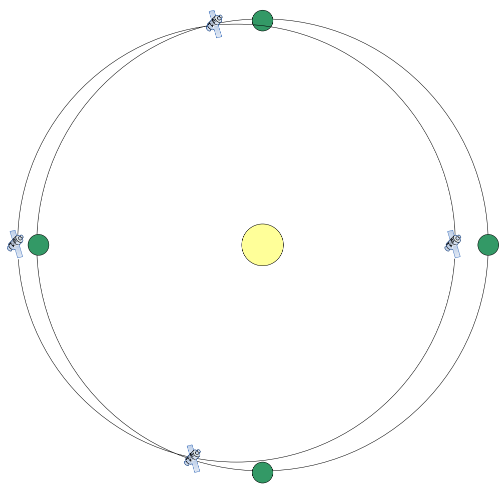

Fig. 4. The reference orbit and the Earth around the Sun

It has been shown that it is possible to have a formation with none or minimal orbital maintenance in such a short range. The trade-off between relative velocity and eclipse time is a disadvantage of the lunar orbit. The solar orbit has the problem of the high sensitivity. The solar orbit is the preferred orbit due to the low relative velocity, but still needs more detailed research. The lunar orbit remains a good alternative.
Details of the two possible orbits, around the Moon and around the Sun, are discussed below.

Formation Flying around the Moon

To increase the predictability of the relative positioning, the reference orbit is chosen to be circular. This decreases the chance of collisions. The formation is build up from different relative orbits with a different relative inclination. In each of these relative orbits several node spacecraft can orbit. They cross the orbital plane of the reference orbit in the flight direction. For more clarification, see Fig. 5. These figures represent the relative orbits with only the Moon as the central body, without any perturbations.

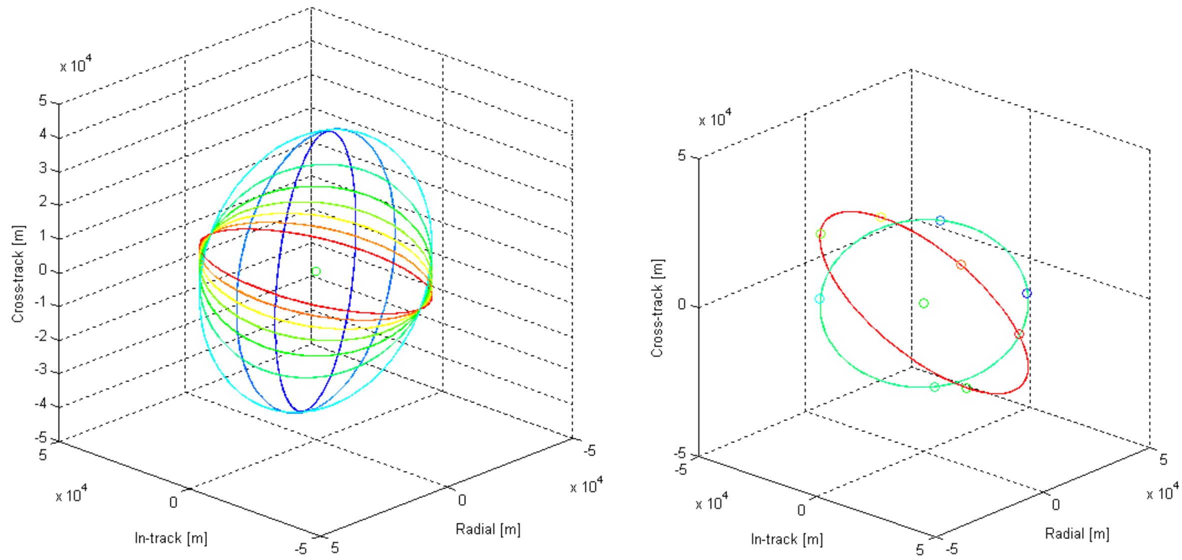

Fig. 5. Left: Relative orbits with different relative inclinations. Right: Two relative orbits with several spacecraft in each orbit

The reference orbit has two major consequences. Firstly, it determines the duration of the eclipse. An orbit in the Earth-Moon plane gives the most eclipse time. A lower altitude also improves the eclipse time. Secondly, the reference orbit determines the relative velocity between the spacecraft. In contrast to the eclipse time, this is improved by increasing the altitude. A balance between the relative velocity and the eclipse time must be found.
When including the perturbations of the Earth's gravity field, the irregularities in the lunar gravity field, the solar gravity field and the solar pressure, a drift of the relative orbits occurs. This drift is mainly in the in-track direction of the reference orbit. This has the advantage that this drift can be compensated by adjusting the semi-major axis. This reduces or increases the revolution time around the Moon and compensates the drift.

Formation Flying around the Sun

The reference orbit is an elliptical orbit around the Earth. The Earth's gravity is a major disturbance on this orbit. The chosen reference orbit remains between 4 and 10 million km from the Earth. As shown in Fig. 6 this orbit is sufficiently stable for at least 10 years. This distance is far enough to reduce the radio noise from the Earth sufficiently.

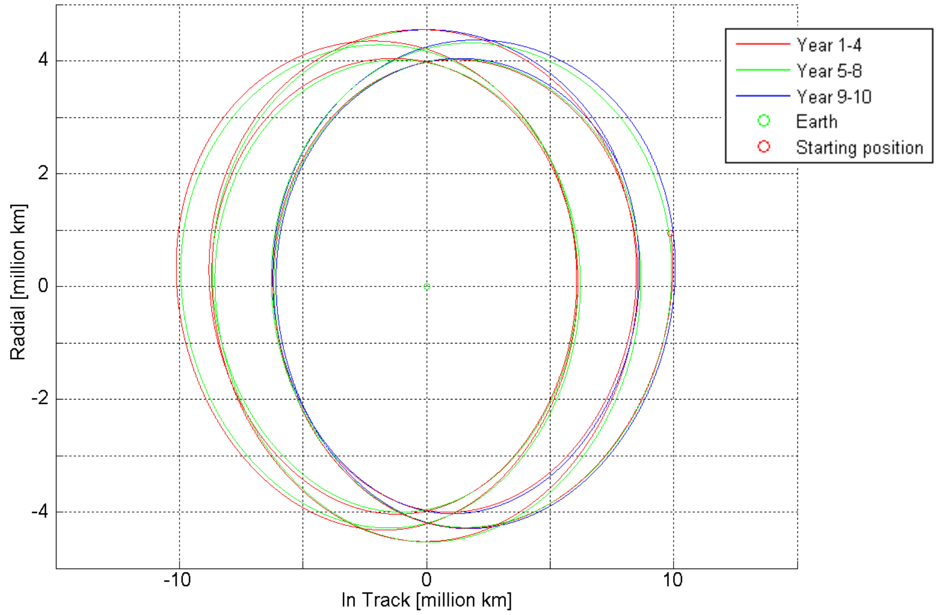

Fig. 6. Reference orbit with respect to the Earth, showing stability over a 10 year period

The relative orbits are designed in a similar way as the orbits around the Moon. The relative orbits have the same period as the reference orbit around the Earth. This causes the formation to drift in all directions. The result can be seen in Fig. 7. When starting close to the reference orbit, the formation will expand. The possibility arises to perform measurements on the different wavelengths. A major advantage of this orbit is the low relative velocity.

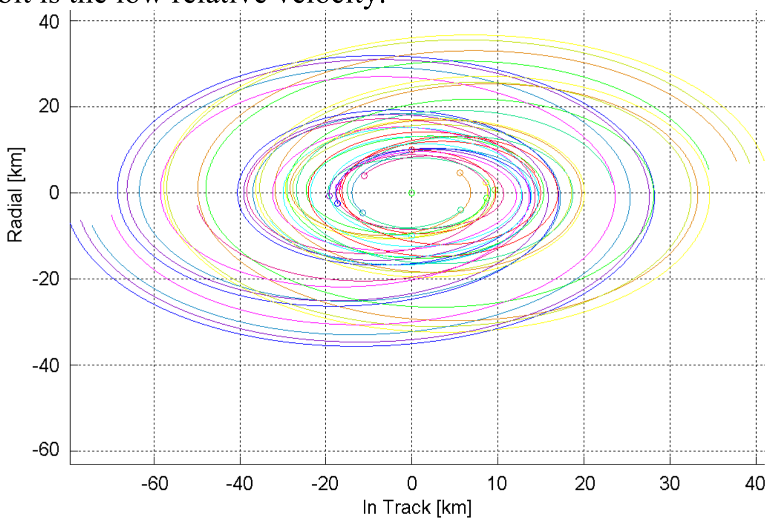

Fig. 7. Relative orbits of the nodes with the mothership in the centre over a three years period

Due to the low relative velocities, these orbits are extremely sensitive to velocity changes. In Fig. 8 a relative orbit is shown with small velocity differences with respect to the ideal orbit. The required accuracy is very high, but can be compensated by the attitude control system.

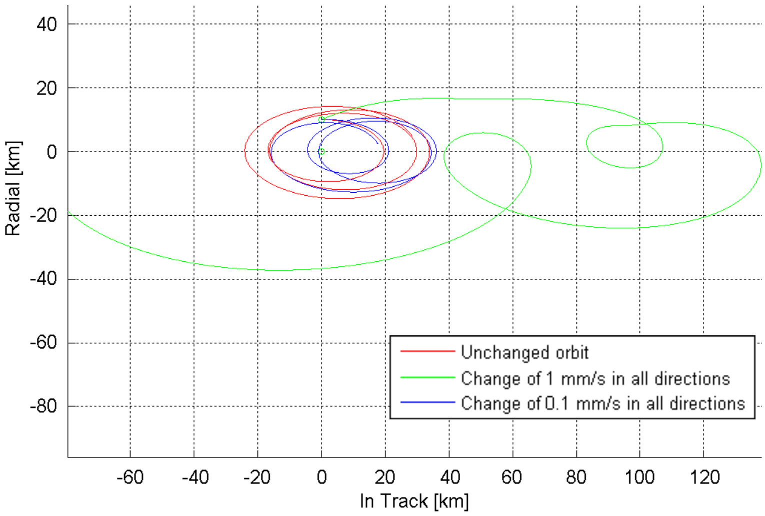

Fig. 8. Relative orbits showing the effect of a minor error, for example at orbit injection

6. SATELLITE DESCRIPTION

In order to realise the array required for the science case, several antennas must be flown in a specific arrangement. These antennas are each carried on a satellite called a node, related to the fact that each is only one node in the array. Each node must therefore only carry the three dipole (actually six monopole) antennas, and a communications system to transfer the signal received from these astronomical antennas to the mothership with its correlator. In this regard they are actually quite simple. The starting point for the design was therefore to realise the requirements in a spacecraft that is as small as possible, while still fitting with the boundary conditions of using tested technology.
As discussed above, however, the data rate that must be transferred to the mothership is 6 Mbps, which is quite high. This can however be achieved with patch antennas with 5 W transmit power, as described above.
There are then effectively two payload for the nodes. One is therefore the equipment for sampling the data, and converting it to an OFDM data stream, then transmitting it to the mothership. The other is the astronomical antennas themselves.
Following an analysis of the antenna properties, it was concluded that for the frequency range from 0.1 to 30 MHz, a dipole of length 5 m (two 2.5 m monopoles) would be sufficient to achieve the characteristics required to sample the signal. Additionally, a cross sectional area of only 1 mm^2 is required, with either steel or CuBe (copper beryllium alloy) as the conducting material.
The required cross sectional area is in fact too small to provide rigidity to the deployed antenna. Tubes of CuBe will provide the required properties for a very low weight. An antenna made of up two 1 m sections and one 0.5 m section, with double C spring hinges (like two tape measures back to back, as shown in Fig. 9) between the sections, allowing them to fold up along each of the six panels of the node. Including hold down and release mechanism to hold the antenna folded and secure during launch, the mass is under 500 g per antenna, including margin.

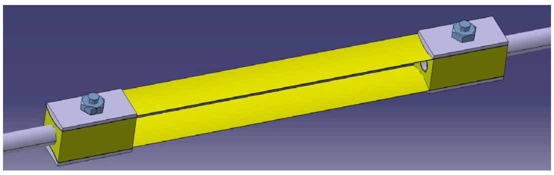

Fig. 9. Double C spring hinge allowing predictable deployment and then rigidity of the astronomical antennas, with a low mass

The pointing requirements of the nodes are not stringent, requiring only a knowledge of the orientation of the antennas, and that the rotation speed is not too fast. If Attitude and Orbit Control System (AOCS) sensors are already provided to provide this orientation, they can also be used to ensure the nodes remain Sun pointed, allowing the solar arrays to be implemented on only one face, saving mass versus a spinning satellite with solar cells on all sides. The nodes therefore have reaction wheels, as well as a sun sensor and star tracker (the need for the star tracker is still TBC). In order to offload momentum from the reaction wheels, a propulsion system is required.
For small satellites a small thrust is required to be able to be generated by the propulsion system as small satellites have a low moment of inertia, and two high thrust will not allow the wheels to be properly offloaded. Therefore a cold gas propulsion system must be implemented, which is not the most efficient solution, and does cause the propulsion system to be a large portion of the node mass. In order to maintain the sun pointing attitude and offload the wheels, approximately 25 m/s of Delta-V is required, which for the 100 kg node required 4 kg of nitrogen. As discussed above, the solar dynamic orbit is very sensitive to orbit injection errors, and so this propulsion system can also be used to correct these errors. The array maintains its required relative positions and velocities with passive formation flying, requiring only some centimetres per second for error corrections.
In small satellites, it is possible to combine the functionality of the On Board Computer (OBC) and its associated mass memory with the Power Conditioning and Distribution Unit (PCDU), to save mass by procuring one unit with some common electronics components. The On Board Data Handling (OBDH) subsystem is however standard off the shelf technology. In fact, the other subsystems are all realised by following the principles of using only existing and tested technology and are off the shelf components.

The wet mass breakdown of a node is given in Fig. 10

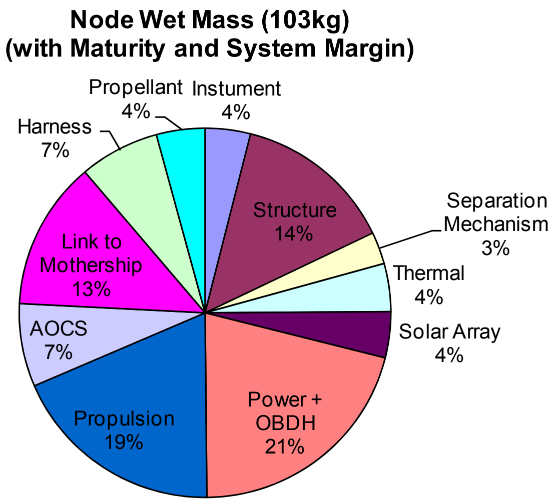

Fig. 10. Node wet mass breakdown by subsystem

7. MOTHERSHIP

In order to complete the transfer to their destination, a change in velocity, Delta-V, must be imparted on the satellites after being released from the launch vehicle to ensure that they arrive at their destination, and are then deployed into the required formation.
The mass of fuel required to complete this transfer is the main limiting factor on the number of nodes that can be carried to fit into the launch vehicle. Taking the Soyuz from Kourou as mentioned, with an Earth escape launch capability of 1800 kg, the mass of fuel required for the approximately 600 m/s Delta-V transfer, utilising a standard chemical propulsion bipropellant system is 340 kg. It is impractical to transfer each node separately to the destination, and so a single transfer stage is required. Also, as the chosen orbit, and in fact the back up too, are destinations in

deep space (outside Low Earth Orbit) the most efficient way to achieve this transfer and deployment is with a transfer stage. The transfer stage has been labelled the mothership.

Additionally, as the signal processing scheme is one with a centralised correlator and centralised transmission of the data to Earth, the mothership can also be used to perform this task.

A system design was therefore also performed for the mothership, to come up with a design that incorporated the required OBC and communications equipment to transfer the data to Earth, as well as a structure sufficient to carry the required propellant and withstand the launch loads, as well as with deployment mechanisms for the nodes.

A standard off the shelf OBDH solution is sufficient for the processing required on the mothership. Standard off the shelf power equipment, including solar arrays for power, as well as AOCS, thermal, and propulsion equipment are also fully sufficient for the mothership, as the requirements on these subsystems are not so demanding. The mothership will remain sun pointed, which is fairly simple in its solar orbit, requiring 3-axis control from the AOCS subsystem. The communication link to the nodes, as described above, will be realised with patch antennas on the six faces of the mothership.

The continuous data rate to be transferred to the Earth, as generated by the astronomical antennas, as well as the housekeeping data from all the nodes and the mothership itself, is actually relatively high. Assuming a 35 m ground station with access for 8 hours per day, a High Gain Antenna (HGA) will be required to achieve the required data rate of approximately 4.5 Mbps for 8 nodes. This requires a 1.5 m diameter parabolic antenna, with a one axis pointing mechanism to ensure the antenna can always point at the Earth as the satellite remains sun pointed (Fig. 6 shows the relative orbit with respect to the Earth, with the Sun always below the figure).

The overall science duty cycle is determined by taking the minimum percentage of time that astronomical data can be gathered and it all both transferred from the nodes to the mothership via the ISL and from the mothership to Earth after correlation. In the current configuration, for 8 nodes, the science duty cycle is just under 100%.

This does however allow for some flexibility in the constellation design. For example, with design or requirement changes, or a capacity launch vehicle, more nodes could be flown, increasing the data rate. As discussed above, the integration time may be able to be longer in the stable solar orbit, which would reduce the data rate, perhaps allowing a Medium Gain Antenna (MGA) to be sufficient, however an HGA has more expandability. Duty cycle can also be traded with the other science inputs, in consultation with the astronomers.

Finally, while the mothership remains in the fleet, another set of astronomical antennas is mounted on the mothership, providing another receiving node in the array.

The mothership dry mass breakdown and the total launched mass breakdown are shown in Fig. 11. A possible configuration of the nodes arranged around the mothership is shown in Fig. 12

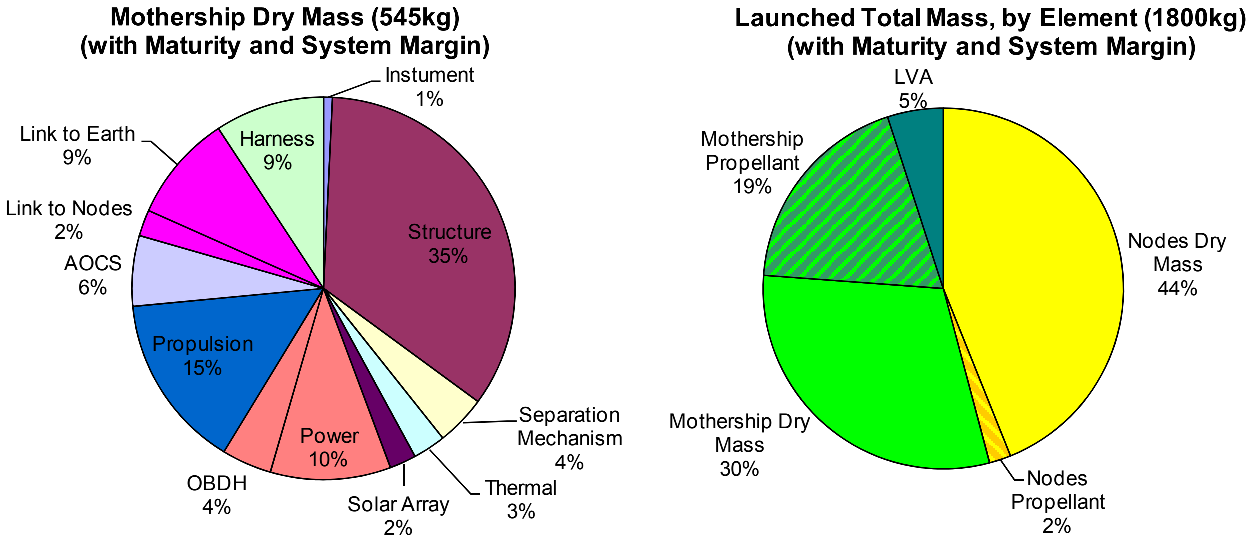

Fig. 11. Mothership dry mass breakdown by subsystem (left), and the total launched mass broken down by element (right)

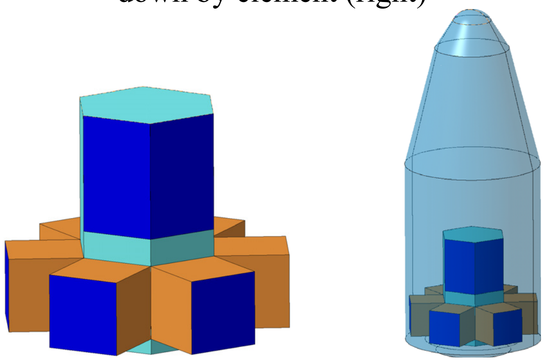

Fig. 12. Six nodes arranged around mothership, and the stack in the Soyuz fairing, showing growth potential

8. FURTHER APPLICATIONS OF A DISTRIBUTED ARRAY

The proposed sensing scenario with eight node spacecraft and a central spacecraft as a star-network with its correlation capacity provides a field-sensing scenario and could be functional as well for other instrumental tasks.
Potential applications might include: seismic sensing networks, once landed on planets or moons; space-weather sensing networks; low-frequency RF-sensing of solar and planetary radio-physical phenomena as a stand-alone or in assistance to sub-surface sounding; or potential interplanetary sensing tasks similar to the Cross-Scale mission plan.
Potential Earth-Observation applications such as passive remote sensing or in a receiving network in a bi-static active RF low-frequency scenario (compare to a Synthetic Aperture Radar scenario with receiving satellites in a cart-wheel scenario) could potentially be considered, though that would need to be further worked out in view of ionospheric cut-off frequency and frequency band to be applied. A cart-wheel sensing in conjunction with UHF radar mission could be envisaged.
With a factor 20 higher frequency, there are obviously impacts concerning the increased accuracy in knowledge of range and range-rate and sizing of the distances allowed.

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10. REFERENCES

[1] Jones et al., 1998, *The Astronomical Low-Frequency Array (ALFA)*, In: Zensus, J. A., Taylor, G.B., Wrobel, J. M. (Eds.), ASP Conf. Ser. 144: IAU Colloq. 164: Radio Emission from Galactic and Extragalactic Compact Sources, p. 393.
[2] Kaiser, M. L., 1990. *Reflections on the Radio Astronomy Explorer program of the 1960s and 70s*, In: Kassim, N. E., Weiler, K. W. (Eds.), LNP Vol. 362: Low Frequency Astrophysics from Space. p. 3-7.
[3] Kassim, N.E., & Weiler, K.W. (Eds.), 1990. *Low frequency astrophysics from space*, Proceedings of an International Workshop, Crystal City, VA, Jan. 8, 9, 1990.
[4] Nieuwenhuizen, M. P., Beenen, R., Simons, W., 1992., *ALFIS: Astronomical Low Frequency Interferometry Satellites. a feasibility study*, TU Delft Faculty of Aerospace Engineering.
[5] Stone, R. G., Weiler, K.W., Goldstein, M. L., Bougeret, J.-L. (Eds.), 2000. *Radio astronomy at long wavelengths.*
[6] Weiler, K.W. (Ed.), 1987, *Radio astronomy from space*, Proceedings of the Workshop, Green Bank, WV, Sept. 30-Oct. 2, 1986.
[7] Chikada, Y et al.: 1987, A 6 × 320-Mhz 1024-Channel FFT Cross-spectrum Analyzer for Radio Astronomy, Proc IEEE 75, 9, pp. 1023–1209, September 1987.
[8] Romney J.D: 1999, Cross Correlators, Synthesis Imaging in Radio Astronomy 2, ASP Conference series, Vol 180, 1999.
[9] Crochiere, R. E. and Rabiner, L. R.: 1983, Multirate Digital Signal Processing, Prentice-Hall, New Jersey.
[10] D'Addario, L. and Timoc, C.: 2002, 'Digital Signal Processing for the SKA', International SKA Conference 2002, Groningen, The Netherlands, 13–15 August 2002. Available at http://www.lofar.org/ska2002/pdfs/D'Addario-correlator.pdf.
[11] Bentum, M.J., Verhoeven, C.J.M., Boonstra, A.J., van der Veen, A.J., and Gill, E.K.A., *A novel astronomical application for formation flying small satellites*, In 60th International Astronautical Congress (IAC), 2009.
[12] Chiu, Y., Markovic, D., Tang, H. and Zhang, N., *OFDM Receiver Design*, 2000.
[13] European Space Agency. *Artemis and Spot-4 chat with laser light*, ESA Bulletin, 108:114, November 2001.
[14] Hu, Y., *An Improved Algorithm for PAPR Reduction in OFDM System*, In Wireless Communications, Networking and Mobile Computing, 2008. WiCOM '08. 4th International Conference on Wireless Communications, pages 1-4, Oct. 2008.
[15] Toyoshima, M., *Trends in satellite communications and the role of optical free-space communications*, Journal of Optical Networking, 4(6):300-311, June 2005.
[16] van Nee, R. and Prasad, R., *OFDM for Wireless Multimedia Communications*, Artech House, 2000.