Final report

Phase I study

EXTREMELY LARGE SWARM ARRAY OF PICOSATS FOR MICROWAVE / RF EARTH SENSING, RADIOMETRY, AND MAPPING

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<table>
<thead>
<tr>
<th>Table of contents</th>
<th>Page number</th>
</tr>
</thead>
<tbody>
<tr>
<td>EXECUTIVE SUMMARY</td>
<td>3</td>
</tr>
<tr>
<td>1. SCIENCE MISSIONS AND POTENTIAL NEEDS</td>
<td>5</td>
</tr>
<tr>
<td>2. CONCEPT AND ARCHITECTURE DESCRIPTION</td>
<td>20</td>
</tr>
<tr>
<td>3. SUBSYSTEMS/ELEMENTS</td>
<td>37</td>
</tr>
<tr>
<td>a. Constellation, orbits and formation flying</td>
<td>38</td>
</tr>
<tr>
<td>b. Space-fed sparse array antenna</td>
<td>40</td>
</tr>
<tr>
<td>c. Phase control commands</td>
<td>44</td>
</tr>
<tr>
<td>d. Doppler nulling</td>
<td>44</td>
</tr>
<tr>
<td>e. Navigation and metrology</td>
<td>45</td>
</tr>
<tr>
<td>f. Communications</td>
<td>47</td>
</tr>
<tr>
<td>g. Central receiver and bus</td>
<td>48</td>
</tr>
<tr>
<td>h. Picosats</td>
<td>50</td>
</tr>
<tr>
<td>1) Picosat propulsion</td>
<td>57</td>
</tr>
<tr>
<td>2) Picosat antennas</td>
<td>58</td>
</tr>
<tr>
<td>3) Picosat functional diagram</td>
<td>60</td>
</tr>
<tr>
<td>4) Picosat propulsion and system weight</td>
<td>61</td>
</tr>
<tr>
<td>5) Picosat production and costs</td>
<td>64</td>
</tr>
<tr>
<td>6) Picosat deployment and retrieval</td>
<td>65</td>
</tr>
<tr>
<td>i. Tether</td>
<td>67</td>
</tr>
<tr>
<td>j. Illuminator</td>
<td>70</td>
</tr>
<tr>
<td>4. ATMOSPHERIC TEMPERATURE PROFILING</td>
<td>71</td>
</tr>
<tr>
<td>5. EXPECTED PERFORMANCE AND UTILITY</td>
<td>73</td>
</tr>
<tr>
<td>6. FEASIBILITY ASSESSMENT and ISSUES REQUIRING RESOLUTION</td>
<td>76</td>
</tr>
<tr>
<td>7. TECHNOLOGIES REQUIRING DEVELOPMENT/DEMONSTRATION</td>
<td>78</td>
</tr>
<tr>
<td>8. CONCLUSIONS/RECOMMENDATIONS</td>
<td>81</td>
</tr>
</tbody>
</table>
EXECUTIVE SUMMARY

The concept proposed is intended to provide a new capability for earth science and applications observations not attainable by space systems current, programmed, or planned by the US or any other nation/consortium. The concept is aimed at the fields of hydrology, and in particular soil moisture for science and trafficability, and coastal and deep ocean salinity observations. The water cycle and hydrology space programs and activities in the US and ESA were reviewed, as well as the findings of Earth Science teams in order to understand the desired performance of future instruments. The requirements derived by the Earth Sciences community call for resolution spot sizes of 100-300 meters and revisit times of hours for such observations at 1.4 GHz. None of even the newest space systems in development and slated for flight in the 2007-2009 period--Hydros and Aquarius by NASA and the SMOS by ESA, which are all single spacecraft in LEO--will provide resolutions better than 35,000 meters or revisit times shorter than several days--far short of the stated performance requirements by the science teams.

The proposed space concept is stationed in GEO and can provide flexible scanning of large or small areas, as well as essentially continuous observations of smaller areas. The stationing in GEO is a key aspect of the concept, but requires a very large antenna both due to the much smaller resolution spot desired than other hydrology space systems and due to providing those finer resolutions from GEO rather than from LEO. This antenna, the heart of the system, is implemented as a highly sparse space-fed phased array some 100 km in diameter, populated with some 1,000-10,000 picosats each acting as a coherent repeater, whose signals all add in phase at a central receiver location above the constellation array. System study has determined that this concept is feasible using an auxiliary CW illuminator also in GEO, and can attain the same sensitivity as Hydros, Aquarius, and SMOS while meeting the resolution, and short revisit times desired.

The analysis has determined that the picosats can be placed in halo type sub-orbits following Hill’s equations, and will describe a slowly rotating plane inclined some 26 degrees to the local horizontal, which is not only quite stable for long times but requires at least an order of magnitude less propulsion for stationkeeping the picosats than if they were in non-Keplerian orbits. The phase control means and accuracies required for coherent signal addition in the central receiver were analyzed, and it was determined that the picosat positions need to be known to an accuracy of 3.3 cm worst case, but their actual positions could be allowed to drift at least 100 meters from those assigned. A technique of coherent distribution of the local oscillator was identified as most promising to ensure that proper phase control of the phase of the picosats is achieved for coherent signal addition. Phase control requirements were developed for antenna focusing in order to attain the 100-300 meter spot size on the ground, as well for beam scanning. The pattern of the sparse array was explored using a computer program and its characteristics were determined to be appropriate to meet the antenna sidelobe levels and beamwidth desired. The determination of picosat position was explored and a concept implementing a local GPS-like (but not GPS) navigation environment identified as appropriate to attain the required accuracy for phase command generation, either by the picosats themselves or by auxiliary units orbiting around the constellation acting as navigation references. A tether through the center of the constellation holds the central receivers without expenditure of propellant, and analyses confirmed that tethers in GEO are both feasible and very lightweight even for lengths of many
hundreds of km. The technologies required to implement the picosats themselves were identified, and activities within The Aerospace Corporation to micromachine all silicon or photostructurable gall satellites noted as being both appropriate and well suited for meeting the desired picosat integration, weight, and dimensions. Some picosat subsystems were examined and found feasible, especially the power subsystem to supply the power needs of the picosat housekeeping, propulsion, and payload needs. Fabrication and mass production of the picosats was identified as a principal technology that needs further definition and development. It was noted that The Aerospace Corporation had fabricated and flown two tethered picosatellites in 2000, though the integration level of subsystems was not yet as high as needed for the concept’s picosats. A concept was defined for adding a depot and a scavenging facility in order to deploy, retrieve, and store picosats; and scavenge dead ones to prevent the formation of orbiting debris.

All the major subsystems and the entire concept were found feasible, with no showstoppers identified. A number of techniques that require detail definition and possibly demonstration were identified, but only two which require technology development and demonstration. These were all described and recommended to be undertaken in Phase II.

The utility of the concept for earth science soil moisture and ocean salinity observations, as well as for road trafficability applications, was demonstrated in graphical form. The system is capable of revolutionizing these fields with a single space system observing almost a hemisphere. No other system in development or even contemplated can come close to the predicted performance and utility, or meeting the requirements as established by the Earth Science community. The concept was verified as representing a truly revolutionary Earth Science observation capability, and thus is recommended that the study proceed into phase II.
1. SCIENCE MISSIONS AND POTENTIAL NEEDS

Earth science has six driving focus areas, which are Atmospheric Composition, Solid Earth, Climate, Carbon Cycle, Weather, and Water and Energy cycle. Data gathering in these areas from space has been a NASA enterprise for decades, and has also been the subject of NOAA and DOD activities particularly with respect to weather and climate. The technologies utilized are as diverse as the phenomenologies that are the targets of the observations, with some shown in Figure 1.

The nature of the advanced concept proposed for feasibility assessment in this NIAC Phase I study, which is fundamentally a technique for implementing a very large yet very lightweight antenna suitable for operation in GEO, makes it applicable to all these science focus areas using RF/microwave techniques, and particularly those which could benefit from being stationed in GEO for more persistent and flexible coverage, and more frequent revisit times, than possible from the predominantly LEO spacecraft which are in use today as well as in development, and which are also the principal focus of future plans.

Large antenna microwave systems in GEO could, in principle, perform similar functions to those of the LEO systems, with the added benefit of more frequent revisits, continuous observations, or more flexible scan and coverage patterns than LEO based systems which are forever doomed to revolve around the earth every two hours or so over constantly changing surface areas. The science focus areas that would benefit from these new abilities most would be weather and the water/energy cycle, as these are the areas in which the phenomena change rapidly enough and vary greatly from place to place in time that the GEO-based system’s characteristics would be most likely to lead to beneficial capabilities. Thus the focus of this Phase I study will be Earth Sciences, particularly emphasizing the Water Cycle and to a lesser degree Weather. There may be other mission areas of applicability but they will have to be addressed later, in Phase II.

a. The water cycle and hydrology
The water cycle, illustrated in Figure 2, has been the focus of many investigations, theoretical and experimental for many years. It is a fundamental aspect of living on our planet, and is the mission that is chosen for the advanced concept study. Water cycle radiometers and scatterometers fly in a number of current spacecraft, and have recently been the subject of mission design of three advanced spacecraft which will, in the next 2-5 years, provide global coverage and greater sensitivity than instruments of present spacecraft, and all will contribute to understanding of hydrology. This chapter will discuss the hydrology science observations desired, summarize the three advanced missions, and contrast their capabilities with the science requirements and desiresments that the science community has identified for the hydrology areas.

A number of roadmapping activities\(^1\) have been and are underway at NASA and elsewhere to set into context the current, programmed, and planned missions and technologies required, and to lay out the shortfalls and technologies required. The roadmaps for the weather and water areas are shown below in Figures 3 and 4.

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\(^1\) NASA 2004 budget request support documents, from NASA web site
Figure 3
The earth Science community and its science teams have identified requirements for observations in order to characterize and understand the water cycle and its underlying hydrology phenomenologies. These are in the areas soil moisture content, freeze-thaw cycles, snow accumulation levels, flooding extent and precise geographical location, emergency management after hurricanes and floods, water content and temperature profile in atmosphere, ocean salinity, coastal salinity and river effluents, and other water-related earth science applications. These are discussed briefly below, with some implications for observing requirements.

The terrestrial hydrologic cycle includes Ground Water, Surface Water, Soil Moisture, and Snow Pack; and glaciers. Its understanding depends on many fundamental questions, including: What are the contributions of each land component to the global fresh water budget? How do natural and anthropogenic processes redistribute water in both space and time? How does the land surface vary with time and how does it influence the dynamic water supply? These are needed to address the overarching question of how we can manage the water resources, mitigate associated hazards, and integrate decision support.

Understanding Ground Water includes understanding regional and continental scale aquifer characterization; ground water barriers, flux and storage; subsidence mapping and hazard mitigation; spatial, temporal, & magnitude control on land subsidence; hydro-tectonics including separating anthropogenic and tectonic subsidence sources, aquifer poro-elastic response to earthquakes and liquefaction; aquifer storage and recovery; and subsidence mitigation. To accomplish this will require L-band observations with sub-monthly acquisitions and comparisons with current archives to establish long-term trends.
To understand Surface Water requires monitoring of vital spatial & temporal changes in surface water resources through areal extent and volume measurements; resolving velocity and circulation patterns in rivers and water bodies; flood, inundation, and hazard assessment; spatial & temporal extent of water level, velocity patterns, biogeochemical production in tropical areas, inundated regions correlated with methane production, natural and human oil spills; ice-debris accumulations, failure locations, and the resulting flood levels; wetland water flow, engineering structures, and water levels and all weather tools for monitoring surface water extent during events. To accomplish this requires L-Band missions with daily to sub-daily acquisitions; L- and C- Band 5-day interferometric repeat time; near-nadir look angles to maximize water body backscattering; along-track interferometry for velocity measurements; VV, HV, VH, HH polarization for vegetation; and comparisons with current archives to establish long-term trends.

A roadmap has been generated to identify the technologies required, and is shown in Figure 5. However it is to be noted that the technologies identified as needed are very modest--apparently no one thinks in terms of advanced technologies such as antennas larger than a few tens of meters. Therefore we concentrate on science community requirements, since they, not NASA technology roadmaps, will set the requirements for future systems. These requirements are approached in the following paragraphs.

![Technology Table](image)

Figure 5

To understand Soil Moisture, which is an essential parameter in measuring the global water budget, we need climate modeling, and flood, landslide and debris flow

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2 NASA 2004 budget request document support, from NASA web site
forecasting. To understand it therefore we will need detailed 4-D soil moisture maps including the state of the water (permafrost, frozen, liquid, etc); how do natural and anthropogenic processes impact the dynamic soil moisture supply; what is the link between ecology, land-use, and soil moisture; and what is the depth distribution of soil moisture. To accomplish this are needed detailed 4-D soil moisture change maps with multi-wavelength capability for varied penetration depths with minimum depth 10 cm and maximum depth 1-2 meters; full polarization desired; 10 meter postings desired; and temporal coverage ideally daily but minimum weekly.

To understand Snow Pack we need to understand the extent and volume of water in snow pack, which involves resolving the snow-water equivalence is essential for constraining the contribution of the snow pack on the global water budget and forecasting water supply; how snow pack properties vary spatially and temporally; real-time data collection for flood prediction associated with snow melt; and glacial mass balance. To accomplish this requires acquisition of data at L-Band as a minimum but multi-wavelength acquisition preferred; full polarimetry; and at least weekly coverage.

To understand Land Surface Change we need to understand how the land surface varies with time and how it influences the dynamic local-global water supply, especially when radar scatters are significantly altered; what is the source, path, deposition, and volume of material transported during large events; the mechanics, kinematics, monitoring, and prediction of mass movement hazards including landslides, sink holes, mine collapse, floods; the volume of material lost in coastal erosion; the relationship between hydrology and habitat in slope stability in post-fire environments; and how hydrologic parameters vary with topographic change.

Quantitative requirements for all hydrology areas are as follows, gathered from a number of sources listed in the references3:

**Flood Mapping**

All-weather, day/night, high-resolution maps of flood inundated areas. Only regional coverage is required for selected monitoring areas. Tasking shall be rapid (24 hours or less) to respond to flooding events as they occur in U.S. and other regions of interest worldwide. The refresh rate needs to be daily to monitor rapidly changing flood conditions. The threshold output product is a map of flood inundated area. The objective product is a map of water depth in the flooded area.

<table>
<thead>
<tr>
<th>Systems Capabilities</th>
<th>Thresholds</th>
<th>Objectives</th>
</tr>
</thead>
<tbody>
<tr>
<td>a. Horizontal Cell Size</td>
<td>25 – 100 m</td>
<td>25 – 100 m</td>
</tr>
<tr>
<td>b. Mapping Uncertainty</td>
<td>25 m</td>
<td>10 m</td>
</tr>
<tr>
<td>c. Measurement Range</td>
<td>flooded/not flooded</td>
<td>flooded/not flooded and water depth</td>
</tr>
<tr>
<td>d. Measurement Precision</td>
<td>25 m</td>
<td>25 m</td>
</tr>
<tr>
<td>e. Measurement Accuracy</td>
<td>TBD</td>
<td>TBD</td>
</tr>
<tr>
<td>f. Refresh</td>
<td>24 hours</td>
<td>12 hours</td>
</tr>
<tr>
<td>Long-Term Stability</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Latency</td>
<td>N/A</td>
<td>N/A</td>
</tr>
</tbody>
</table>
| Geographic Coverage           | US                         | All continents              | 3 NPOESS technical requirements document, Version6, 4 April 2001
Landscape freeze/thaw state determined at sufficient spatial resolution and temporal fidelity to enable discernment of temporal dynamics in seasonal freeze/thaw processes for application to monitoring of terrestrial productivity and snow hydrologic dynamics, and integration with weather forecast models. Applications include monitoring of seasonal and interannual thaw cycles for parameterizing the land surface in weather forecast models, quantification of vegetation growing season length and carbon flux dynamics, and monitoring boreal flooding for hydrological management. All-season, all-weather capability is a necessity; active microwave sensor is required. Thresholds for microwave products spatial accuracy need only match the level of performance of related MODIS products. Automated processing is required for near real-time meteorological applications.

<table>
<thead>
<tr>
<th>System Capabilities</th>
<th>Thresholds</th>
<th>Objectives</th>
</tr>
</thead>
<tbody>
<tr>
<td>Coverage</td>
<td>All regions north of 40 degrees north latitude (pan-boreal coverage)</td>
<td>All regions north of 30 degrees north latitude, and regions higher than 1000 m.</td>
</tr>
<tr>
<td>Horizontal Resolution</td>
<td>1 km</td>
<td>250 meters</td>
</tr>
<tr>
<td>Refresh (Temporal revisit based on required temporal delineation of seasonal dynamics)</td>
<td>48 hours</td>
<td>12 hours</td>
</tr>
<tr>
<td>Measurement accuracy (expressed as percent error of observed landscape freeze/thaw state)</td>
<td>10%</td>
<td>5%</td>
</tr>
<tr>
<td>Measurement precision (expressed as percent of dynamic variation between frozen and thawed states)</td>
<td>10% (corresponds to 0.5 dB over a 5 dB dynamic signal)</td>
<td>10%</td>
</tr>
<tr>
<td>Mapping accuracy (Geolocation accuracy)</td>
<td>250m</td>
<td>60m</td>
</tr>
</tbody>
</table>
Ocean Salinity

A measure of the quantity of dissolved materials in sea water. A formal definition is “the total amount of solid materials, in grams, contained in one kilogram of sea water, when all the carbonate has been converted to oxide, the bromine and iodine converted to chlorine, and all organic matter is completely oxidized. Units of measurement are parts per thousand, by weight.”

<table>
<thead>
<tr>
<th>Thresholds</th>
<th>Objectives</th>
</tr>
</thead>
<tbody>
<tr>
<td>a. Vertical Coverage</td>
<td>N/A</td>
</tr>
<tr>
<td>b. Horizontal Cell Size</td>
<td>0.25 km</td>
</tr>
<tr>
<td>c. Vertical Cell Size</td>
<td>2 m</td>
</tr>
<tr>
<td>d. Mapping Uncertainty</td>
<td>0.25 km</td>
</tr>
<tr>
<td>e. Measurement Range</td>
<td>N/A</td>
</tr>
<tr>
<td>f. Measurement Precision</td>
<td>N/A</td>
</tr>
<tr>
<td>g. Measurement Accuracy</td>
<td>0.5 ppt</td>
</tr>
<tr>
<td>h. Maximum Local Average Revisit Time</td>
<td>N/A</td>
</tr>
</tbody>
</table>

Coastal Salinity

Coastal coverage refers to the areal extent consistent with the US Exclusive Economic Exploitation Zones (EEZs) which extend 200 nm from shore. Coastal coverage shall entail roughly 300 km swath coverage, but pertains to all coasts worldwide to support civil and military observations. Measure the quantity of dissolved materials in sea water. A formal definition is “the total amount of solid materials, in grams, contained in one kilogram of sea water, when all the carbonate has been converted to oxide, the bromine and iodine converted to chlorine, and all organic matter is completely oxidized. Units of measurement are parts per thousand (ppt), by weight.”

Systems Capabilities

<table>
<thead>
<tr>
<th>Thresholds</th>
<th>Objectives</th>
</tr>
</thead>
<tbody>
<tr>
<td>a. Vertical Coverage</td>
<td>0 to 300 m</td>
</tr>
<tr>
<td>b. Horizontal Resolution</td>
<td>0.25 km</td>
</tr>
<tr>
<td>c. Vertical Cell Size</td>
<td>2 m</td>
</tr>
<tr>
<td>d. Mapping Uncertainty</td>
<td>0.25 km</td>
</tr>
<tr>
<td>e. Measurement Range</td>
<td>0 to 40 ppt</td>
</tr>
<tr>
<td>f. Measurement Precision</td>
<td>0.5 ppt</td>
</tr>
<tr>
<td>g. Measurement Accuracy</td>
<td>0.5 ppt</td>
</tr>
<tr>
<td>h. Refresh</td>
<td>3 hours</td>
</tr>
<tr>
<td>i. Long-Term Stability</td>
<td></td>
</tr>
<tr>
<td>j. Latency</td>
<td>15 (TBR) minutes</td>
</tr>
</tbody>
</table>

Coastal Sea Surface Temperature (SST)

Coastal coverage refers to the areal extent consistent with the US Exclusive Economic Exploitation Zones (EEZs) which extend 200 nm from shore. Coastal coverage shall entail roughly 300 km swath coverage, but pertains to all coasts worldwide to support civil and military observations. Sea surface temperature is defined as the temperature of the surface layer (upper 1 meter) of ocean water. It has two major applications: 1) sea surface phenomenology, and 2) use in infrared cloud/no cloud decision for processed cloud data. The requirements below apply only under clear conditions for selected lakes, rivers and coastal regions that require high resolution data.

Systems Capabilities

<table>
<thead>
<tr>
<th>Thresholds</th>
<th>Objectives</th>
</tr>
</thead>
<tbody>
<tr>
<td>a. Horizontal Cell Size</td>
<td>0.1 km</td>
</tr>
<tr>
<td>b. Worst case</td>
<td>0.2 km</td>
</tr>
</tbody>
</table>
b. Mapping Accuracy

<table>
<thead>
<tr>
<th></th>
<th>Nadir</th>
<th>Worst case</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>0.1 km</td>
<td>0.2 km</td>
</tr>
</tbody>
</table>

c. Measurement Range

<table>
<thead>
<tr>
<th></th>
<th>-2° to 40° C</th>
<th>-2° to 40° C</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>0.4 °C</td>
<td>0.1 °C</td>
</tr>
</tbody>
</table>

d. Measurement Precision

<table>
<thead>
<tr>
<th></th>
<th>0.7° C</th>
<th>0.2° C</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

e. Measurement Uncertainty

<table>
<thead>
<tr>
<th></th>
<th>12 hours</th>
<th>6 hours</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>f. Refresh</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>g. Long-Term Stability</td>
<td>TBD</td>
<td>Same as Threshold</td>
</tr>
<tr>
<td>h. Latency</td>
<td>2 hr</td>
<td>1 hr</td>
</tr>
<tr>
<td>i. Geographic Coverage</td>
<td>U.S. coastal areas and selected coastal areas and lakes worldwide</td>
<td>Same as Threshold</td>
</tr>
<tr>
<td>j. Orbit Constraints</td>
<td>Sun Sync Polar</td>
<td>Sun Sync. Polar</td>
</tr>
</tbody>
</table>

b. Current and programmed space missions

In order to understand what the capability in the water cycle/hydrology areas will be in the next decade, and therefore to understand how the science needs will or will not be met in this time frame it is necessary to understand the advanced space systems that are in development now. The three new missions are the NASA\(^4\) Hydros mission which will generate a global map of continental soil moisture, the NASA Aquarius mission to generate a global map of sea surface salinity, and the ESA SMOS mission which will generate maps of both soil moisture and sea surface salinity. Each is briefly addressed below.

**Hydros**

The Hydros NASA mission\(^8\) will provide the first global view of the earth’s changing soil moisture and land surface freeze/thaw state. These together define the land hydrosphere state. Its science mission objectives are shown in Figure 6.

---

4 http://hydros/gsfc/nasa/gov
5 Hydros science returns: Dara Entekhabi, MIT team PI
6 Hydros: A NASA Earth Science Pathfinder. NASA GSFC site
8 HYDROS Science Team Meeting November 3 and 4, 2004 Hilton Hotel, San Francisco
Figure 6

The spacecraft is shown in Figure 7. It is scheduled to be launched in 2009.
Aquarius

The Aquarius mission responds to this need for global observation of SSS. Its science goals are to observe and model the processes that relate salinity variations to climatic changes in the global cycling of water and to understand how these variations influence the general ocean circulation. By measuring salinity globally and synoptically for 3 years, Aquarius will provide an unprecedented view of the ocean’s role in climate. The Aquarius science objectives address the important new climate information that will be gained from salinity measurements, and also provide new insights into ocean circulation and mixing processes. Aquarius will resolve unknown patterns and variations of the global SSS field, especially in large under-sampled regions. Aquarius will provide an important reference from which longer-term climatic ocean changes will be detected in the future.

*Water Cycle:* Aquarius will measure spatial and temporal salinity variations to determine how the ocean responds to varying evaporation-minus-precipitation (E-P) surface-water fluxes, ice melt and river runoff on seasonal and interannual time scales.

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Ocean Circulation and Climate: Aquarius will investigate how salinity variations modify ocean density and influence density-driven circulation and heat flux in three latitude zones:

- **Tropics:** Air-sea interactions and climate-feedback processes, El Niño/La Niña variations.
- **Mid-Latitudes:** Formation processes of surface-mode waters and their subduction into the ocean interior. (Surface-mode waters carry unique surface temperature and salinity signatures to intermediate depths and serve as tracers for ocean interior circulation.)
- **High-Latitudes:** Salinity anomalies that influence the ocean’s large-scale overturning circulation and have lasting impacts on climate. Ancillary science objectives include analyzing air-sea CO2 flux, monitoring sea-ice concentration, and retrieving soil moisture while the satellite is over land.

The Aquarius mission is shown in Figure 8.

![Aquarius spacecraft](image)

**Figure 8**

Key Aquarius Instrument Facts

- **Instrument Type:** L-Band Radiometer and Scatterometer
- **Scan:** 3-beam push broom with 350-km swath width
- **Accuracy:** 0.2 psu rms, monthly, at 100 km resolution
- **Calibration:** Global ocean-observing system surface *in situ* salinity observations (>10,000/month)
- **Duty cycle:** Continuous
- **Data Rate:** 500 Mb/day
- **FOV:** ±10.3° (~300 km swath), center displaced 29° off nadir to shadow side of 6 a.m./6 p.m. orbit
- **Incidence Angle:** 23.9°, 33.8°, 41.8° (3 different fixed antenna beams)
- **Instrument IFOV:** ± 2.6° each beam (64°—74 km, 77°—92 km, 88°—118 km)
- **Antenna aperture:** 3-m parabolic reflector, offset feeds.
- **Temperature Resolution:** Radiometer Brightness Temperature (Tb): 0.15 K (calibration), 0.06 K noise equivalent delta temperature (NEDT)
- **Temporal Resolution:** Minimum 8 samples per month per 100-km square at equator
- **Transmission Frequency:** Scatterometer at 1.26 GHz, polarimetric Repeat Cycle: 7 days

The Aquarius spacecraft is scheduled to be launched in 2008.
SMOS

The European Space Agency is in development of the second in their series of earth observing spacecraft, the SMOS (Soil Moisture and Ocean Salinity). This will be a LEO spacecraft featuring an Interferometric antenna\textsuperscript{10}. It is illustrated in Figure 9

![Figure 9](image)

The mission parameters of SMOS appear below.

- Launch: early 2007
- Duration: Minimum 3 years
- Instrument: microwave imaging Radiometer using Aperture Synthesis
- Instrument concept: Passive microwave 2-D interferometer
- Frequency: L-Band (21 cm – 1.4 GHz)
- Number of receivers: 69
- Receiver spacing: 0.875 lambda
- Polarization: H + V, polarimetric optional
- Spatial resolution: 35 km at centre of field of view
- Tilt angle: 32.5 degrees
- Radiometric resolution: 0.8-2.2 degrees K
- Temporal resolution: 3 days revisit time at Equator
- Mass: 683 kg
- Orbit: Sun synchronous, 763 km altitude

c. Requirements implications for new space systems

A number of papers have been written on future directions for remote sensing\textsuperscript{11}, and desirable for ocean remote sensing and soil moisture radiometry\textsuperscript{12}. The most important

\textsuperscript{10} Mission objectives and scientific requirements of the Soil Moisture and Ocean salinity (SMOS) mission, Version5.
European Space Agency web site.
capabilities collected from these three new systems insofar as they meet science desiresments are resolution, sensitivity, revisit time, scan pattern, and coverage. There is no data on affordability but it is clearly a most important consideration. We therefore derive a set of important criteria against which to measure the three space systems in development, and against which the capabilities of the proposed new system concept can be measured. This will lead to a direct comparison between currently planned missions and the proposed concept, and will enable a set of science-driven requirements to be evolved against which the new concept can be designed.

- High resolution on the ground
- High sensitivity
- Rapid / frequent revisit
- Flexible scan area and pattern, or continuous dwell
- Coverage of nearly a hemisphere from one space system
- Affordable

To that end a comparison of the desired and attained capabilities of the three new space systems were placed in a table, shown as Figure 10. From this figure it becomes clear where the shortfalls exist between the capabilities of current and programmed systems and the desired science-driven objectives.

For measurements of soil moisture for science purposes the sensitivity, revisit time, and operating frequencies are adequate, but the resolution is not, since 35 kilometers spot sizes are typical whereas 100-300 meters are desired. For soil moisture for trafficability observations the sensitivity and frequency are adequate, but the resolution is not, just as for earth science, and in addition the revisit time is not adequate since revisit times of a maximum of one day are desired. For ocean salinity measurements in the deep sea the sensitivity and frequency are adequate, but the resolution needs to be at least 1 km instead the 35 km of present instruments, but the revisit time is adequate. However for coastal salinity measurements the resolution needs to be 100 meters and the revisit time needs to be hours, not the several days of the three new space systems.

Thus the shortfalls are identified as principally in resolution and revisit time. In addition for coastal hydrology flexible scan patterns would be desired, but there are no documented requirements for a flexible scan pattern because no one thought that such a capability was feasible.

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12 “Future directions in ocean remote sensing”. Phillip.R. Schwartz, The Aerospace Corporation, Chantilly, VA. (publication imminent)
As a result of the foregoing the system conceptual definition will be carried out responding to the shortfalls indicated in the Figure XXX. While it is recognized that there are other missions and capabilities that the concept could well also satisfy, addressing them will be deferred to Phase II where a concentrated attack can be made on meeting such needs, actual and potential.

A number of contacts were made at NASA, Aerospace Corporation, and numerous web sites of government agencies were accessed, including NASA Hq. and Centers, NOAA, and others to collect science phenomenology, needs present and future, roadmaps, and other information from which to understand potential requirements and needs that the proposed system concept, heretofore unavailable, could perform

The principal contacts made in collecting and assessing the science requirements discussed in this chapter include:

**NASA Headquarters:**
Granville Paules  
John LaBreque  
Craig Dobson  
Eric Lindstrom  
Jarred Entin

**NASA Goddard Space Flight Center**  
Edward Kim

**JPL**  
Kyle McDonald  
Anthony Freeman
2. CONCEPT AND ARCHITECTURE DESCRIPTION

The basic functions and concept definition for radiometric measurement of soil moisture and sea surface salinity are described here for a system concept operating from GEO, which is capable of revisits in as little as a few hours simultaneously with attaining a resolution spot size of 100-300 meters.

A discussion of the reasons to choose GEO is necessary at the outset.

Stationing in LEO: While in principle a LEO system can provide the required frequent revisits simultaneously with 100-300 meter resolution spot size, a very large number of constellation spacecraft, numbering in the many dozens, would be required and each would have to have an antenna with diameter of 300-1,000 meters. Such antennas, while smaller than that required for the GEO concept, are still not attainable with current or future filled aperture techniques and would need to be implemented with the same sparse array picosats technique that will be used in the new Phase I concept. The environmental disturbances are the most severe in LEO, requiring the most propulsion in the picosats for formation flying. However the presence of dozens of such constellation systems in LEO, each with many thousands of picosats, would create a huge orbital collision danger and orbital debris environment. In addition the aggregate cost due to the many dozens of such constellation systems would be prohibitive. Thus, while LEO systems would be possible they would not be practical or affordable.

Stationing in MEO: MEO systems would suffer from some of the same problems as LEO systems though fewer numbers would be required, probably in the 10-20 constellations number. Each would require a constellation antenna of 3-10 km in diameter. In addition they would operate in a very bad radiation environment which would create design difficulties and life limitations. Though the disturbances are more moderate they still will require propulsive corrections, just with somewhat smaller fuel loads. The total systems would still be extremely expensive.

Stationing in GEO: A system operating in GEO or GSO, though requiring the largest constellation size and number of picosats to attain the required resolution, is no more difficult to implement than the smaller systems at lower altitudes. The environment is the most benign, and picosat propulsion requirements would be minimized. There would only need to be one such constellation for almost a hemisphere of coverage, could provide frequent, even continuous coverage and revisits, and its constellation, though large, is no more difficult to implement than
those for the lower altitudes. In addition the orbital debris and collision environment is moderate and more tractable.

It is for these reasons that a space system located in GEO, or perhaps GSO, was chosen as the only means by which the resolution and timeliness found to be required in Chapter 1 could be simultaneously, practically, and affordably attained. These reasons are summarized in the Figure 11.

![Figure 11](image)

The chapter begins with a derivation of the required sensitivity of a passive soil moisture radiometer operating in GEO.

**Soil moisture radiometry**

Passive hydrology radiometers are sensitive to the brightness temperature of the earth, which is in turn sensitive to the amount of water in the soil. The radiometer reads a signal proportional to the total temperature its antenna sees. This temperature is $T_A = T_{\text{sys}} + \eta T_B$, or the sum of the internal temperature (proportional to internal noise) and the brightness temperature. This brightness temperature is $T_B = kT_{\text{soil}}$ and the soil’s temperature is dependent of the percentage of water in the soil. In effect the radiometer will be sensitive to the emissivity of the soil, which is

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13 “Comparison of soil moisture retrieval algorithms using simulated HYDROS brightness temperatures”, P. O’Neill, Hydrological Sciences Branch, NASA Goddard Space Flight center, Greenbelt, MD 20771; and W. Crow, A. Hsu, USDA ARS; and Njoku, T. Chan, JPL; and JC Shi, UCSB.
directly relatable to % saturation of the soil by water, and is the desired quantity to be observed. Another way to discuss the sensitivity of a radiometer is to define the target temperature change which is equivalent to its internal noise (kTB). Then the effective noise temperature is the minimum change in temperature that will be detectable, which is identical to the internal receiver noise level, measured in degrees Kelvin rather than watts. This is from \( W = kTB \) where \( k \) is the Boltzman’s constant, \( T \) is the temperature, and \( B \) is the receiver bandwidth, since \( T = W/kB \).

In practice then the Noise Equivalent Temperature change, or \( NE\Delta T \) is the usual measure of instrument sensitivity for passive radiometers. An instrument sensitive to 1 degree K temperature changes will in practice be able to resolve 1% changes in the saturation level of water in the soil, there being about 100 degrees K observed difference between completely dry and completely saturated soil. The radiometer in effect measures the percent of saturation level in the soil in terms of volume--that is \( \Delta T = \frac{1 K}{0.01 m^3/m^3} \). Typical space instruments have sensitivities of 0.05 to 3 degrees K.

In practice the \( NE\Delta T \) of an instrument is about
\[
\Delta T = \frac{600}{\sqrt{Bt}}
\]
where \( B \) is the receiver bandwidth, \( t \) is the integration time of the measurement, and 600 is the approximate total temperature of the receiver including internal noise, antenna noise, and a variety of background noise sources, which in effect is double the approximately 300 K receiver temperature. Taking a bandwidth of 10 MHz as typical (narrower means less signal and wider means more interference with other signals in same microwave region around 1.4 GHz),
\[
\Delta T = \frac{0.2}{\sqrt{t}}.
\]
For a typical integration time of 0.3 seconds such as SMOS will have a \( \Delta T = 0.36^\circ K \).

**Sea Surface Salinity**

The basis for remote sensing of Sea Surface Salinity (SSS) is the dependence of the dielectric constant of sea water on salinity at microwave frequencies. The dielectric constant determines the surface emissivity \( (e) \), and this determines the measurable parameter, the brightness temperature \( T_b \), by the relationship \( T_b = e \) (SST), where SST is the physical temperature of the seawater. At L-band (1.4 GHz) for values of SSS (32–37 psu) and SST typical of the open ocean, the dynamic range of \( T_b \) is ~4 degrees K. The 1.413 GHz frequency was chosen because of its sensitivity to salinity and it is in a protected radio-frequency band. The SSS sensitivity is almost negligible above 3 GHz. At much lower frequencies the larger antenna size, and ionospheric and RF interference, make the measurement impractical. At L-band frequencies the penetration depth of the ocean surface is about 1 cm. Salinity measurements are in practical salinity units (psu), in which the total dissolved salt capacity of sea water is about 2 psu. To gain needed sensitivity we need to measure 0.1-0.2 psu, which requires an instrument sensitivity of \( \Delta T = 0.05 - 0.1^\circ K \). Density of coastal water is inferred from salinity measurements so the measurements are important both in the coastal areas and the deep oceans.
The initial system concept as conceived and proposed is shown in Figure 12.

![Figure 12](image)

The concept has two principal features not available in current or planned space earth observation systems:

1. It is a passive radiometer, much as are most spacecraft radiometers, except it is located in GEO. It is therefore capable of continuous or frequent observations of small and large areas in a hemisphere, not possible with spacecraft in LEO.

2. The concept has at least 2 orders of magnitude better surface resolution in order to obtain detail information not otherwise possible

3. It is aimed at both soil moisture and sea surface salinity observations

The fundamental problem arising from these desired capabilities is to obtain a smaller spot size from GEO than from LEO, which clearly will require a very large antenna size since two orders of magnitude size increase are needed to obtain the same resolution from GEO as from LEO, and in addition at least two more orders of magnitude size increase are required to obtain the desired small spot size. Therefore the spacecraft antenna must be four to five orders of magnitude larger than those in LEO on current or planned spacecraft.

The antenna diameter required in GEO to resolve a given size spot at 1.4 GHz is shown in Figure 13.
Since the targets are ground spot sizes in the order of 100-300 meters it is clear that the antenna diameter will have to be in the order of 30-100 kilometers at 1.4 GHz in GEO. The frequency used cannot be very different because water interacts well with radiation at that frequency, penetrates sufficiently deep into average soil that moisture content in a volume can be measured, rather than just a surface film which would give a false measure of soil saturation, and operates in a protected band to minimize interference. Antennas of that size need special consideration and indeed are at the heart of this concept. Figure 14 illustrates the difficulty of implementing such very large antennas using conventional or even advanced technologies.
A conventional rib-mesh deployable antenna 100 km in diameter would weigh an astronomical 120,000,000,000 kg. An antenna of that size would weigh 12,000,000,000 kg if built using the most advanced inflatable technologies, similar to those of the NASA/JPL Inflatable Antenna Experiment which was tested in space with limited success. Even if the film were cut away, leaving only 10% of the area so as to form a 10% sparse antenna, it would still weigh 1,200,000,000 kg. Thus none of these techniques make for a feasible antenna. There must be a better way.

Indeed the “better way” is to replace precision and heavy structures with information, a fundamentally advantageous technique advocated by Ivan Bekey, the PI, for decades. In this case that means forming a very sparse array of small spacecraft containing repeaters whose combined signals are sent to a central receiver, arrayed in a constellation of the required diameter in space. There is neither a structure nor film forming the aperture, nor is there a truss connecting the aperture to the receiver—all elements are formation-flown in space. This technique is illustrated in Figure 15.
The characteristic pattern of this type of antenna is shown in Figure 16.

The overall diameter of the antenna constellation determines the size of the spot on the ground, as expected. The amplitude of the near-in sidelobes is determined by the illumination of the array, that is the amplitude taper from the edges to the center. The amplitude of the far sidelobes below that of the mainlobe is determined by the inverse of the number of radiating elements in the array. The generation of grating lobes is avoided by randomizing the position of the radiators in the array. These are well-known characteristics of sparse aperture antennas. Thus the antenna characteristics can be tailored by its design, and there is no reason why it could not be made to serve as the receiving antenna of this concept. The only real question is whether the near-in sidelobes can be reduced sufficiently so that their signal does not overpower the mainlobe signal. This should be readily attainable by the amplitude taper imposed by the illuminating feed. Uniform illumination results in the first sidelobe being 13.5 dB below the mainlobe, but tapers such as a Taylor taper can reduce the first sidelobe to levels below 25 dB to 45 dB below the mainlobe, clearly solving the problem.
To demonstrate that the pattern of Figure 16 is realistic a sample pattern calculation using a rigorous model was made in a related space program, on a smaller but sparse antenna, which contained fewer elements than that of the present constellation. The resulting pattern, shown in Figure 17, does indeed have the general characteristics as the idealized pattern shown in Figure 16, though the lobe widths and levels are different. The antenna will be completely designed in Phase II.
The antenna must be focused on the ground, rather than the usual focus on infinity, in order to achieve the desired small spot sizes. This is because the smallest spot size that an antenna can attain in the normal far field mode cannot be smaller than the diameter of the antenna itself, which extends out to the Rayleigh distance, and then is set by the diffraction limit beyond that distance. The Rayleigh distance is usually taken to be about \( \frac{2D^2}{\lambda} = 1 \times 10^{11} m \). Since the distance from GEO to the ground is 3.6 x10^7 m it is clear that the antenna is operating in the near field, and well within the Rayleigh distance, and therefore it can be focused. Thus the minimum spot size attainable is set by the diffraction limit rather than by the antenna diameter, and is 100 m for a 100 km constellation. Focusing the antenna is done by the usual technique of changing phase of the distributed radiators across the array so as to represent a conic phase surface. The phases required across the antenna can be set open loop because the focusing distance is just the distance from the spacecraft to the ground area being observed, which is completely known from simple geometry and does not vary in operation. This focusing is illustrated in Figure 18.
Having defined a feasible concept for implementing the antenna, which is the heart of the concept, allows definition of the rest of the system concept. It is vital to note at the outset of this process that whether the antenna array is operated as a conventional phased array, a space-fed phased array, or even as separate coherent receivers with signals combined digitally, its functioning will be the same. The type of antenna will be discussed further later.

The initial system concept envisioned the system operating as a passive receiver, much as most other soil moisture and ocean salinity instruments in space. The first task therefore is to examine the validity of the assumption that there is sufficient signal in GEO for a passive radiometer operating mode. Passive radiometers operate by receiving black body thermal radiation\textsuperscript{14} from the earth. Since the earth’s temperature is on the average roughly 300 degrees Kelvin, its radiation will have the familiar form shown in Figure 19.

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\textsuperscript{14} The Thermal Radiation Formula of Planck (1900), Luis J. Boya, Departamento de Física Teórica, Facultad de Ciencias, Universidad de Zaragoza.— 50009 Zaragoza, Spain
Referring to that figure it is clear that the preferred operating frequency of 1.4 GHz is very far from the peak of the black body radiation, which occurs roughly at 10 microns or 30,000 GHz. Because of that the “signal”, which is the radiation at the desired frequency, is 7 orders of magnitude below the peak, and very weak. We therefore calculate the signal strength arriving at the GEO spacecraft and determine the space system parameters needed for that signal to exceed internal receiver and antenna noise. We start with the Planck’s black body radiation:

\[
B_s = \frac{2hf^3}{c^2} \left[ \frac{1}{e^{(hf)/(kT)}} - 1 \right]
\]

whose units are \(W/m^2\text{srHz}\) or \(W/sr\text{Hz}\). This is the thermal energy radiated by an area \(m^2\) into an angle \(sr\) steradians, per Hz system bandwidth. The constants are the speed of light, \(c = 3 \times 10^8 m/s\), the frequency \(f = 1.4 \times 10^9 Hz\), Planck’s constant \(h = 6.6 \times 10^{-34} Js\), Boltzman’s constant \(k = 1.38 \times 10^{-23} J/K\), and the approximate temperature \(T = 300^oK\).

Evaluating the equation we get that the outgoing energy from the ground spot being observed is \(B_s = 1.8 \times 10^{-19} \frac{W}{m^2\text{srHz}}\). Now letting the ground radiating spot be a desirable 300 meters diameter its area is \(7.07 \times 10^4 m^2\). We let the bandwidth be a modest 10 MHz so that \(B = 1 \times 10^7 Hz\), because wider bandwidth could result in more interference from cellular phones, GPS, and other devices operating in the same frequency spectrum. The solid angle subtended by the antenna must be calculated. It is \(sr = \left(\frac{d}{R}\right)^2\) radians, where \(R\) is the range from GEO to the surface \(R = 3.6 \times 10^7 m\) and \(d\) is the equivalent diameter of the receiving antenna effective capture area. This needs expansion: the constellation area is not the antenna receiving area. The effective receiving aperture area is the capture area of each picosat antenna multiplied by the number of picosats. In practice this area will be many orders of magnitude smaller than the area of the constellation given its very large diameter. In effect the constellation diameter sets the
footprint or spot size while the effective receiving aperture sets the sensitivity of the antenna to incoming signals. This area is just \( a = d^2 \frac{\pi}{4} \) where \( d \) is the diameter of this equivalent capture area. Then substituting these values we get the received energy \( E = 9.82 \times 10^{-23} d^2 t \) watt seconds given the effective capture area of the antenna. This is illustrated in Figure 20.

![Figure 20](image)

Now we determine the level of noise this signal must overcome to be detected. Note that the system noise is the sum of the receiver internal thermal noise, the noise equivalent of the temperature of the antenna and the background, and that of the antenna feed and waveguide components. The receiver noise is \( N = kTB \) where \( k \) is Boltzman’s constant, \( T \) is the temperature, and \( B \) is the system bandwidth, which was set at 10 MHz for the radiator and the same must be used for the receiver. Thus the receiver noise will be \( N = 4.14 \times 10^{-14} W \). However we never have a perfect receiver so we add 1.5 dB of excess noise and 0.5 dB of losses in the circuits so that the noise level rises to \( N = 6.55 \times 10^{-14} W \). This is equivalent to a system temperature of 475 degrees K.

Now in order for the flux from the ground to be detectable it has to exceed this noise level. We let the needed signal level be equivalent to a 600 deg K temperature, so its level must be at least \( S = 8.27 \times 10^{-14} W \). We can then determine the required effective receiving area of the picosats so as to receive this level of power from the ground. To do that we can set

\[
S = 8.27 \times 10^{-14} W = E = 9.82 \times 10^{-23} d^2 t
\]
and let t=1 second and then solve for d. Thus \( d^2 = \frac{8.27 \times 10^{-14}}{9.82 \times 10^{-23}} = 8.42 \times 10^8 \text{ m}^2 \) and the equivalent area must be \( a = \frac{d^2 \pi}{4} = 6.61 \times 10^8 \text{ m}^2 \). That is we need an equivalent effective capture area of the receiving pico-sats taken together to equal this number. But the receiving area of each pico-sat, if outfitted with omnidirectional antennas so as to free it from complex attitude control, is simply \( a = \frac{\lambda^2}{4\pi} \) which at a wavelength of 21 cm (1.4 GHz) is \( a = 3.51 \times 10^{-3} \text{ m}^2 \). Therefore the number of pico-sats that would be required to form the required effective capture area is \( p = \frac{6.61 \times 10^8}{3.51 \times 10^{-3}} = 1.88 \times 10^{11} \) pico-sats. This is clearly an impractical number of pico-sats. It can be lowered somewhat by using a downward facing antenna with some gain on each pico-sat, though that requires at least 2 axis stabilized pico-sats. Even then there will probably be a limit of about 20 dB for the gain of such antennas simply due to size limits at 1.4 GHZ if the tail is not to wag the dog, and in order to have their antenna pattern cover much of the earth so as to avoid having to point each pico-sat to point the beam. Thus we can subtract at most about 20 dB from these numbers, or reduce them by a factor of 100. Thus we get a reduced number of pico-sats of “only” \( p = 1.88 \times 10^9 \) pico-sats. While better it is still an impractically large number.

The required capture area, and thus the number of pico-sats, can only be reduced by increasing the area radiating on the ground. Increasing the bandwidth does help—it does increase the signal but since the receiver bandwidth must be equally increased to receive that signal it also increases the receiver noise by an equivalent amount, and there is no net gain. The number of pico-sats required in a constellation in GEO is thus plotted in the graph of Figure 21.
Referring to this figure it is clear that while the number of picosats in the GEO constellation might be reasonable if the ground area covered by the resolution spot is in the order of 100-1,000 kilometers, the number will be astronomical for spot sizes of 100-300 meters, which are desired for this concept. This conclusion is valid whether an omnidirectional or 20 dB gain antenna is used at each picosat. Even a 30 or 40 dB gain antenna would not change that conclusion, though it would enormously complicate the system. The inescapable reality is that it is impractical to implement passive radiometry from GEO with very small footprints (spot sizes).

It is for this reason that we turn to using an active illuminator to increase signal strength. In this case the signal from the ground will be dependent not on thermal black body radiation from the earth, but rather from the reflected illuminator energy. The fundamental question is whether the use of an active illuminator invalidates the phenomenology of soil moisture and ocean salinity measurement. It does not appear to be so. Whereas the passive radiometer system is sensitive to signal changes due to changes in the emissivity of the soils due to water content changes, the reflection of same frequency signals from the soil is dependent of reflectivity changes. But reflectivity is simply the inverse of emissivity, and so the sensitivity to water content in the soils should be unaffected, except for being inverted--that is 100% saturation will result in the lowest emissivity but highest reflectivity, and vice versa. In fact this is precisely the phenomenology that is used in some systems which use active radar (SAR, InSAR) as a sensing means, though from LEO and for large spatial coverage spots only. The hydrology measurements of these systems required include ECHO Interferometric SAR measurements to characterize ground water in regional and continental aquifers, subsidence mapping and hazard
mitigation, hydro-tectonics, and groundwater management, and are the same as those for passive hydrology.¹⁵

We thus proceed with an analysis of an actively illuminated system concept. In this modified concept we station a continuous wave (CW) source radiating near but not precisely at 1.4 GHz in GEO, equipped with a large conventional antenna which sends its signals to a ground area much larger than the receiver spot but encompassing it. Since the signal is envisioned as CW it will be essentially noiseless, and will not contribute to the receiver noise level as would a radar signal. This revised concept is illustrated in Figure 22.

For this actively illuminated system we can make the receiver bandwidth very narrow because the signal is a discrete frequency, and the receiver needs only to be able to track Doppler and the signals stemming from the dwell time on each ground spot. Since dwell times of 0.3-1 second are anticipated, as is usual on satellite radiometers, a bandwidth of 10 HZ seems a reasonable starting point. In this case we begin the analysis with the internal receiver noise. For a condition of 10 Hz bandwidth and 475 degrees K temperature, including losses, the receiver noise will be

\[ N = kTB = 6.56 \times 10^{-20} W. \]

We want the signal to be at least 3 dB greater, or

\[ S = 1.31 \times 10^{-19} W. \]

Now we note that the signal power originating in a area A on the ground illuminated by a power density of \( P_d \), for soils with average reflectivity of 0.5, 0.3 seconds integration time, and for a spot diameter of 300 meters is

\[ S = \frac{P_d A d^2 \pi}{4 \sqrt{l}} \frac{\pi}{4 \pi R^2} \text{watts} \]

¹⁵ Hydros: A NASA Earth Science Pathfinder. NASA GSFC site
Solving for $P_d$ we get

$$P_d = \frac{S4\pi R^2}{Aa\sqrt{\tau}}$$

which becomes $P_d = \frac{1.1 \times 10^{-7}}{a} W/m^2$

where $a$ is the effective capture area of all picosats, and with equivalent diameter $d$. Now we note that the power density delivered by the illuminator is $P_d = \frac{GW}{4\pi R^2} w/m^2$, where $G$ is the gain of the illuminator antenna and $W$ is its transmitted power. Therefore we can equate the two expressions and obtain

$$\frac{1.1 \times 10^{-7}}{a} = \frac{GW}{4\pi R^2},$$

and solving for $a$:

$$a = 1.89 \times 10^9 \frac{GW}{W/m^2}$$

Now we assume some practical illuminator antenna size and transmitter power. We mimic commercial communications satellites which can readily have 1-10 kW of radiated power and a 10-20 m antenna. In fact all we need is a single CW transmitter rather than dozens of broadband traveling wave tubes, and so the task is much simpler and the equipment much cheaper. A 20 m diameter parabolic antenna operating at 1.4 GHz will have a gain of $G = \frac{5D^2}{\lambda^2} = 4.54 \times 10^4$ or 46.7 dB, and the effective radiated power of an illuminator with that antenna and a 1 kW transmitter will be 45.5 megawatts. Therefore the required picoSat effective capture area defined above must be

$$a = \frac{1.89 \times 10^9}{(4.54 \times 10^4)(1,000)} = 41.6m^2$$

Now since each picoSat’s effective capture area, with omnidirectional antennas, is $3.51 \times 10^{-3} m^2$, then the total number of picosats required will be $n = \frac{41.6}{3.5 \times 10^{-3}} = 11,900$ picosats. That’s much more feasible than any solution of a passive system.

The total number of picosats will vary with the gain of the picoSat antennas, the size of the ground spot being observed, the illuminator power and its antenna gain, and the receiver bandwidth. The number of picosats was calculated for the case of omnidirectional antennas, and appears in the graph of Figure 23.
Likewise the calculation was repeated for picosats with a 50 cm long Yagi antenna and appears in Figure 24.
It is clear that a reasonable number of picosats (1,000-10,000) is readily obtained with an illuminator power of 1 kW and 20 m antenna while achieving a spot size of 100 meters from GEO, if 50 cm long end-fire Yagi antennas are used for the picosats. If the picosats use omnidirectional antennas then the illuminator power must be increased to 10 kW with a 20 m antenna for the same number of picosats, or the spot size increased to 300 meters.

The system concept is therefore deemed feasible if an active continuous wave illuminator is used. It must be pointed out that this is NOT A RADAR in the usual sense. The illuminator does not convey any information that the receiver can use to extract angle, range, or phase information from the return. It only increases the steady state power originating in the spot area over that available from black body thermal self-emission. Thus we may call this concept an augmented radiometer. The illuminator is envisioned as a pure CW signal so that the receiver bandwidth can be very small, however it may be advantageous to “chop” the signal by turning the illuminator signal on and off at a low rate in order for the receiver to be better able to reject steady state noise sources such as cosmic, galactic, etc. This will be examined in Phase II.

3. SUBSYSTEMS/ELEMENTS

The principal subsystems of the concept are the constellation array and its implementation, the central receiver/mother ship, the picosats and their internal components and external antennas, the metrology and navigation system, the tether, the communications and information processing system, the deploy and retrieve system for the picosats, the launch
and orbit transfer system, and the illuminator. Each of these will be described in the following subsections in depth appropriate for Phase I feasibility assessment, but their detail characteristics and performance determination will have to occur in Phase II.

a. Constellation orbits and formation flying

The picosats need to be in a constellation that is both controlled and requires little propellant to maintain the relative positions of the picosats. While it is possible to hold each picosat in a relative position to the others in earth-centered coordinates, say along the local vertical and local horizontal and thereby form a static constellation, the propellant requirements for such stationkeeping in non-Keplerian orbits rapidly become excessive. As an example, the mass of propellants required for propulsively stationkeeping a 100 gram picosat 50 km above the Keplerian GEO orbit for 10 years is 0.825 kg, even with a propulsion system with Isp of 3,000. This is more than 800% greater than the basic picosat mass, and is clearly prohibitive.

The ideal choice is to place the picosats into a set of orbits obeying Hill’s equations in which they describe Keplerian orbits, but that rotate in sub-orbits around a central point on the base orbit when viewed in relative coordinates centered on that point. Since such orbits are Keplerian, their propulsion requirements are minimized. In these equations a particular family of solutions with no constant and secular (in time) terms can be found:

\[
\begin{align*}
x &= 2\xi z_0 \sin \omega t, \\
y &= -\xi z_0 \cos \omega t, \\
z &= z_0 \cos \omega t, \\
dx/dt &= -2\omega \xi z_0 \cos \omega t, \\
dy/dt &= -\omega \xi z_0 \sin \omega t, \\
dz/dt &= -\omega z_0 \sin \omega t
\end{align*}
\]

The parameter \(\xi\) is a scale factor which gives a family of solutions for sub-orbits, and the magnitude of the radius vector \(r\), of the sub-orbit around the center satellite is \(r = (x^2 + y^2 + z^2)^{1/2} = z_0[4\xi^2 + (1-3\xi^2)\cos^2\omega t]^{1/2}\). Among the infinite number of solutions that depend on the value of \(\xi\), only two particular values are of interest in this application. The above equation indicates that the sub-orbit is a circle when \(\xi\) equals \((1/3)^{1/2}\). The second value of \(\xi\) is \(\frac{1}{2}\), which would make the sub-orbit an ellipse whose projection on the plane normal to orbit plane is a circle. Through mathemetic derivations (vector products), one can prove that the sub-orbit is in a plane and the angles between the normal vector of the plane and the \(z\) axis (Earth spin axis) are \(\delta = \arctan(1/\xi)\) which is 60 deg when \(\xi = (1/3)^{1/2}\) and 63.4 deg when \(\xi = \frac{1}{2}\). These angles, of course, mean that the sub-orbit plane will lie at 30 or 26.6 degrees to the local horizontal at the base orbit. Of the two values of \(\xi\), the first choice of 26.6 degrees is preferred because it gives a halo sub-orbit with a constant radius in its orbit around the central point. Furthermore a number of previous studies have shown that a circular halo sub-orbit in its plane requires less propellant for constellation maintenance.

The 1000 to 100,000 picosats may be populated in multiple concentric rings or in multiple spiral arms. It appears that a constellation with multiple concentric rings of halo sub-orbits has the advantage of easy identification of a particular picosat, though a comprehensive comparison can only be made with detailed simulations of both options. Based on a first-cut system design 12,000 picosats can be uniformly populated on concentric rings with largest diameter, or the constellation diameter, being 100 km. A constellation with 31 concentric
halo rings at 1 km spacing in radius is selected as a preliminary example. The spacing along the circumference direction is also set at 1 km. The innermost ring has a diameter of 20 km which allows more than enough room for the central tether and depot. This is another advantage of using the concentric ring option. The orbit elements of three sample picosats in the selected constellations are given in Figure 25, below.

<table>
<thead>
<tr>
<th>Halo ring diameter</th>
<th>Semi-major axis</th>
<th>Eccentricity</th>
<th>Inclination</th>
<th>Ascending node</th>
<th>Argument of perigee</th>
<th>Mean anomaly</th>
</tr>
</thead>
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<tr>
<td>100</td>
<td>42165.9</td>
<td>0.000592</td>
<td>0.05884</td>
<td>0</td>
<td>27</td>
<td>20</td>
</tr>
<tr>
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<td>0.05766</td>
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<td>0.01177</td>
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<td>27</td>
<td>178.32</td>
</tr>
</tbody>
</table>

The three types of natural forces dominating the orbit geometry at geostationary altitude are the Earth gravity harmonics, luni-solar attractions and the solar radiation pressure. These perturbing forces gradually push the picosats away from the desired reference orbits. Numerical results on past analyses show that the position deviation from the reference Keplerian orbit would exceed 100 m in less than two hours, 1000 m in little over 5 hours and 10 km in about 12 hours. However, the relative distance between two picosats remains nearly the same even after 10 days, due to the nearly equal perturbing forces on the two closely separated picosats. Previous analyses and video simulations revealed that, without orbit control, the halo formation remains unchanged for 10 to 15 day, but the whole cluster or the center of the halo suborbit moves outside the control box of a geostationary point, and thus the picosats will require propulsive control. In addition there will be some differential drift. All of this requires propulsive correction of the picosat positions in the constellation, however the total impulse requirements per picosat will be an order of magnitude lower than if the constellation geometry were maintained in the same constellation by thrust in non-Keplerian orbits. In fact estimates indicate that the requirements are certainly less than 100 m/sec per year and may actually be on the order of 5-10 m/s/year.

In these orbits the picosats will describe a set of concentric “sub-orbits” around a point on the GEO orbit. We will refer to them as “halo” orbits. These orbital equations and their visualization in orbit-centered coordinates as a halo are rigorously derived applied to the orbital rendezvous situation, and are accepted in the orbit mechanics community. Such sub-orbits were utilized in the AFRL Techsat 21 program, and in the first implementation of the GPS program, where four GEO constellations of halo orbit satellites were used. These were not implemented for survivability reasons, but they were thoroughly analyzed and their feasibility is unquestioned.

In use the picosats are deployed each into a slightly different orbit, and fly in circles around the GEO point in relative coordinates. They appear to be glued to a transparent plane tilted at some angle to the local horizontal, the preferred angle being 26.6 degrees, for that results in circular sub-orbits and also minimizes the propellant expenditure to stationkeep the picosats.
The locations of the picosats in the constellation are made quasi-random during deployment so that the antenna will not have grating lobes in its radiation pattern. The constellation is illustrated in Figure 26. While this halo constellation is not the only one that can organize and maintain a swarm of picosats in a large volume of space in such a geometry as to act as a refractive antenna lens, it has been studied in detail at the Aerospace Corporation and elsewhere, and appears well matched to the needs of this concept. Other interesting constellation concepts are possible and will be studied further in Phase II prior to settling on a preferred constellation.

**THE PICOSAT ORBITAL CONSTELLATION**

Picosat positions randomized within the rings. Ring radii also randomized

[Diagram of constellation with central receiver, picosat constellation plane, and local horizontal]

Figure 26

**b. Space-fed sparse array antennas**

The large constellation array comprises a set of picosats in a plane. These picosats could be used in a number of ways to comprise the required receiver at 1.4 GHz. The prime requirement is for each picosat to receive the signal from the ground, shift its frequency, and to relay it to a central receiver where all picosat signals are added coherently while the noise is not. The signals thus will be $N$ times the signal of that of any one picosat, and the signal to noise ratio as seen by any one picosat will be multiplied by $N$ (non-coherent addition would only result in $\sqrt{N}$). The requirements to ensure phase coherence and their implications for position and phase or time control are addressed separately below. A conceptually attractive configuration is to use the picosat plane as a space-fed array, with a central receiver above plane of the constellation. In a space fed array the picosats act as
refractive elements akin to a lens in optical systems. The advantage of a space-fed array is
that in principle position errors of the picosats do not, to a first order, cause phase errors in
signals repeated by the picosats and received in a central receiver above or below their plane
because the path length from the ground to the picosat and from the picosat to the central
receiver remain constant with picosat position changes, to a first order, and thus they can be
simple repeaters whose signals are combined in phase at the central receiver. This is not
strictly speaking necessary, of course, as the received signals by each picosat could be sent
to a receiver at the center of the suborbit plane for coherent addition. However in that
configuration the picosat position errors contribute phase errors on a one-to-one basis.
Alternatively the signals could be digitized and sent to the ground for coherent addition, but
that entails thousands or tens of thousands of separate signals, each operating at a very
negative signal to noise ratio, and would be difficult to implement. For those reasons we
concentrate on the space-fed array, and will calculate the performance advantages.

The space-fed sparse array analysis is illustrated in Figure 27. Each picosat receives a small
portion of the essentially plane wave arriving from the ground source. The job of the
picosat is to repeat that signal toward the central receiver, changing its phase or time delay
in the process. The purpose of that change is to convert the incoming plane wave into a
spherical wave converging on the central receiver as a focus where they will be added
coherently. In order for the ensemble of picosats to achieve that the wave passing through
the center of the array must be delayed relative to the wave at the edge of the array, since the
latter wave has further to travel. If the system bandwidth were large true time delay would
be needed in the picosats to prevent dispersion. However the system operates almost as a
CW system, with a bandwidth of 10 Hz at 1.4 GHz, and so phase correction suffices. A
happy side benefit is that since two sine waves cannot be distinguished between phases $2\pi$
apart, the phase corrections can be done by advancing or by delaying the phases, and a $2\pi$
correction range covers all possible phase corrections. This is referred to as Modulo-$2\pi$.

Now consider the actual corrections that will have to be applied by each picosat. First note
that for coherent addition we generally would like the phases of all signals to arrive within
about $1/20$ of a wavelength from each other: $\lambda/20$. At the frequency of 1.4 GHz the
wavelength is 21 cm and so the tolerance will be about 1 cm in position. Thus the position
of each picosat must be known to about 1 cm for phase control, but its actual position
control can be much looser, and will be set by constellation orbit maintenance and other
considerations, not phase control reasons.
Thus a space-fed array acts as a corrective lens in space and does not require that the picosats comprising it be in precise locations 1/20 wavelength from a perfect plane in space. In addition to the corrections for actual picosat positions a global phase shift pattern will be imposed on the picosats, over and above the position control-dicted phase delays, in order to steer the beam and focus the antenna. These global phase patterns will probably be commanded from a central intelligence source collocated at the central receiver.

A very important additional function must be met by the phase control—the compensation for the 30 degree tilt of the constellation plane with respect to local horizontal, which must be compensated by a global phase shift across the array. In effect the beam must be squinted 30 degrees constantly, upon which is superimposed the steering phase shift program for dynamic beam steering array from boresight, as well as the individual picosat position-related phase shifts.

Now let’s derive the actual phase control needed. Consider the picosat on the axis of the array. The total path length to the receiver consists of the path from the ground to the picosat plus that from the picosat to the central receiver. If the picosat should move upward along the axis, the ground-picosat path will lengthen while the picosat-receiver path will shorten. However, on axis the sum of the path lengths is constant regardless of the picosat position. Thus the space fed array is insensitive to picosat position on axis, and in principle would not require any phase correction on axis. This is illustrated in Figure 28.
This complete insensitivity is limited to on-axis picosats, of course. However a similar calculation for an axial position error of a edge picosat at the edge of a 100 km array results in a sensitivity of 3.3 cm motion before the phase error accumulates to the 1/20 wavelength. Similarly an analysis has shown that position errors in the sub-orbit plane are far less sensitive in terms of causing path changes. In fact motions in the sub orbit plane can be 22.4 meters for a center picosat and 26.4 meters for an edge picosat before the phase errors add to 1/20 of a wavelength-- a reduction in sensitivity of a factor of 1/224-1/264. This demonstrates one of the benefits of using a space-fed phased array compared to a purely planar array with central receiver in the plane, in which all picosat positions must be determined to 1/20 of a wavelength regardless of where or in which direction they move--that is its sensitivity is 1/1. The space fed array has other advantaged that will be discussed presently.

It is also important to realize that the modulo $2\pi$ phase compensation can also implement the 30 degree squinting required to focus the energy from the picosats, which are in a plane 30 degrees from local horizontal, on the central receiver located on the local vertical through its center above the array.
c. Phase control commands

The requirement for all picosat signals to add coherently in the central receiver means that they have to arrive in phase. For practical purposes this means that they have to be in phase to about 1/20 of the wavelength, which is equivalent to about 1 cm in free space at 1.4 GHz. As shown above the beneficial characteristics of space-fed arrays loosen this requirement to a worst-case control of 3.3 cm in one direction, and over 20 meters on other directions for picosats at the edge of the constellation and much more for those at the center, so that the worst case requirement loosens to phase control to an accuracy of 1/6 of the wavelength.

Since modulo $2\pi$ control suffices the phase control needs only to have 6 increments. This means that a 3 bit phase shifter is adequate. Such shifters are easy and conventional.

Now the velocity of the outermost picosat in a 100 km constellation is 3.6 m/s relative to the center or to the central receiver. While the velocity is low the phase of the picosat must be adjusted frequently because of the 3.3 cm position tolerance. This means that each picosat must adjust its phase every 10 milliseconds. While that is not a difficult driver for the chipset within each picosat, the global phase control of the ensemble of picosats in the entire array must use a command link from the central unit to each picosat so that phase shift overlays can be commanded for beam steering and beam focusing.

Nonetheless the command link to implement this is simple because the steering commands are relatively infrequent and because the low velocity of the picosats means they require relatively infrequent phase overlay commands. For that matter the beam focusing commands are not expected to vary at all, as they are simply overlays with constant relative phase shift from the center to the edge of the array, and can be set once for every picosat for its lifetime because the expected size of the box within which they will orbit is in the order of 100 m, a very small fraction of the 100 km dimension of the array.

d. Doppler nulling

In order for the signals from each picosat to add with those of the other picosats in the central receiver their phases must be the same within the 1/20 of a wavelength as discussed above. However each picosat travels in its own sub-orbit, and in general will have some velocity with respect to the central receiver most of the time. The component of velocity along the vector from picosat to receiver will produce a Doppler shift of the frequency, and since the Doppler shifts will be different for each picosat the signals cannot be added unless the Doppler shifts are compensated or cancelled.

A technique has been conceived by Bekey Designs, Inc. in activities unrelated to and preceding this NIAC Phase I contract, which achieves complete cancellation of all picosat signal Doppler shifts in constellations of space fed arrays similar to those employed in this concept. This technique is the intellectual property of Bekey Designs, Inc. It will be disclosed to NIAC in the course of a Phase II contract, and utilized in Phase II system definition work. Suffice it to say for Phase I purposes that there will be no Doppler shifts in the picosat signals and therefore are not a feasibility concern for the system concept.
e. Navigation and metrology

The coherent addition of picosat signals at the central receiver requires phase correction of their signals, and that in turn requires precise knowledge of their actual relative position in the array. As shown in section GGGGG above, the position of the picosats must be known to 3.3 cm in one direction as a worst case, and that number was used for all picosats for all directions for conservatism. In this section the means to determine that position are addressed.

Two major options exist. One is to determine the picosat positions with one or more instruments located at the central receiver, such as a radar or lidar, and send the position information to each picosat for its use in setting the phase correction needed. The other is for each picosat to determine its own position independently and then compute its own required phase correction based on that data.

The use of one or more active devices to determine the positions of the picosats has the advantage that the picosats remain simple and inexpensive, however the disadvantage is that of requiring complex, heavy, and likely expensive radar/lidar devices at one or probably more locations around the constellation. While their determination of range and range rate is straightforward, the angle determination is always difficult to do accurately and is many times solved best by triangulation from several such devices. The picosat location must be computed at a central location, however, and then transmitted to all the picosats, which requires complex computation and substantial communications overhead.

The second technique, which requires that the picosats determine their own position precisely relieves the system of the radar/lidar devices, their intercommunications, and their need to communicate with perhaps up to 100,000 picosats, however it does so at the price of having to set up a local navigation environment around the constellation, and at the price of requiring each picosat to have a navigation unit such as a GPS chip built in, which increase power and weight in each picosat. The expense of having up to 100,000 such navigation receivers may also be large.

A comprehensive trade study needs to be done to determine the best technique including variations using different remote sensors and other types of related systems, however for the purposes of establishing feasibility the latter technique is adequate and preferred. A thorough analysis of all options is deferred to Phase II, and the latter technique is illustrated in Figure 29.
This technique establishes a GPS-like local navigation environment around the constellation. It must be emphasized that it is GPS-like, rather than being GPS itself, because the short ranges and the ability to avoid anti-jam and other security features of GPS make for a much simpler system and simpler chips for use in the navigation receivers in the picosats. It also avoids the use of the GPS system itself, which has poor coverage and weak signals in GEO. In this implementation a number of navigation reference units containing stable clocks are placed at the ends of a tether vertically oriented through the constellation center, and a number of other reference units are placed in sub-orbits around the periphery of the array. These could, in principle, orbit in the same plane as the array or at a complementary angle, whichever is better for assuring a diversity of angles for computation of the picosats’ position based on time-difference-of-arrival from the reference units.

The accuracy of such a navigation system can readily be within the 3.3 cm relative needed. This is because the relative velocities are small, being around 4 meters/second maximum, which therefore allow relatively long integration times, the ranges are short assuring high signal/noise ratios all the time, there is no Earth to block the view and so all reference units are visible to each picosat all the time, there is no atmosphere or ionosphere to distort or attenuate the navigation signals, and finally there is little or no interference from the many radiating sources that plague ground-based GPS users.

The navigation electronics in each picosat could consist of newly designed and fabricated chips, or it could use nearly off-the-shelf GPS chips, with some mods. In the former case
there will be the cost of design and production of limited numbers, 100,000 being a tiny production run in the electronics industry. This will make the new chipsets expensive, though they will be lightweight and have modest power needs. In the latter case GPS chips are now mandated to be included in all new cell phones sold by the end of 2005, and thus will be very inexpensive and readily available in any quantity. It is estimated that such chips will cost no more than $10-30. Radiation shielding will have to be added, however, though it must be noted that an off the shelf commercial Cisco internet router has now been in space for a year with no special shielding and has operated normally, so that shielding may or may not be needed for the GPS chips. In addition there might have to be some changes made if the waveform used is not identical to the GPS, which would add to the weight and cost of the chip. It would also draw more power than a new dedicated chipset. Either of these two approaches is feasible and requires a trade study to determine the best choice. Suffice it to say that at least two feasible solutions exists for precision metrology, so that the phase shift can be set within each picosat to the required precision so that the retransmitted signals arrive at the central receiver in phase for coherent addition.

f. Communications

This section will address a number of communications aspects. This first is the determination of the transmitter power required by the picosats to transmit their phase-corrected signals to the central receiver. This will be a 10 Hz bandwidth link, matching the 10 Hz bandwidth of the picosat receivers. The power required for these links will be 0.30 milliwatts if a frequency of 2.5 GHz is used, as an example, as shown in the calculation of Figure 30. There will probably have to be two such channels per picosat in order to maintain polarization information, as that is an important factor in observing the earth phenomena. Thus the total transmit power per picosat will have to be 0.6 milliwatts—still a trivial number.

<table>
<thead>
<tr>
<th>Quantity</th>
<th>Basis</th>
<th>Number</th>
<th>Gain, dB</th>
</tr>
</thead>
<tbody>
<tr>
<td>Noise</td>
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<td>8.00E-22</td>
<td>229</td>
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<tr>
<td>Noise figure</td>
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<td></td>
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<tr>
<td>Deg K</td>
<td>300</td>
<td></td>
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</tr>
<tr>
<td>Hertz</td>
<td>10</td>
<td></td>
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<tr>
<td>SN ratio</td>
<td>10</td>
<td></td>
<td></td>
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<td>Polarization loss</td>
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<td>RCVR losses</td>
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<td>Antenna beam, deg</td>
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<td>7.50</td>
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<tr>
<td>Power required</td>
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<table>
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<th>Wavelength, m</th>
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</thead>
<tbody>
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<td>0.12</td>
</tr>
<tr>
<td>Constellation diameter, km</td>
<td>50</td>
</tr>
<tr>
<td>Vertical distance, km</td>
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</tr>
<tr>
<td>Range, km</td>
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<tr>
<td>RCV antenna diameter</td>
<td>0.15</td>
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<tr>
<td>XMT antenna diameter</td>
<td>0.02</td>
</tr>
</tbody>
</table>

Figure 30
The link can be readily closed and is not demanding in any aspect.

The second is the determination of the communications link required to transmit the received composite signal from the central receiver to the ground. This is also a 10 Hz link for a one beam system and is straightforward. Two major options exist, one being to send the information to the illuminator which has a kilowatt transmitter and 20 meter antenna for re-transmission to the ground, and the other to directly transmit the signal to the ground. Figure 31 shows that the latter is simple, requires a 1.3 watt transmitter for each of the two polarizations, or 2.6 watts total and a 34 cm antenna at X band, and is a reasonable solution. If the system employed 10 beams for greater coverage rate the system bandwidth would be 100 Hz and the ground receiving antenna size could be increased to 1 meter with no increases in spacecraft transmitter power required. In any event both the communications links are trivial and could use off the shelf hardware.

<table>
<thead>
<tr>
<th>Quantity</th>
<th>Quantity</th>
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<th>Loss, dB</th>
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<tr>
<td>Margin des</td>
<td>Margin</td>
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<td>10.00</td>
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<tr>
<td>Sums</td>
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<td>286.63</td>
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</table>

| Required power       | Power required | dB watts | 1.28 |

Watts required: 1.3430406

Frequency, MHz: 10,000
Wavelength, m: 0.03
Constellation: 100
Vertical distance: 36000
Range, km: 36,000

| XMT antenna | 0.344 |
| RCVR antenna | 0.344 |

Figure 31

**g. Central receiver and Bus**

The central unit contains the principal receiver, and ancillary equipments typical of “bus” functions including communications, computation, processing, command, and support functions such as power, attitude control, thermal, etc. It is envisioned that the central unit will reside at the top of the tether and constitute its upper mass, and thus will require only very small propellant expenditures for stationkeeping.

The most important function of the receiver is to receive the signals from the thousands of picosats, add them coherently, and send the resultant composite signal to the ground for
processing and assimilation. The receiver can have just one channel for viewing one ground area at a time, or a multiple channel receiver with multiple independent feeds illuminating the antenna aperture (the picosat constellation) for viewing more than one area on the ground simultaneously. The communications requirements were discussed above and are trivial.

These multiple feeds can consist of a phased array, or of discrete feedhorns, each being functionally the same. Figure 32 illustrates a multiple horn-fed system which would generate a fan beam on the ground. This could be used in any scan mode or in a pushbroom mode, at will. An alternative would be to illuminate the picosat array with a 3x3 beam pattern, or even a 5x5 pattern, for greater coverage of smaller compact areas. The only requirement is for each horn to feed a separate receiver, and those separate signals to be transmitted to the ground. Another alternative is to use beam-forming networks much as is routinely done in communications satellites, to generate the multiple ground spots.

In actuality there will probably have to be two receivers per beam using cross polarized filters in order for the received signals to be sensitive to both polarizations of the illuminator signal as reflected from the earth, and adapt to Faraday rotation of the waves as they transit the ionosphere. Thus is easily accommodated by using circular polarization in the microwave plumbing, with cross-polarized detectors.

The bus per se is expected to be very conventional, and typical of most spacecraft buses being built today. In fact it is likely that several existing spacecraft buses could be used off the shelf. It will have solar array power, thermal control, attitude control, structure, and a communications system to receive commands from the ground and to transmit up to 20 signal channels to the ground, each channel being 10 Hz wide. The details of the bus are unimportant in this Phase I feasibility assessment as it requires neither new technology nor advanced techniques, and will be defined in detail in Phase II.
h. Picosats

The picosats are probably the most advanced technology development subsystem in this concept. They require to be very low weight and a high level of integration, and in order to be affordable in the numbers contemplated they must be mass produced to take advantage of learning curves. The intent is to avoid fabrication and hand assembly, as is the case in practically all current space systems, including the so-called microsats and even smaller Cubesats. A working definition of satellite by mass is that microsats weigh 10-100 kg, nanosats weigh 1-10 kg, and picosats weigh less than 1 kg. Femtosats have also been discussed as weighing 0.01-0.1 kg (10-100 grams), so the working definition for the satellites for this concept, which are estimated to weigh in the order of 50-200 grams, is picosats.

These spacecraft will have to be mass produced automatically in a production line containing a microcircuit furnace, in which most components will be grown, micromachined, or both. The aim is to attain as much hands-off production of the entire spacecraft as possible.

There is background for supposing that this is achievable in the near future, albeit with development and demonstration of the key technologies. A number of investigators, especially The Aerospace Corporation, have defined totally integrated nanosatellites
micromachined from silicon or glass substrates, and in fact have flown some in space to prove the concept. We will consider first silicon satellites.

The silicon satellite, presents a new paradigm for space system design, construction, testing, architecture, and deployment. Integrated spacecraft complete with some degree of attitude and orbit control can be designed for mass-production using batch-fabrication techniques. Integrated circuits for C&DH, communications, power conversion and control, on-board sensors, attitude sensors, and attitude control devices can be manufactured on thick silicon substrates that provide structure, radiation shielding, and thermal control. Some conventional components such as batteries and individual solar cells will still be required, but the total number of parts and assembly time will be drastically reduced. The spacecraft, as shown in Figure 33, is essentially a multi-ASIM module.

![Figure 33. A hypothetical silicon satellite.](image)

Silicon wafers are routinely produced with diameters up to 20 cm, which will increase to 30 cm within a decade. Low-volume (10 to 1000 wafers) production of custom circuits and MEMS uses wafers with diameters less than 15 cm. Simple ASIM-based integrated satellites will have dimensions of 10 to 20 cm while more complex configurations using additional non-silicon mechanical structure (i.e. truss beams, honeycomb panels and inflatable structures) will be much larger.

The benefits of batch-fabricated silicon satellites will be reduced parts count due to integrated electronics, sensors, and actuators on a single substrate; the ability to add redundancy and integrated diagnostics without significantly impacting production cost; decreased material variability and increased reliability due to rigid process control; rapid prototype production capability using electronic circuit, sensor, and MEMS design libraries.
with existing (and future) CAD/CAM tools and semiconductor foundries; elimination of labor-intensive assembly steps (welding, wiring cable harnesses, etc.); automated testing of systems and subsystems; and paper-less documentation of designs, fabrication processes, and testing.

Low cost per function is a direct result of the fabrication process; semiconductor batch fabrication techniques evolved within the constraints of consumer-driven market economics. Low mass and volume are simply byproducts of the fabrication process that can be exploited for space applications.

A major question is how much solar power can small satellites produce? Assuming that all of the available surface area is covered by solar cells, the extremes occur for a cubic satellite and for a satellite spread out into a ~10 micron-thick sheet; i.e. a flat all-solar-array satellite. Figure 34 shows the power extremes for 20% solar conversion efficiency, random pointing for a cubic satellite, and optimum pointing for a thin sheet satellite with average density equal to silicon. Picosatellites through microsatellites can produce power levels in the 1 to 100 Watt range while femtosatellites are in the microwatt to milliwatt range. Based on this it is reasonable to assume that about 1 watt might be available in the concept’s picosat (it might also grow to a nanosat). This area is very complex, and the design of the best shape of the picosat so as to balance the power generation, the heat rejection, and the attitude control limitations will be thoroughly investigated in Phase II.

The overall thermal balance of a spacecraft is determined by its orbit, its geometry, its surface properties, and its internal design. Small satellites with body-mounted solar arrays can have power-to-mass ratios equivalent to large satellites with deployable solar arrays, yet still be power-limited. High fractional surface coverage for solar cells is generally the rule, which results in spacecraft whose thermal balance is determined by solar cell absorptivity and emissivity. This does not allow much latitude in controlling spacecraft temperature.
ranges for small satellites. However the saving grace is that in GEO the day-night extreme swings in temperature are avoided except for the few eclipses that occur once a year, and earth albedo input is small compared to that of the Sun. The net result is that passive thermal control is possible for nearly spherical nanosatellites - microsatellites. When dimensions drop below 2 cm, the temperature extremes exceed typical electronics and battery limits in LEO but are likely to remain tractable in GEO.

Atmospheric drag considerations are predominant in LEO, but become unimportant in GEO. In order to show that that is the case consider that the main atmospheric effects are atomic oxygen erosion and orbital decay due to atmospheric drag. The drag force $F_D$ on a satellite is given by $F_D = \frac{1}{2}\rho V^2 SC_D$ where $\rho$ is the local atmospheric density, $V$ is the satellite velocity, $S$ is effective cross-sectional area of the spacecraft, and $CD$ is the satellite drag coefficient (CD~2 for most satellites). Orbit decay rates are often parameterized by introducing the ballistic coefficient $W$, defined as the satellite mass $M$ divided by the cross-sectional area $S$ times the drag coefficient $CD$. A spherical solid silicon satellite with a diameter of 10 cm would have a ballistic coefficient of ~150 kg/m$^2$. Figure 35 shows the maximum ballistic coefficient as a function of mass for a cubical satellite and a 10-micron-thick sheet satellite. Note how the ballistic coefficient is constant (and extremely small!) for the thin sheet satellite while it is a function of mass for a cubic satellite.

![Figure 35](image)

Figure 36 shows calculated orbital decay rates as a function of ballistic coefficient and altitude for worst-case solar-maximum conditions (4 to 6 years from now). Currently, we are in solar-minimum conditions, which produces decay rates about an order-of-magnitude lower at 300 km altitude and about 3 orders-of-magnitude lower at 700 km altitude. For satellites with ballistic coefficients of ~100 kg/m$^2$, note that at 700 km altitude the solar-maximum orbit decay rate is only ~10 km per year, while it will be exponentially lower with increasing altitude. At GEO altitudes drag is negligible in comparison with other forces.
acting on the picosats regardless of ballistic coefficient, and orbit maintenance will be but a tiny fraction of formation flying requirements. This discussion was included because atmospheric drag is such a very important consideration for spacecraft in LEO, and it is one the reasons that LEO was ruled out as an operating altitude for the concept.

Figure 36

A second technique for manufacturing the integrated picosats might be to use photostructurable glass or ceramics as the substrate for the picosats. The photostructurable glass/ceramics (PSGCs) are a subset of the more common glass/ceramics (GC). Glass/ceramic materials incorporate an in-situ nucleation process that results in the crystallization of the amorphous glass phase. This conversion process is nominally called devitrification. The initial material in the glass phase is melted and molded into the desired shape and then converted to the crystalline ceramic state. Because the resulting material is not 100% crystalline, but a composite of amorphous and crystalline phases, it is less brittle than crystalline ceramics. Glass ceramic materials are used in a wide range of conventional application.

The Aerospace Corporation has developed a maskless direct-write laser processing technique that permits true three-dimensional patterning and fabrication of microstructures using a particular PSGC material called FoturanTM. FoturanTM, manufactured by Schott Corporation, has zero porosity and is transparent for visible and near infrared wavelengths. The processing technique is not only useful for microfabrication but it also permits the local modification of the material strength by controlling the ratio of crystalline to amorphous phase. The developed technique also has several advantages over traditional lithography: the patterning process uses a pulsed UV laser direct write processing technique, which does not require masks; undercut structures can be fabricated by placing the laser focal volume beneath the intended structure during the patterning processes; multilevel embedded structures, such as tunnels/trenches can be patterned and fabricated; the chemical etch rate
of the exposed material is a function of the laser irradiation dose – by controlling the laser irradiance during patterning it is possible to pattern variegated aspect ratio structures on a common substrate and attain the release of all the structures with one timed-etching step; no protective masking is necessary during the etching process; and the penetration depth of the patterning process can be varied from several hundred microns to nearly a centimeter by changing the laser wavelength.

The technique relies on a pulsed UV laser to directly expose individual features in the PSGC material, and the exposure patterns are digitally rendered via CADCAM software, so that a processing and manufacturing methodology can be developed for all-digital mask-less processing. Minimum feature size attainable is related to the laser spot diameter and the size of the etchable metasilicate crystals, which from x-ray diffraction studies this is $\sim 1\mu m$. The technique is illustrated in Figure 37.

![Figure 37](image)

A cold gas microfabricated propulsion module was built by The Aerospace Corporation as a proof of concept demonstrator. The approach was to eliminate individual propellant tubes, propellant tanks, and filters by fabricating these structures as 3-dimensional voids in a single structural material, using in-house developed microfabrication techniques suitable to provide the requisite dimensional control. This propulsion module appears in Figures 38 and 39. It is a 10-cm diameter, 2-cm thick glass-ceramic cold gas propulsion module fabricated by Aerospace Corporation for DARPA. This laboratory prototype floats on an air table and contains 8 thrusters with 40-mN thrust levels to provide 2 axes of translational thrust plus rotation about the third orthogonal axis. Seven patterned glass/ceramic layers are bonded together to create the overall vehicle structure, the propellant tank with anti-sloshing vanes, a gas plenum, propellant delivery lines, and thruster nozzles. No bending of tubing, welding of tubes, or lathing of nozzles is required. The top layer is a conventional printed circuit board that contains the command and control electronics, a bi-directional wireless data link, a MEMS rate gyro, and a magnetometer.
Either of the two techniques—silicon or glass/ceramic substrates could be used to microfabricate the picosats in essentially automated furnace/factories. The details of how to set up the integrated micromanufacturing, as well as the detail characteristics of the picosats and their performance will be determined in Phase II.

While the above figures address a picosat designed and tested in a ground laboratory environment, let there be no question that picosats are feasible. Aerospace Corporation, among others, has built and flown in space picosats having some of the characteristics desired for the present concept. In particular Figure 40 shows two self-contained, tethered picosats that were developed by The Aerospace Corporation and flown for DARPA in 2000. The flight tests were fully successful, and serve as proof in principle that the low mass integrated picosats envisioned for this concept are feasible. The details of design, development, and testing, and the resultant weight and other characteristics of these picosats will be developed in Phase II.
1) Picosat propulsion

The propulsive systems envisioned for integration into the picosats for both attitude control and translation are Micromachined Field Effect Electrostatic propulsion (MMFEEP). These thrusters are the only known means to attain high specific thrust and high specific impulse simultaneously, both necessary features of propulsion devices for the picosats if they are not to dominate their weight and size. These MMFEEP thrusters are tiny and can be bonded to the outside of the picosat at the appropriate locations. These MMFEEP thrusters are now in the early stages of development by DARPA, and should be available in the time frame of all the other technologies of the concept. These are metal ion electric thrusters, illustrated in Figure 41.

![Micromachined Liquid Metal FEEP THRUSTER](image-url)
The MMFEEP thrusters have an order of magnitude lower operating voltage than conventional ion propulsion because the micromachining creates micron-sized gaps which produce very high field intensities with low voltages, as the Taylor cones that form under the field have tip radii of nanometers. They also feature continuously variable proportional thrust since they can be pulsed in nanoseconds, and a number of very attractive characteristics, listed in Figure 42.

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<th>ADVANTAGES OVER CONVENTIONAL ION PROPULSION</th>
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<tr>
<td>Conventional Ion</td>
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<tr>
<td>Micromachined Liquid metal FEEP</td>
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<tr>
<td>Specific impulse</td>
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<tr>
<td>Specific voltage needs*</td>
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* At Isp = 3,000 sec  ** At Isp = 1000 sec  *** Includes power processor

A major option exists to provide picosat attitude control, and perhaps even translation control, using solar pressure on orientable thin film surfaces, which would reduce or even eliminate the need for propellant-based thrusters. Such solar pressure devices would have infinite Isp. This option may be feasible because of the low ballistic coefficient of the picosats and the generally small perturbing forces experienced in GEO. Such solar pressure-operated propulsion subsystem will be explored in full in Phase II and traded against propellant-based MMFEEP thrusters for attitude control as well as translation control, and integrated into the picosat total design.

2) Picosat Antennas

There are two antennas in each picosat: the antenna used to receive the signal from the ground at 1.4 MHz and the antenna used to re-transmit the phase controlled and frequency-sifted signal toward the central receiver. They must be either omnidirectional if the picosat is to be able to avoid stabilization, or have gain and length vertically if the picosat is to be gravity-gradient stable. A third option exists in which the picosats are three-axis stabilized and then their antennas have more degrees of freedom.
The downward-facing receive antenna of a gravity-gradient stabilized picosat must have an acceptance angle or full beamwidth of nearly 18 degrees plus the stabilization errors so that it does not require reorientation as the beam of the array is scanned over the earth. For gravity gradient passive systems this tends to be no better than 5 degrees, and so the antenna angle should be no narrower than about 23 degrees. Such an antenna will have a gain of 30, or 15 dB. The upward facing antenna must be able to see the central receiver from any constellation location, also without reorientation. From the edge of a 100 km constellation the receiver tethered upward or downward 50 km subtends 45 degrees so the upward-facing antenna should subtend not less than 90 degrees. If the central receiver is tethered 100 km above the constellation array then the antenna can subtend 60 degrees and have a gain of 4 or 6 dB.

The implementation of the antennas can be a number of dipoles or patches spaced around the picosat for omnidirectional pattern, or an array can be used for higher gain. To keep the antenna weight down to the minimum so the antenna does not dominate the picosat weight a Yagi or similar end-fire array can be used, constructed entirely of thin gauge shape memory alloy wire which deploys to its intended shape after release of the picosat is attained. Such wire antennas are extremely lightweight and can be folded to very compact devices whose shape is compatible with the concept of a picosat. Such antennas are clearly feasible and very lightweight. A crude estimate of the antenna weight is as follows: The elements can be made from wire 100 microns thick (that is equivalent to a size 38 AWG, which is available commercially in a variety of alloys). There are 18 wires in the antenna, each 10 cm long (1/2 wavelength). The volume per wire is

\[ V = \left(1 \times 10^{-4}\right)^2 \times \frac{\pi}{4} \times 0.10m = 8 \times 10^{-19} \text{m}^3. \]

The weight per wire is \( W = V \rho \) where \( \rho = 5000 \text{kg/m}^3 \), so that

\[ W = 8 \times 10^{-19} \text{m}^3 \times 5000 \text{kg/m}^3 = 4 \times 10^{-6} \text{kg}. \]

Thus the weight of all 18 wires is

\[ W = 5 \times 10^{-6} \text{kg} \times 18 = 9 \times 10^{-5} \text{kg}. \]

Double this to account for the cross supports. Double it again for a second set of dipoles at right angles, for circular polarization. The result is \( 3.6 \times 10^{-4} \text{kg} \), or 0.36 grams. Thus two such antennas per picosat will weigh under 1 gram. A conceptual level design for a picosat with cross polarized Yagi 18 dB antennas appears in Figure 43.
3) **Picosat functional block diagram**
The picosats are complex spacecraft even though they are physically small. They contain a number of major electronic elements as well as complete “bus” or housekeeping functions, as shown in Figure 44. The implementation of all of these functions into a fully integrated picosat will be done with a goal of keeping the picosat weight under 100-200 grams. However should the weight grow the only implication is that the total weight of the space system will grow and therefore its cost will increase--it has no effect on basic feasibility. The question of the limits to such growth before the system becomes too expensive will be addressed in Phase II.
4) Picosat propulsion and system weight

The weight of the picosat can be crudely estimated as follows, recognizing that confident weight estimates will have to be deferred to Phase II. The picosat will need to have propulsion systems aboard for attitude control and for translational stationkeeping.

The picosat weight without propellants is estimated at 50-100 grams, including antennas, solar arrays, and all electronics. To be conservative the 100 gram weight is used at this time. Stationkeeping requirements are taken from previous studies of similar constellations at 50 m/s per year for overcoming luni-solar and solar pressure disturbances. This is conservative since no North-south stationkeeping is required.

For 10 year life this becomes 500 m/s total. Given MMFEEP propulsion with $I_{sp} = 3000$, this requires $W_i/W_f = e^{500 m/s \times 10 m/s^2} = 1.017$. Therefore the picosat initial mass must be increased by only 1.7 % or 1.7 grams for stationkeeping propellant for 10 years’ operation.

A more accurate method calculates the propellant needs for a picosat to stationkeep within its assigned box in the circulating array. We set a 100 meter “box” within which the picosat can drift freely while being continuously phase corrected. Since the average spacing of the picosats is over a kilometer a 100 m box is reasonable. Assume further that we take 1 hour to correct the 100 m error, and that it only needs to be done three times per day, a good estimate given previous studies of these orbits.

Then: $s = \frac{at^2}{2}$ and $a = \frac{2 \times 100 m}{(3600 \times 1)^2 \times s^2} = 1.5 \times 10^{-5} m/s^2$. The force required to produce this acceleration is $F = ma = 0.1 kg \times 1.5 \times 10^{-5} m/s^2 = 1.5 \times 10^{-6} N$.

Now if we accelerate for 1/2 hour and decelerate for 1/2 hour then $\Delta V = at = 1.5 \times 10^{-4} m/s^2 \times 3600 \times 1 = 5.4 \times 10^{-2} m/s$. Since this is required to be applied three times a day, the 10 year total is $\Delta V = 5.4 \times 10^{-2} \times 3 \times 365 \times 10 = 591 m/s$.

This requires $W_i/W_f = e^{3000 m/s \times 10 m/s^2} = 1.02$, or only 2 % of the initial mass, or 2 grams of propellant, which correlates reasonably with the 1.4 grams derived above. The thrust required translates into $1.5 \times 10^{-6} N \times 1 kg/N = 1.5 \times 10^{-6} kg$ of thrusters. The power to supply this thrust via MMFEEPs will be $P = 1.5 \times 10^{-6} N \times 1.4 \times 10^4 W/N = 0.021 W$. The solar arrays to provide this power will weigh $2.1 \times 10^4 kg$ at 100 W/kg. Thus the total 10 year weight penalty for translation propulsion will be just over 2 grams. This is a small price to pay for stationkeeping a 100 gram picosat for 10 years.

The picosat can now be roughly sized. The size and weight of each picosatellite is driven by the payload requirements. The payload requirements will be determined by
a system trade study that minimizes total constellation mass (the receiver satellite, tether, counter weight, plus the picosatellites). One initial fixed requirement is the receive frequency of ~1.4 GHz. This determines a characteristic size for the picosatellite antenna since efficient receive antennas are at least \( \lambda/4 \) long (5.4 cm for 1.4 GHz). With crossed dipole receive antennas, as shown in Figure 45, the antennas span ~11 cm while the picosatellite body can be a few cm in diameter or smaller. A disadvantage of this approach is that it is impossible to isolate another set of dipoles, needed for the transmitter, from the receiver dipoles. Both sets of antennas directly see each other. The solution is to separate receive and transmit frequencies, and utilize sharp cutoff filters in the receiver circuits. While this does indeed provide a great deal cross-talk prevention, second and third-order effects such as receiver desensitizing and intermodulation can still seriously affect receiver sensitivity. Physical separation of the antenna beam patterns would provide additional isolation to combat the higher-order effects, and active isolation techniques may also be helpful in attaining the required isolation.

![Figure 45. Picosatellite configuration for crossed dipole antennas.](image)

A patch antenna is a planar antenna with moderate gain (~6.5 dB) and narrow frequency bandwidth that can be mounted on opposite sides of the spacecraft body to provide good isolation between antenna beam patterns. Figure ZZ showed a pair of 260-gram mass picosats, fabricated by The Aerospace Corporation for DARPA, with rectangular 915 MHz patch antennas on their upper surfaces. These picosats have external dimensions of 10 x 7.5 x 2.5 cm and easily accommodate the ~6-cm rectangular patch antennas. At 1.4 GHz, the patch antennas would be ~4 cm and the satellite body could be 6 cm square (or larger).

It may be necessary to use higher gain antennas such as Uda-Yagi or helix antennas in order to benefit from the much higher front-back gain rations that such antennas
possess. These antennas require a reflector (preferably a solid ground plane or dense mesh to maximize isolation) with a diameter of ~/2. Thus the spacecraft body could be 11-cm in diameter, or less, and fit between the transmit and receive reflectors and serve double duty as picosat body and transmit-receive isolator.

An additional constraint on spacecraft size is the required power level and associated area of photovoltaic cells. The initial estimate of required average power level is between 0.1 and 1 watts. Further refinement requires system optimization studies which would be performed in a Phase II effort. To accommodate the cosine loss for random pointing of a flat array, plus expected eclipse periods at GEO, the solar array should produce a maximum power between 0.2 and 2 Watts. With dual-junction 20% efficient solar cells, this requires a square array with side length between 3 and 9 cm.

The picosat body will have a characteristic diameter between 5 and 10 cm based on the antenna and solar power requirements. The thickness will be between 1 and 3 cm to provide thermal inertia and to accommodate batteries, the avionics, and radiation shielding. Radiation shielding requirements provide a minimum thickness estimate. Typical commercial CMOS electronics can currently tolerate a total dose between 10 and 100 krads. Figure 46 shows the yearly radiation dose for equatorial orbits as a function of altitude for two aluminum areal shielding densities. For GEO altitude (35,786 km), an areal density near 3 g/cm² is required to provide a 10 krad total dose over a 10 year mission. This represents an aluminum thickness of about 1.1 cm. Since shielding will be required on top, bottom, and the sides, the picosatellite body will have a minimum thickness of 2.2 cm, assuming aluminum construction. A 3-cm thickness is probably the practical minimum to accommodate integrated circuits between the shielding.

Based on the above a minimum picosat mass, based on aluminum average density with dimensions of 3-cm thickness, and 5-cm characteristic diameter is 160 grams (0.16 kg). This is certainly in the size and weight range expected at the outset, and will lead to practical total system weights (and therefore costs), as 10,000 such picosats would weigh 1,600 kg total.
5) Picosat production and costs

The large number of units imply a true high volume production process. This is completely new to space, but common in commercial products. Producing this many identical units will be the first time that true mass production will be applied in space, and will therefore be the first time that the savings in costs that mass production allows will be beneficially applied.

The system cost would be totally outlandish using today’s Cost Estimating Relationships (CERs). If we estimate the dry mass to be produced as 5,000 kg (of 50,000 picosats at 100 grams each), if they were produced the way spacecraft are today the cost per unit would be 20,000 $/kg, or $ 2,000 per picosat for a total cost of $ 100 million. But if we apply mass production techniques the cost for the last unit, as well as the average cost will be very much lower. Curves of the so-called “learning factor” that applies in mass production are shown in Figures 47 and 48. Referring to these curves it is seen that if we use learning of 0.75, which is readily achieved in large electronics industrial production runs, the cost of the last unit will be 1% of the initial unit cost, or $20. In addition, the total cost of producing the 50,000 units will be in the order of $1,000 per dollar cost of the initial unit (which cost $2,000) and thus the total production run would cost only $2 M. This illustrates the potential power of large scale mass production, which is applicable to space systems such as the present concept which depend on large numbers of picosats—a factor of 50 reduction in hardware cost. This has major implications for the affordability of the resultant concept, and therefore the likelihood that it will be seriously considered rather than rejected outright for cost reasons.
6) **Picosat deployment and retrieval**

Each picosat requires injection into a slightly different orbit. In addition there must be means to dispose of picosats that are nearing the end of their useful life, as well as those which have already died, so that neither becomes orbital debris. Such debris creation is not only against international agreements but could pose a danger to the constellation itself, causing collisions and eventual chaos.
The deployment of the picosats will take place from a deployment mechanism that is located in the GEO orbit. The deployer could translate to the desired locations and eject each picosat one at a time. The $\Delta V$ required for picosat injection is very low, and in the worst case is 3.6 m/s. However the central deployer must provide the total $\Delta V$ for all the picosats and so it must provide a total of 3,600 m/s - 36,000 m/s, and it must provide it for a mass initially comprising all the picosats which could weigh 1,000 kg or more. Therefore this mode of deployment is ruled out as being too demanding on propulsion.

A better way is for the deployer to deliver the picosats to the desired injection points in-track in GEO, which can be done slowly and requires very small $\Delta V$, and then to release each picosat and have it inject itself into its desired orbit. For a 100 gram picosat the $\Delta V$ requires the expenditure of only 8 micrograms of propellant at an Isp of 3000, a negligible amount. Thus the deployer is envisioned as a depot that delivers picosats along the GEO orbit and returns to the center of the constellation when done. It must have internal packaging means to hold and release all the picosats, but only enough propellants to translate in track in GEO. For that matter a major option exists for each picosat to translate itself, from the depot located always at the center of the array to its desired injection point, a maneuver that requires even less propellant that the injection maneuver itself. This would free the depot from having to move from the constellation center, and is the preferred approach. This approach is illustrated in Figure 49.

![PICOSAT DEPLOYMENT, RETRIEVAL, AND SCAVENGING CONCEPT](image)

Figure 49

Once a picosat is nearing the end of its life, as determined by internal instrumentation of remotely from the central unit, it would translate itself back to the array center by first translating into GEO and then along the GEO arc to the depot. It would then fly into the depot under its own propulsion and be caught in sticky foam or other means in the depot. The depot would have doors to allow picosats to fly in and out. Rendezvous would not be a problem since picosat locations are known to 3.3 cm at all times, and open loop trajectories should be adequate for picosat retrieval.
The situation of retrieval of picosats that have unexpectedly died is more complex, but must be provided for. In that case a scavenging unit would be dispatched to meet a picosat and swallow it into a sticky foam inside packaging. This scavenger would need to hold only the fraction of picosats expected to die before the end of the system life, say 10% of the total number of picosats. Once the scavenger swallows a picosat it would stay at that picosat’s suborbit until called upon to swallow another one, in order to minimize propellants.

In this concept the depot would hold about 10,000 picosats and the scavenger would hold about 1,000 picosats. Thus the scavenger would need about 9 kg of propellant at \( I_{sp} \) of 3000 total using an assumed 300 kg initial weight. This implementation concept is clearly feasible, does not need rendezvous and docking sensors and command/control functions, and solves the deployment, retrieval, and scavenging needs of the system. Its characteristics will be defined in Phase II.

i. Tether
Tethers are elegant, very lightweight, tension-only multipurpose devices which are underutilized by designers principally because of a perceived risk or failure to understand their characteristics. However the truth is that tethers have already been successfully flown in space 17 times with only failure, and that was only a partial failure. Few advanced technologies can make that claim, and thus, with proper design, we can depend on space tethers in the system concept.

All tethers to date have flown in LEO rather than in GEO, where the concept is stationed, however there is a body of rigorous analysis of tethers in GEO performed on a previous study for the PI by Enrico Lorenzini and Mario Cosmo at the Smithsonian Astrophysical Observatory which will be applied to the present concept.

The elements of the system in GEO are acted upon by external perturbations, the most important of which are the solar radiation pressure and the third-body (Sun and Moon) gravity disturbances. The solar radiation pressure and the third-body gravity perturbations have very different effects on the tether stabilizer and the picosats. The solar radiation pressure produces a differential position error between the tether stabilizer and the picosats which depends on the difference of the inverse of the ballistic coefficients of these two bodies. This difference is large because the bodies probably have very different ballistic coefficients. Unlike the solar radiation pressure, the third-body gravity perturbations produce negligible differential position errors between the tether and the picosats because they are relatively close to one another and the gradients of the Solar and Lunar gravity in GEO are very small. In other words, the gravity perturbation moves all elements of the system together, and their principal effect will be perturbation requiring E-W and N-S stationkeeping, which are well understood from communications and other GEO satellites. In actuality the system could be allowed to drift in the North-South directions since there is
no need for it to precisely over the equator, and so East-West drift compensation is all that is needed.

The tether will bow under solar pressure because the tether’s ballistic coefficient is much smaller than that of the end masses. Thus bowing is calculated in Figure 50.

In order to keep this bowing from interfering with the picosats the tether will be anchored in GEO by a dedicated mass, in this case the picosat depot, which wants to be at the center anyway. Thus the configuration diagram is shown in Figure 51, showing that the bowing will not interfere with the picosats and need not be further addressed.
The anchor masses at the tether ends can be small. Since one of the masses will be the central receivers and bus, and the mass at the lower end will be a navigation reference unit surely weighing more than 30 kg, the system can be statically stable. The tether must be protected from micrometeoroid flux in GEO to avoid being cut frequently. To that end it can be constructed from multiple interconnected strands, whose effect is to guarantee a 10 year life with probability of being cut being less than 0.1 %, as illustrated in Figure 52. The details of the tether will be further defined in Phase II.
The dynamics of the tether have been explored for a similar configuration and are benign, with amplitude excursions generally being less than 0.2 degrees after 10 days. The amplitude of these librations will be controlled by thrusters on the central depot, which needs them anyway for stationkeeping, by a $\Delta V$ correction applied every two weeks or so. The specific maneuvers and their protocol will be defined further in Phase II.

j. Illuminator

The illuminator is not very exotic or difficult. Indeed it uses mostly the same technology as commercial and some simpler defense communications satellites. The initial estimate calls for a 20 meter reflector antenna, which is a scale-up of the Thuraya 12.5 meter reflector, though other antennas are available that are capable of the 20 meter size and would also suffice. The 1 kW transmitter is a simple development as traveling wave tubes need not be used, since only a very narrow bandwidth is needed. Thus a number of lower power transmitters could be paralleled, a technique employed in commercial spacecraft as well as some defense spacecraft, or a new higher power transmitter developed. In principle the transmitter is a CW transmitter and the narrower the bandwidth of the transmitter the less transmitter noise will be supplied to the illumination spot. In fact such transmitters already exist in other applications and off the shelf units may be available from defense applications. In the worst case an unmodified communications satellite could be used with its transponders paralleled, since modern satcoms generate 5-10 kW or RF power. It would be more expensive than if a separate CW transmitter were used, but would be completely off the shelf COTS.

It needs to be said that the illuminator power could be as low as 100 watts and the antenna size as low as 10 meters, depending on the system parameters chosen for the receiving constellation and the scan/dwell pattern and duration. Thus the illuminator is not a challenging development, nor does it require any new technology.
The illuminator should be located relatively close to the receiver so that the illuminated spot covers the area of regard of the receiver. However in this case relatively means roughly in the same hemisphere because there is great latitude in the relative position so long as there is common visibility. The illuminator must be placed between GEO “slots” assigned to communications satellites, but that is easy since those slots are 1500 km apart in the GEO orbit arc. Interference with communications satellites must be avoided, of course, but that should be fairly straightforward with a CW transmitter at 1.4 GHz. The illuminator antenna must have low far sidelobes, or must be fitted with a shield in the horizontal direction, both of which are straightforward. Interference with the receive constellation array and central receiver must also be avoided, and will be more difficult and critical if the receiver is not to be saturated by unintended transmissions from the illuminator. Again antenna design should provide a straightforward solution, which will be addressed in Phase II.

The power density placed on the ground by the illuminator is expected to be in the order of 3x10^{-9} \text{ W/m}^2, which is greater than ITU allowances for some modulated communications signals. The ITU regulations are unclear on unmodulated, or CW, signal levels, however. The subject will have to be researched for guidance regarding CW signals, as they are generally are filtered out by all communications and radar sets. Discussions may be required with experts on the ITU and WARC statutes, and with the FCC, regarding unmodulated illuminator allowed power density levels at discrete frequencies. These are planned activities in Phase II.

4. ATMOSPHERIC TEMPERATURE PROFILING

In addition to soil moisture and ocean salinity an attractive mission that was identified in Section 1 is the determination of the temperature profiles of columns of the atmosphere as an input to climate models. A passive radiometer to do just that was sized using an operating frequency of 60 GHz, with a completely analogous technique to that used for the passive radiometer operating at 1.4 GHz. The required number of picosats to be able to receive the thermal emissions from the earth passively at 60 GHz was determined and is shown in Figure 53.
The analysis was done for a GEO system, but since it appeared to require very large numbers of picosats it was also done for such a radiometer in LEO. Referring to the figure it is clear that the attainment of reasonable resolution, say about 1 km, would require well over a million picosats even if 20 dB antennas were used. In fact the number of picosats doesn’t drop to 100,000, the upper limit considered in this study, unless the ground spot size is increased to about 50 km, which is an unattractive choice due to smaller scale weather phenomena which would be missed. If the constellation were operated in LEO, however, the 1 km resolution spot or column width measured could be attained with only some 40,000 picosats, and a 2 km column could be observed with only 10,000 picosats.

Thus a passive atmospheric temperature profiler at 60 GHz could be implemented with a sparse array radiometer constellation, but only from LEO. The constellation size would shrink to about 1 km in LEO and while that is much smaller than the 100 km GEO constellation, the LEO environment is much more densely populated with spacecraft, and the collision danger is much greater. In addition, the practicality of operating such a system in LEO, as well as its sizing has to be examined more closely as the ΔV requirements grow substantially in LEO as compared to those in GEO due to larger disturbances as well as orbit mechanics. Nonetheless it appears that from a technical standpoint such a profiler could be feasible but only from LEO, implying less frequent revisits to a given area. While the preliminary technical conclusion is that such a profiler in LEO is not too attractive, its utility of such a capability will be assessed in Phase II before a final decision can be made whether or keep it or reject it.
5. EXPECTED PERFORMANCE AND UTILITY

The measure of how useful the concept will be is closely tied to meeting the expectations, or desires, as discussed and presented in Chapter 1. In assessing these we have to remember that there are no hard requirements for small spot size, continuous dwell, radiometers because no one thought that such a capability might be possible. Nonetheless the science community has identified a number of performance parameters that would meet their needs in characterizing the phenomenology, including area coverage rate and sensitivity attained, which are significantly more demanding than current or planned systems can provide.

The utility of the concept will thus be addressed from two different aspects: Area rate coverage for different sensitivities attained, and area rate covered for various spot sizes attained.

**Area rate covered and sensitivity attained**

One measure of the utility of the concept is the area that can be observed at a given sensitivity, as measured in noise equivalent temperature change detected (NEΔT). This area rate can be compared with that of conventional space systems when scanning equivalent areas in equivalent times. This measure was calculated and is shown in Figure 54. Conventional LEO systems do not have the capability to trade sensitivity for area coverage rate, as they simply scan a given footprint below the spacecraft, with the spacecraft furnishing the scan by virtue of orbital motion. These systems thus have a fixed area scan rate at the fixed system sensitivity.

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**Figure 54**

DAILY AREA RATE AS A FUNCTION OF SENSITIVITY: 10 BEAM RECEIVER

- United States
- California
- Chesapeake Bay
- San Francisco Bay
- Atlanta
- Georgia Tech

[Graph showing area rate covered as a function of sensitivity for different spot sizes.]
The illustration was calculated for a 10 beam central receiver which images 10 spots at a time, and uses the spot size as a parameter. This number of beams and receivers is conservative, considering that from Chapter 1 the SMOS, to be launched in 2007, will have 65 such receivers. The area rate is proportional to the number of beams and receivers used, and may be readily scaled. Results are not shown for a one beam system but they are one tenth of that shown for the ten beam system. Likewise a 100 beam system would have ten times the area rate than the 10 beam system shown.

The SMOS system will also have an area rate of $1 \times 10^9$ square kilometers/day with a sensitivity of 0.3 Degrees K, with a fixed resolution of some 35 kilometer “spot” size, and cover the globe every 3-5 days. The new concept cannot equal that area rate at that sensitivity, but it was not intended to do so. In fact it was intended to scan smaller areas and attain much better spot/resolution sizes, and to scan those area more frequently at a greater sensitivity, or both. Thus, for example, the new system could cover the USA every day with SMOS-like sensitivity, yet attain a ground resolution of 1,000 meters. But it not so fixed: it could scan an area the size of California daily with a 316 m spot size. It could also scan an area the size of the Chesapeake Bay three times daily with the same sensitivity, or a major city every 2 hours or so. neither current or planned systems in LEO can do that.

The system could, in addition, trade sensitivity for area rate, and attain the same sensitivity as Aquarius will in 2007-8 (0.05-0.1 deg K) for ocean salinity measurements, scanning the Chesapeake Bay every day at the same sensitivity but with a spot resolution of 300 meters. Or it could scan the Chesapeake every day with a 1,000 meter resolution, but with a sensitivity of 0.005 degrees K, or the San Francisco Bay with a sensitivity of 0.0005 degrees. Alternately it could scan the San Francisco Bay daily with a sensitivity of 0.05 Deg K and a resolution of 100 meters. It could also scan the San Francisco Bay every 2 hours or so at the same 0.05-0.1 deg K sensitivity as Aquarius and a resolution of about 800 meters. These performance numbers are nothing short of astounding, and will be extremely useful in hydrology as discussed in Chapter 1.

**Area covered and revisit times obtained**

A different measure of utility is the area coverage rate for different revisit times. This is illustrated in the graph of Figure 55.
In this case the curves indicate the benefits to be gained in frequent revisit times as well as resolution, with the same sensitivity as SMOS or Aquarius. Superimposed on the chart are revisit times most useful for soil moisture and/or ocean salinity measurements for science purposes, for which revisit times of a few days suffice; and for trafficability purposes for which revisit times of at most a few hours are important as discussed in Chapter 1. As before, shorter revisit times at the same SMOS 1-2 degree K sensitivity are possible, yet with much greater resolution. As examples an area the size of all of Alabama could be scanned every hour, for measuring the ability of roads soaked by rains to carry heavy traffic, with a ground resolution of 100 meters. Likewise all of California could be scanned every 2-3 hours with a resolution of 300 meters, and indeed about one quarter of the entire US could be scanned every few hours with a resolution of 1,000 meters. A different measure of merit would that all of Rhode Island could be scanned with a resolution of 100 meters every 15 minutes, or all of Alabama with a resolution of 300 meters every 15 minutes. The illustration was calculated for a 10 beam central receiver, and uses the spot size as a parameter.

As discussed above the number of beams could be as few as one or as many as 100, with area rate varying inversely proportional to the number of beams implemented. Trafficability and fine resolution, yet wide area coverage with frequent revisits such as these examples show have never been available, nor are any planned as no one thought they might be attainable. Its availability will be a major revolution to various science and application area users.
An additional degree of flexibility is the ability to point the system at various points on
the earth either in a repeating or in a random fashion, with complete flexibility as to how
many places can be revisited how often, within the total capabilities as shown by the
above figures. In addition, a large area can be randomly or pseudo-randomly revisited to
observe unpredictable events such as snow patches and ice packs in late summer, or
ground water content in areas of known snow content. In fact, the system could be
automatically commanded to randomly revisit a large number of critical spots for real
time observations, without the necessity for specific instructions to be laboriously coded
and sent the space system.

The earth sciences community has never had such a capability, and it could be
completely revolutionary. Its advantages, though intuitively obvious, will have to be
quantified in concert with the users and scientists themselves in Phase II.

**Atmospheric temperature profiler**

In addition to the soil moisture and ocean salinity missions, the concept was assessed for
obtaining atmospheric temperature profiles. As discussed above, it appears that from a
technical point, such a profiler could be feasible. However, the utility of such a system is
not clear and will also be assessed in Phase II.

6. **FEASIBILITY ASSESSMENT and ISSUES REQUIRING RESOLUTION**

The initial vision of the concept has held up well in the Phase I feasibility study, and the system
concept evolved differs from that going into the study principally in that it uses an active CW
illuminator to attain the desired small spot size and high sensitivity from GEO. Though there are
many additional design details that need to be defined, none affects the feasibility of the concept.
The final system concept as evolved in this study is illustrated in Figure 56.
The foregoing sections have described the system and all major subsystems of the concept. Though there are some technologies that must be advanced before the system can be considered for development, no issues or technology aspects have been uncovered which would render the concept infeasible—that is no showstoppers have been found.

Nonetheless two essentially non-technological issues have been identified that will need addressing in the deeper analyses planned for Phase II. The first is the maximum level of illuminating CW signal levels that the concept will be allowed to radiate toward the ground or toward other satellites to preclude interference. These signal levels are expected to be resolved when eventual consultations are undertaken with the ITU or ITU experts during Phase II. Even if the results of such consultations turn out to be more restrictive than expected, there are engineering approaches already identified that would mitigate such potentially interfering radiation which have to be but defined and designed, such as radiation shields and choosing frequencies for minimum interference. Therefore this issue is not expected to affect concept feasibility.

The second potential issue is the willingness of the science community to accept reflectivity-based, rather than emissivity-based soil moisture and ocean salinity observations. Though they are, in principle and in fact, equivalent there may be complexities in the observed phenomena that will require further addressing so that the science community becomes comfortable with
reflectivity-based radiometry observations. Mitigating this concern is the fact that active illumination from spacecraft in Scatterometers is already used in moisture observations, and will be enhanced with the launch of Aquarius, Hydros, and SMOS. However it must be said that the radar observations in some spacecraft will be used together with passive radiometer observations, which will not be available in the present advanced concept. While this issue is also not expected to affect feasibility, it would likely affect scheduled observations to be performed with the new concept so as to generate data that could maximally benefit from the passive radiometry data generated from SMOS, Hydros, or Aquarius.

No technological major issues have been found that could not be resolved in the analyses and designs performed to date, or that are not expected to yield to further in-depth analyses. The depth of these analyses and considerations have been necessarily limited due to the time and resource limitations of Phase I, but will be identified and receive the requisite attention in Phase II. These technologies and techniques are discussed in the chapter below.

7. TECHNOLOGIES REQUIRING DEVELOPMENT/Demonstration

As expected, a number of techniques identified for the concept will require definition and demonstration during Phase II. However, only a very few of them will also require technology development.

a. Techniques requiring definition and/or demonstration (but not new technologies)

The following techniques require detail exploration and definition in order to fully define the system concept and determine its performance. None of these techniques requires new technology, but some may benefit from simulation and/or laboratory setup test.

- Picosat relative position determination to 3.3 cm
  The actual means to determine the picosat positions to at most 3.3 cm in real time must be defined. Autonomous vs. externally obtained measurements and their implementation must be traded, and the best choice defined and its implementation and performance developed. This includes the metrology techniques as well as the navigation and reference techniques to be used.

- Accurate phase control of picosat signals using distributed reference
  The accuracy control requirements of phase shifts throughout the picosat constellation so that their signals arrive in phase at the central receiver must be analyzed. The accuracies required at every step, and the hardware tolerances implied, must be defined. This includes the distribution of coherent local oscillator signals to all picosats and the command means to exert control of the global phase shift pattern at the antenna for scanning and focusing.

- Picosat Doppler shift nulling
  The picosat orbital relative motions will generate Doppler shifts, which if not nulled will prohibit in-phase signal addition at the central receiver. A proprietary technique for
canceling all Doppler shifts has been conceived, and needs to be detailed and its performance limits defined.

• Mainlobe and sidelobe pattern of the sparse array with final parameters

The pattern of the sparse antenna array must be calculated, and its most relevant parameters—near-in sidelobe level, far sidelobe level, and suppression of grating lobes—must be quantified once the exact number of picosats and their proposed distribution in the constellation are set. Existing computer programs are adequate for this task.

• Control of tether dynamics, particularly out of plane librations

Though preliminary calculations indicate no problem, the actual tether dynamics must be determined once the actual tether parameters and the masses of the upper, middle, and lower anchors is set. Once the dynamics are derived their magnitude may or not be such as to require libration control. If control turns out to be desirable the parameters of the libration controller will have to be set for one of the several known effective control techniques.

• Simulation of picosat constellation dynamics, allowable picosat position errors, and determination of the $\Delta V$ required in the picosats

The constellation parameters must be set and chosen from Halo or Bedspring architectures based on trade study. Once defined the constellation and picosat drifts must be calculated and simulated, and the pattern degradation with time determined. Then the $\Delta V$ required for picosat position maintenance can be calculated, and the optimum strategy for configuration control can be set. Global constellation patterns as a function of time can then be determined.

• Autonomous determination of imminent picosat end-of-life

The actions to be taken for disposal of picosats are different for picosats that still exhibit control yet are nearing their end-of-life contrasted with those that have already died. It would be beneficial, though not absolutely necessary, to devise an autonomous means for sensing, or predicting, imminent picosat demise and the hardware implications defined.

• Autonomous scavenging of dead picosats by the scavenger unit

The means to gather and secure dead picosats within the scavenger unit must be defined and their performance determined. While it would appear straightforward, picosats must be restrained and held captive while the unit goes after another failed picosat, and there must be no escape from the unit to prevent creation of orbital debris.

• Techniques for deployment and storage of picosats in depot and scavenger

Techniques for passive and/or active means to store and release picosats in the depot unit must be defined, and their performance, weight, and volume determined. This will
determine launch vehicle sizing, among other aspects of the deploy sequence of operations.

- **Algorithms for picosat self-deployment and self-retrieval**
  It is envisioned that the picosats will deploy themselves from the depot unit, translate to their injection point in GEO, and inject themselves into their assigned orbit. Algorithms for effecting these operations, as well as the inverse operations of self-retrieval into the depot or scavenger units, must be defined.

- **Isolation of transmitter and receiver in the picosats**
  Means will have to implemented for isolation of the transmissions and receptions of the picosat microwave transmitters and receivers. This may include shields, filters, feed-forward, or combinations of all three. The isolation requirements and design requirements for the hardware to implement the isolation will have to be defined, and the performance of a preferred technique calculated or demonstrated.

- **Implementation of a narrow band RF receiver on a chip**
  Attainment of a microwave receiver with 10 Hz bandwidth, though in principle straightforward, will have to be designed and its implementation verified. Techniques and technologies known for this function include using multiple stage filters and multi-stage heterodyning techniques, all of which are state-of-the-art in conventional sized electronics, however their implementation on chip-sized receivers in integrated satellites weighing in the order of 0.1 kg must be demonstrated.

b. **Technologies requiring development and demonstration**
Only two new technologies have been identified which will have to be developed and demonstrated in order to make the concept a reality, and lower their risk to the point where NASA can seriously consider incorporation the concept into its long range roadmap for earth sciences.

- **Micromachined FEEP ion thrusters**
  The micromachined FEEP thrusters that are the heart of the propellant-based propulsion concept will have to be designed for the sizes and parameters required, once the rest of the subsystems of the picosats are defined. The fundamental elements of these MMFEEP thrusters will have to be fabricated and tested. Once their performance parameters are better understood the required thrusters can be confidently defined and incorporated into the total picosat design, which is the subject of the task below. This characterization will almost certainly require both analytical and laboratory experimentation, and the testing and characterization of a laboratory thruster element on the bench.

- **Integrated, solid state, mass produced picosats**
The picosats will have to be highly integrated designs suitable for mostly automated mass production in order for the costs of many thousands of such devices be reasonable. Techniques to manufacture them will have to be defined, traded and investigated, including using silicon as well photostructurable glass. All the subsystems will have to be designed and sized for the application, including all typical housekeeping functions. The payload elements of receivers, transmitter, phase shifter, LO receiver, and ancillary functions will need to be defined and designed so that all the bus and payload systems can be integrated into one picosat. The mass and characteristics of the picosats must then be determined, and fabrication techniques explored so as to be able to gain the appropriate cost reduction factors from their “learning curves”. This process will require analytical as well as laboratory exploration, and a series of demonstrations at staged levels in order to determine feasibility, reduce perceived risk, and convince doubters.

Each of these techniques and technologies will be fully addressed in Phase II. The enabled results of the Phase II study will thus be:

- Complete system and subsystem definition, sizing and characterization
- Complete system performance capabilities
- Definition and, where appropriate, limited demonstration of the new techniques and technologies, as appropriate
- A time-phased roadmap of technology development and demonstration, including laboratory, ground, and space demonstrations as appropriate

8. CONCLUSIONS/RECOMMENDATIONS

This Phase I study has clearly indicated that a revolutionary capability in earth sciences observation, particularly of the water cycle hydrology but also for other areas, could be available if the system concept is implemented. The concept of placing the earth sciences observatory in GEO so as to obtain frequent revisits or constant observation of selected small areas, yet with resolution spot sizes of 100-300 meters with a single space system was borne out. The extremely large antenna which is the heart of the system has been analyzed and found to be feasible with current technologies. The elements of the sparse antenna, namely the picosats, require new technologies, which have been defined and flagged for further attention and development in Phase II.

The study looked at the overall system architecture as well as at all major subsystems and defined all at a first level for feasibility assessment. All the major techniques and technologies necessary for their implementation were examined for feasibility and no showstoppers were found. Indeed there is only one area of technology which requires substantial development and demonstration and that is the picosat itself, with all its subsystems integrated including its thrusters. Though many of the major subsystems and techniques will require detailed analysis and in some areas simulation to fully understand the final system design and predict its performance, none other of the techniques identified requires technology development.
The utility of the system concept was shown to be very high, particularly in the earth hydrology science areas of soil moisture and ocean salinity, and the area of trafficability sensing of water-soaked roads. The capability of the system concept far exceeds that of current, programmed, and planned earth science spacecraft, and meets or far exceeds all requirements that the science community has developed for these applications. It will indeed be revolutionary. There are no plans to pursue such a capability even in long range roadmaps because no one had thought that such a capability could be feasible. Other areas of earth science and applications have been identified in which the concept could also show major advantages.

Thus while the feasibility of the concept has been established, many details of system configuration, trade studies to define best architectures and configurations, and definition of detailed subsystem designs have yet to be performed. These must, of necessity, be deferred to Phase II.

In view of this very positive Phase I assessment it is recommended that the system definition be undertaken in a Phase II, and a proposal to that effect will be submitted to NIAC.