

Architecture Study on Telemetry Coverage for Immediate Post-Separation Phase¹

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This paper presents the preliminary results of an architecture study that provides continuous telemetry coverage for NASA missions for immediate post-separation phase. This study is a collaboration effort between Jet Propulsion Laboratory (JPL), Goddard Space Flight Center (GSFC), and Applied Physics Laboratory (APL). After launch when the spacecraft separated from the upper stage, the spacecraft typically executes a number of mission-critical operations prior to the deployment of solar panels and the activation of the primary communication subsystem. JPL, GSFC, and APL have similar design principle statements that require continuous coverage of mission-critical telemetry during the immediate post-separation phase. To conform to these design principles, an architecture that consists of a separate spacecraft transmitter and a robust communication network capable of tracking the spacecraft signals is needed. The main results of this study are as follows:

- 1) At low altitude (< 10000 km) when most post-separation critical operations are executed, Earth-based network (e.g. Deep Space Network (DSN)) can only provide limited coverage, whereas space-based network (e.g. Space Network (SN)) can provide continuous coverage.**
- 2) Commercial-off-the-shelf SN compatible transmitters are available for small satellite applications.**

In this paper we present the detailed coverage analysis of Earth-based and Space-based networks. We identify the key functional and performance requirements of the architecture, and describe the proposed selection criteria of the spacecraft transmitter. We conclude the paper with a proposed forward plan.

I. Introduction

In prior missions, spacecraft that do not utilize Space Network (SN) usually lack telemetry coverage during the launch vehicle separation phase. As a result, no real-time data is transmitted to the Ground during mission-critical events like launch vehicle separation, power-up of major components and subsystems, deployment of solar panels, and propulsive maneuvers. In the event of a catastrophic mishap during the immediate separation phase, no flight telemetry data is available for anomaly investigations. Some recent mission mishaps during the launch vehicle separation phase, in which the exact causes of failure cannot be determined, are summarized as follows:

Comet Nucleus Tour (CONTOUR) – The CONTOUR is part of the NASA Discovery series of solar system exploration satellite. Launched on July 3, 2002, CONTOUR's primary objective was to perform close fly-bys of comets Encke and Schwassmann-Wachmann-3 nuclei with the possibility of a fly-by of a third known comet

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D'Arrest, or an as-yet-undiscovered comet. After booster ignition for solar orbit injection on August 15, 2002, contact with the probe could not be re-established. Ground-based telescopes later found three objects along the course of the satellite, leading to the speculation that it had been destroyed. Attempts to contact the probe were ended on December 20, 2002. The mission design did not provide for telemetry coverage during the solid rocket motor burn. As a result, the Mishap Investigation Board was unable to determine with certainty the exact cause of the failure.

Mars-96 – Mars-96 was the most ambitious planetary mission launched by Russian. The combined spacecraft include an orbiter with more than twenty instruments, two landers each with seven instruments, and two penetrators each with ten instruments. The spacecraft was launched by the Proton booster from the Baikonur Cosmodrome in Kazakhstan in the night of November 16, 1996. The launch was normal through the first burn of the fourth stage, putting the combined stage and spacecraft into a parking orbit. The second burn of the fourth stage was supposed to last for several minutes. For reasons still unknown, the second burn terminated after only 4 seconds. The spacecraft then separated from the fourth stage and started its own engine as programmed. However because of the pre-mature termination of the second burn which also left the spacecraft in an unexpected orientation, the final burn only resulted in an orbit of 70 km altitude. After three orbits, the spacecraft impacted near the coast of Chile. As no telemetry coverage was planned to cover the launch vehicle separation phase, the exact cause for the failure is unknown.

Launch vehicle separation mishaps and other mission critical event failures prompted NASA to establish guidelines to provide critical event telemetry coverage that is necessary for analysis and ability to incorporate information in follow-on projects. Both the Jet Propulsion Laboratory (JPL) and the Goddard Space Flight Center (GSFC) have similar flight system design, development, verification, and operation principles regarding spacecraft critical event support. The GSFC rule states that “Continuous telemetry coverage shall be maintained during all mission-critical events. Mission-critical events shall be defined to include separation from the launch vehicle; power-up of major components or subsystems; deployment of mechanisms and/or mission-critical appendages; and all planned propulsive maneuvers required to establish mission orbit and/or achieve safe attitude.”

This paper presents the preliminary results of an architecture study that provides continuous telemetry coverage for NASA missions for immediate post-separation phase. This study is a collaboration effort between Jet Propulsion Laboratory (JPL), Goddard Space Flight Center (GSFC), and Applied Physics Laboratory (APL).

The rest of the paper is organized as follows: Section II discusses the architecture study approach to evaluate the architecture options. Section III provides the coverage analyzes using Earth-based network (e.g. the Deep Space Network) as well as the space-based network (e.g. Space Network). Section IV discusses the preliminary flight system findings on various options from existing Commercial-Off-The-Shelf (COTS) transmitters and transponder. Section V presents a preliminary list of cost drivers for this architecture, and Section VI provides the concluding remarks and forward plan.

II. Concept of Operation, Key Requirements, and Figure-Of-Merits

This section describes the study approach to evaluate different architecture options. We first define a high-level concept of operation that envisions where, what, when, and how to provide telemetry coverage for immediate launch vehicle post-separation phase. Next we discuss the key functional and performance requirements that define the parameters that candidate architecture and design options must meet to be evaluated. We then provide a set of design options, and we describe a set of figure-of-merit's (FOMs) that captures the characteristics that can be used to measure the relative effectiveness of various options.

A. Concept of Operation

The Launch phase of a typical deep space mission begins when the spacecraft transfers to internal power on the launch pad and ends when the spacecraft is declared stable, healthy, is ready to accept commands, and the launch telemetry has been played back. In this paper we propose to equip the spacecraft with a low-power and low-mass transmitter that is activated immediately after the spacecraft is separated from the launch vehicle (or the upper stage), and to subscribe to Earth-based communication infrastructure for telemetry downlink.

Typically during the immediate post-separation phase the spacecraft performs a number of critical operations like powering-up of major components or subsystems; deployment of spacecraft mechanical structures and/or mission-critical appendages; and all planned propulsive maneuvers required to establish mission orbit and/or achieve safe attitude. During this time the transmitter sends back real-time low-rate critical event telemetry, and is expected to run on spacecraft internal power until the solar panels are deployed and the primary telecommunication subsystem is activated. To ensure continuous telemetry downlink, the spacecraft has to have a wide enough antenna

pattern to cover both nominal and off-nominal spacecraft altitude. The Earth-based network, on the other hand, is required to provide continuous communication coverage during the immediate post-separation phase.

B. Key Functional and Performance Requirements

The proposed communication architecture is intended to provide continuous telemetry coverage for NASA spacecraft during the launch vehicle separation phase, both for the nominal and off-nominal scenarios. High-level requirements that drive the development and implementation of this architecture are as follows:

1. Spacecraft to support a telemetry downlink rate of 2 kbps or higher in all spacecraft altitude.
 - Rationale: The spacecraft needs to provide a data rate of 2 kbps or higher to downlink its critical telemetry during the launch vehicle separation phase, whether it can maintain its altitude or not. This is to support emergency response and mishap investigation.
2. Earth-based network to provide continuous telemetry coverage for the launch vehicle separation phase for all spacecraft launch trajectories.
 - Rationale: The Earth-based network has to provide continuous communication telemetry coverage at any altitude anywhere on Earth during the launch vehicle separation phase.

C. Architecture Options

For Earth-based infrastructure, we investigate two options: ground-based antenna network (e.g. Deep Space Network (DSN) and Ground Network (GN)) and space-based antenna network (e.g. Space Network (SN)). We found that most critical operations after launch vehicle separations are executed at altitudes that are less than 1000 km. Based on this fact, we simulate the ground-based antenna network coverage using DSN and its extension, and compare it with the space-based antenna network coverage using SN. The detailed simulation results are discussed in Section 3, and it is obvious that the SN can provide far better telemetry coverage for spacecraft at altitudes less than 10000 km.

Based on the above findings on Earth-based infrastructure, we identify a number of Commercial-Off-The-Shelf (COTS) SN compatible transmitters as candidates for further evaluations:

- 1) Wallops's Low Cost Tracking and Data Relay Satellite System (TDRSS) Transmitter (LCT2)
- 2) L3's T710/T710A/T710B [2]
- 3) L3's T709 [3]
- 4) L3's T719 [4]
- 5) TES (Thales)'s EW C12
- 6) AeroAstro's S-band Radio System [5]
- 7) SDST Modification
- 8) L3's MST-765 [6]

The detailed comparison between the transmitter options is discussed in Section 4. We have not looked into spacecraft antenna options in this study, but we expect that this will not be dependent on the choice of transmitter³.

D. Figure-Of-Merits (FOMs)

We propose to use the following FOMs to evaluate the SN transmitter options:

- 1) **Mass:** The mass of the transmitter and its supporting structure.
- 2) **Volume:** The dimensions of the transmitter and its supporting structure.
- 3) **Input power:** The DC power at the input of the transmitter. Note that the spacecraft is running on battery during the launch vehicle separation phase before the solar panel deployment.
- 4) **Output power:** The output power of the transmitter that determines the telemetry rate.
- 5) **Integration and testing cost:** The cost of flight system accommodation and its associated testing cost. We assume transmitter uses standard interface to avionics.
- 6) **Development cost:** The cost to modify, upgrade, and space-qualify the COTS transmitter. Due to the short duration of the launch vehicle separation phase, we assume the use of Class C or D parts.

III. Network Coverage Analysis

In this section, we provide the coverage analysis of both the ground network and the space network for spacecraft in different altitudes. Our objective is to perform the simulations to understand the coverage performance

³ The SN transmitters have either one port or two ports. The ones have one port would need additional wiring and a diplexer, but we do not believe this would be a cost driver to the overall design.

of the two networks and make appropriate recommendations to ensure that the link supports within the network are robust and efficient.

The primary earth ground stations consist of antennas at Goldstone, Canberra, and Madrid, which we refer to them as *DSN3*. Although the individual sites can operate at lower or higher horizon masks depending on the bandwidths, in our simulations, we assume 10 degrees for all ground stations. *DSN3* can provide excellent coverage as one of the three sites can link to any deep space or interplanetary mission. Spacecrafts closer to earth may not be in view with any *DSN3* site. Thus the aerial coverage region above a ground station grows as a function of altitudes. Figure 1 shows the areas of coverage by the different sites of the ground network. Each color-coded boundary indicates the area covered by a particular station when the spacecraft must be at least of a certain height. For example, the red oval around Goldstone shows that, the elevation angle of any spacecraft 1000 km or above within that oval is higher than 10 degrees. The coverage altitudes are as follows: 1,000 km (red), 5000 km (yellow), 10000 km (cyan), 20000 km (green), 35000 km (blue), 40000 km (magenta), and 400000 km (black). For perspective clarification, we want to mention that the average altitudes of geostationary orbits and the earth-moon distance are 35000 km and 400000 km, respectively. The imperfect separation between Canberra and Madrid ($>120^\circ$) has resulted the several no-coverage zones between *DSN3* and the spacecraft depending on the altitude. Also both Goldstone and Madrid are in the northern hemisphere, which exclude any communication between *DSN* and the spacecraft over South America

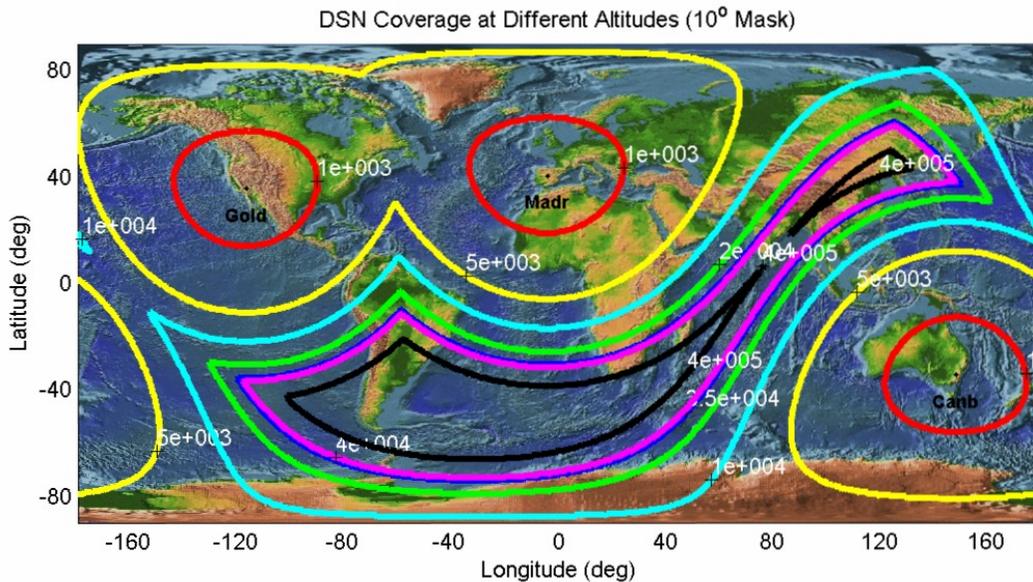


Figure 1. Coverage by Goldstone, Madrid, and Canberra at different altitudes 1,000 km (red), 5000 km (yellow), 10000 km (cyan), 20000 km (green), 35000 km (blue), 40000 km (magenta), 400000 km (black)

In a previous study, Lee showed that continuous coverage of the entire moon with at least 20-degree elevation mask would require at least six earth ground stations [1]. The resulting six stations include the current *DSN* sites along with Hartebeesthoek, South Africa, Santiago, Chile, and Usuda, Japan. These well separated stations were optimized among a number of existing facilities and capabilities in the world. The conservative 20-degree horizon mask was imposed due to the proposed future lunar Ka-band links whose atmospheric losses can be large at low elevation angles. However, as mentioned earlier due to our proposed S-band links, the horizon masks in our studies are assumed to be 10 degrees. It was also pointed out in [1] that the road map for extending the stations in an efficient order for supporting the future lunar exploration. Namely, the ground network coverage would improve optimally if the extension follows the chronological sequence of Hartebeesthoek, Santiago, and then Usuda. Such optimality can be justified as follows. Due to the Earth's tilt and the Moon's inclination, the Moon spends most of the time within the $\pm 28^\circ$ latitudes. As a result, the no-lunar communication zones by *DSN3*, from the largest to the smallest, are over the South East Africa, South America, and East Asia skies. We denote the different ground networks as *DSN4*, *DSN5*, and *DSN6*. Particularly, *DSN4* consists of *DSN3* and Hartebeesthoek, *DSN5* is *DSN4* plus Santiago, and *DSN6* is *DSN5* with Usuda. The ground network coverage for the optimal six stations is displayed in Fig. 2. As expected, large percentages of the surface area 5000k or more above the earth are covered

by these stations. Indeed, they cover 99% of the surface 10000 km or higher than the earth. Figure 3 shows the percentages of sky covered by DSN3-DSN6 at different altitudes.

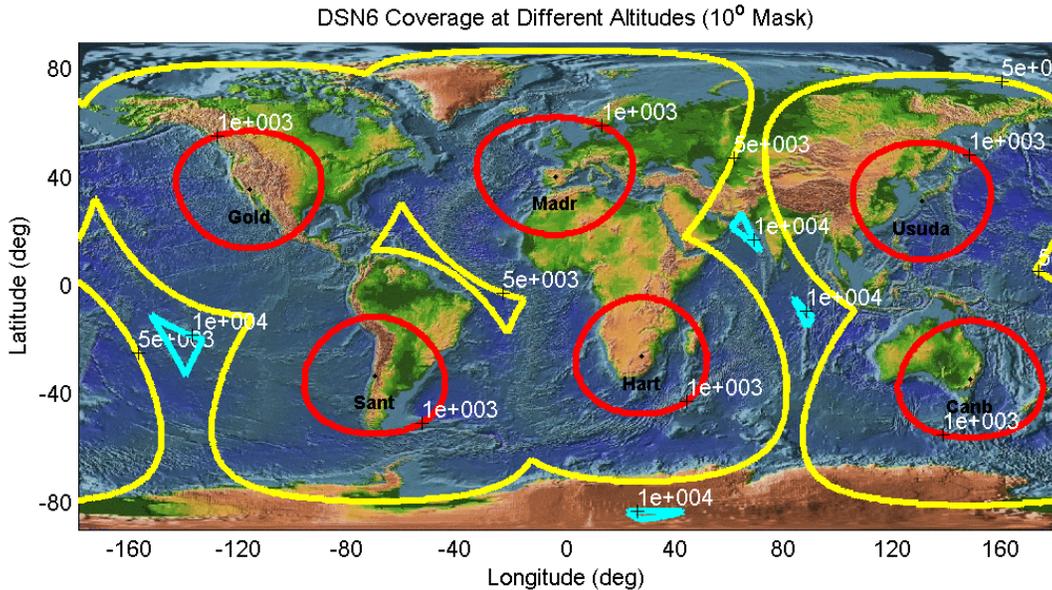


Figure 2. Coverage by Goldstone, Madrid, Canberra, Hartebeesthoek, Santiago, and Usuda at different altitudes 1,000 km (red), 5000 km (yellow), 10000 km (cyan)

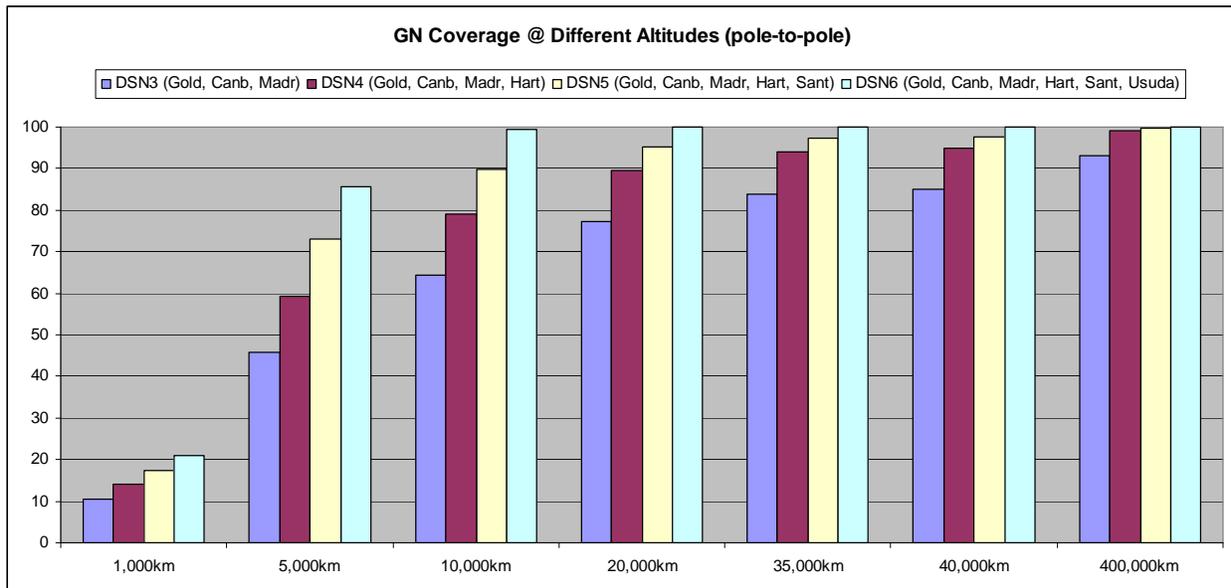


Figure 3. Pole-to-pole coverage percentages of different networks of ground stations at various altitudes (10° mask)

Since our proposed radio is designed for post separation phase, it is meaningful to investigate the launch profiles of actual missions. We obtain information on eleven launches, which consist of Earth missions as well as deep space missions. They include the Mars Phoenix Lander, Dawn (Asteroid Belt Mission), MUSES-C (Asteroid Sample Return Mission), Selene (a Lunar Satellite), Mars Reconnaissance Orbiter, FSU, Mars Exploration Rover-A, Genesis (the Solar Wind Sample Return Mission), and Contour (the Comet Nucleus Tour Mission). The latitudes of these missions as a function of post-launch time in minutes are displayed in Fig. 4. All of these missions remain within the ± 40 -degree band around the equator. To assess the coverage of the ground network coverage more

closely to the actual launches, we focus on the coverage to the $\pm 45^\circ$ -band around the equator and the aerial coverage percentages of this band for various altitudes are shown in Fig. 5.

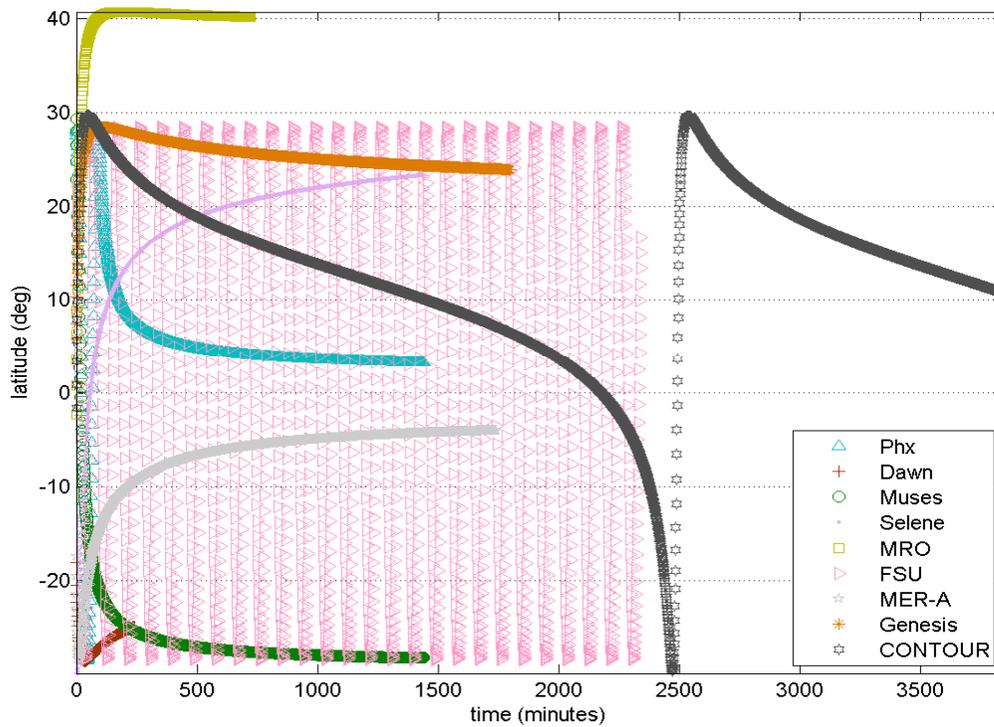


Figure 4. Latitudes of eleven launch profiles

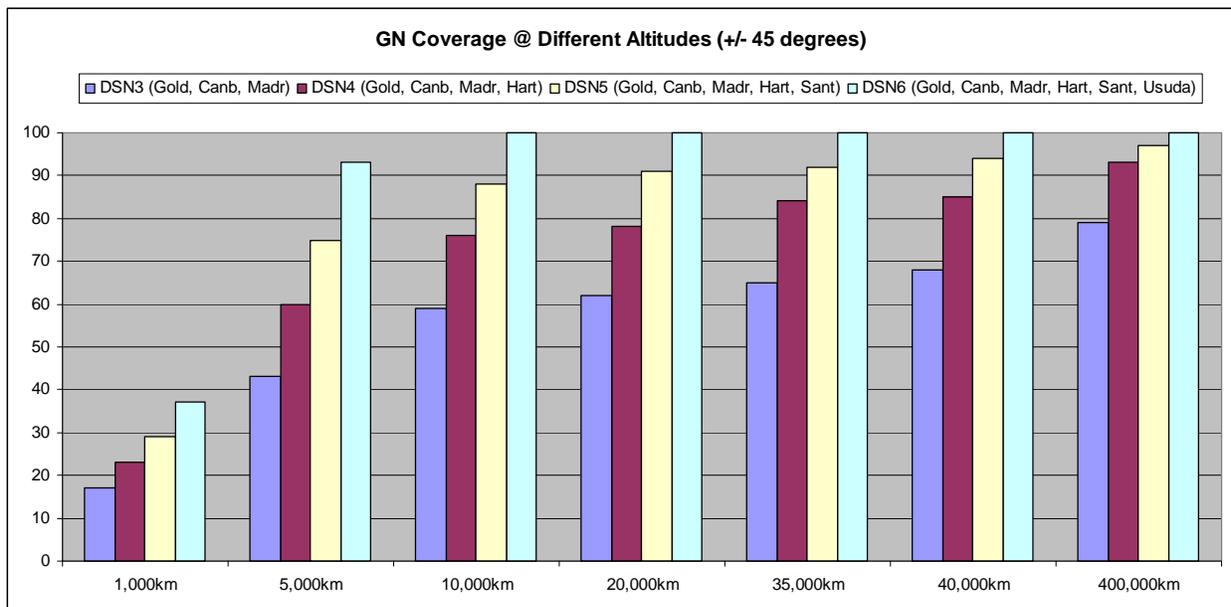


Figure 5. Percentages of the $\pm 45^\circ$ -band around the equator covered by different networks of ground stations at various altitudes (10° mask)

The space segment of the Space Network (SN) is the Tracking & Data Relay Satellites System (TDRSS) which consists of 9 satellites and provides services to both NASA and non-NASA missions⁴. These satellites are in geosynchronous orbit with allocated inclinations and longitudes. There have been two generations of TDRSS with identical payloads in each generation. While the field of view of the first generation TDRSS (F1-F7) is a rectangle of dimensions $\pm 22^\circ$ E-W and $\pm 28.0^\circ$ N-S, the field of view of the second generation (F8-F10) is an extended ellipse $\pm 76.8^\circ$ E-W and $\pm 30.5^\circ$ N-S. The left picture of Fig. 6 shows the areas of coverage of the first-generation-TDRS F-3 at the 10000-km altitude. The black dot on the orthographic map represents the footprint of TDRS and the blue areas show the zones of exclusion. The biggest zone of exclusion lies on the opposite side of F-3 and is due to the occultation of Earth. The smaller exclusive zones on the side of F-3 are caused by the rectangular field-of-view cone cutting through a sphere. We break the area of coverage into two types. Points in the sky that are in view and are on the same hemisphere with the TDRS are called the same-hemisphere in view and are drawn in green. The burgundy region denotes the points that are in view with TDRS, but lie on the opposite hemisphere. We deliberately distinguish the types of in-view coverage to address the communication performance. Notice that the distance between a spacecraft in the higher altitudes and a TDRS varies quite significantly when it is in view. For example, a spacecraft at 30,000-km altitude can communicate with a TDRS at around 6,000 km to 55,000 km on the same hemisphere, whereas on the opposite hemisphere the distance could be at 80,000 km, rendering more required transmitting power on the spacecraft. The picture on the right of Fig. 6 is the area of coverage of the second-generation-TDRSS F-9 at 20,000-km altitude. The pattern is somewhat different due to the broader field of view, TDRS's inclination, and altitude of coverage.

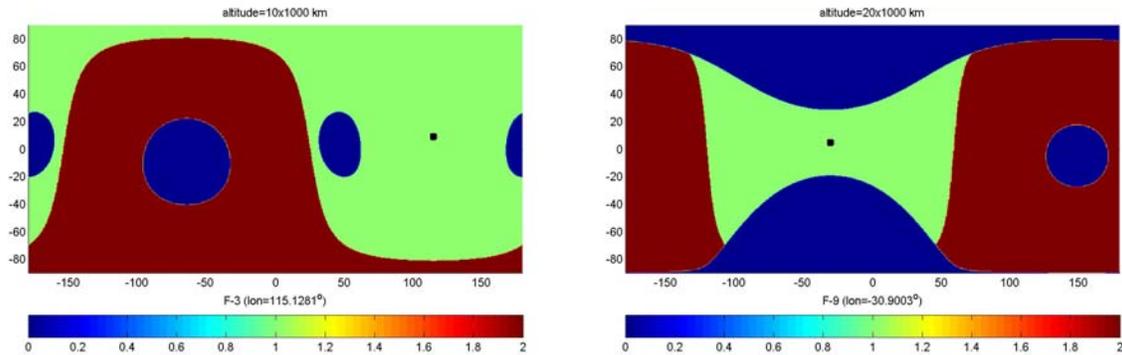


Figure 6. Aerial coverage of the 10000-km altitude by the 1st-generation TDRS F-3 (left) and the 20000-km altitude by the 2nd-generation TDRS F-9 (right). Regions in black, green, burgundy, and blue represent position of TDRS, the same-hemispherical in view, the opposite-hemispherical in view, and the zones of exclusion, respectively.

The next natural question is what would be the number of TDRS that can efficiently cover the post-launch phases. Two sets of SN TDRSS are considered. One consists of F3 and F9 and the other consists of F3, F8, and F9. We denote them as SN2 and SN3, respectively. The pictures in Fig. 7 show the orthographic map of coverage areas SN2 at the 10000-km altitude (left) and SN3 at 20000-km altitude (right).

⁴ TDRS 7 and 9 have been committed to Department of Defense (DOD) and are generally not available for normal operation support. NASA is procuring 2 additional second generation satellites. Nominal dates for operations are 2012 and 2013. There is an option for 2 additional satellites.

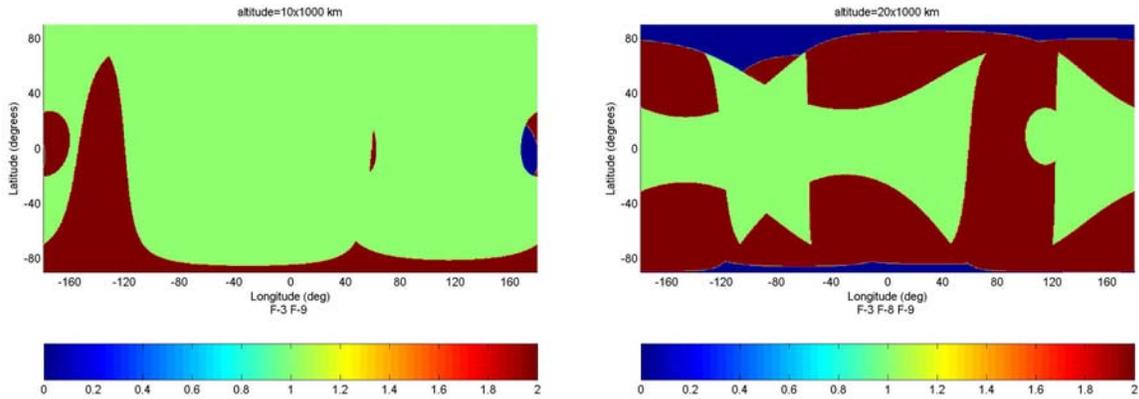


Figure 7. Aerial coverages of the 10000-km altitude by the SN of F-3 & F-9 (left) and the 20000-km altitude by the SN of F-3 F-8 and F-9 (right). Regions in black, green, burgundy, and blue represent position of TDRS, the same hemispherical in view, the opposite hemispherical in view, and the zones of exclusion, respectively.

As shown in Fig. 7, the two-TDRS SN2 can cover almost all areas at the 10,000-km altitude and most of the coverage is on the same hemisphere. The three-TDRS SN3 can cover most of the 20,000-km altitude areas; however, a vast area of coverage is on the opposite hemisphere and some polar regions are not covered. Figure 8 summarizes the coverage percentages of both SN2 and SN3 at different altitudes and also shows the percentages for the same and opposite hemisphere coverages. We can see that, for lower altitudes (<10,000 km), both SN2 and SN3 can almost completely cover all areas and achieve so with high percentages of same hemisphere coverage. And as expected, larger zones of exclusion by SN as the spacecraft climbs to higher altitudes.

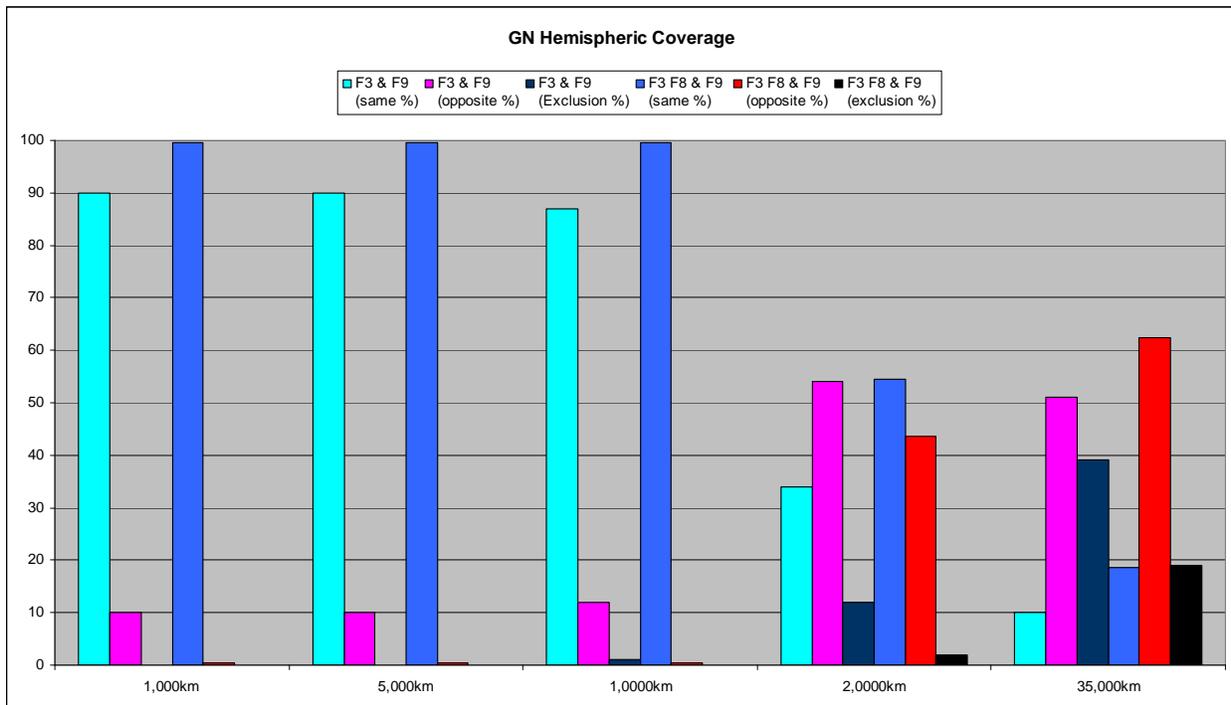


Figure 8. Summary of aerial coverages of two Space Network F-3 & F-9 and F-3 F-8 and F-9 at different altitudes

Let us now address the efficiency for the post separation coverage and our recommendations. Our results, summarized in Fig. 9, indicate that the 10,000-km altitude is an optimal break point between the space network and

the ground network. The space segment should be SN3 (F-3, F-8, and F-9) and the ground segment should be DSN6 (Canberra, Goldstone, Madrid, Hartebeesthoek, Santiago, and Usuda). SN2 (F-3 and F-9) is not recommended because it cannot completely cover all areas in the 10,000-km altitude. Particularly 99% of the area in the 10,000-km altitude is covered. On the other hand, SN3 (F-3, F-8, and F-9) covers 100% of the 10,000-km altitude areas and 99.5% of the coverage are on the same hemisphere. SN can be recommended for higher altitudes; however, it would require the support of many more TDRS. Using SN2 or SN3 to support spacecrafts in altitudes higher than 10,000 km will result in larger zones of exclusion and longer communicating distance due to opposite hemisphere coverage. When the spacecraft's altitude is 10,000 km or above, only DSN6 can provide 100% coverage and thus full and continuous coverage is possible when transiting from the SN3 to the DSN6.

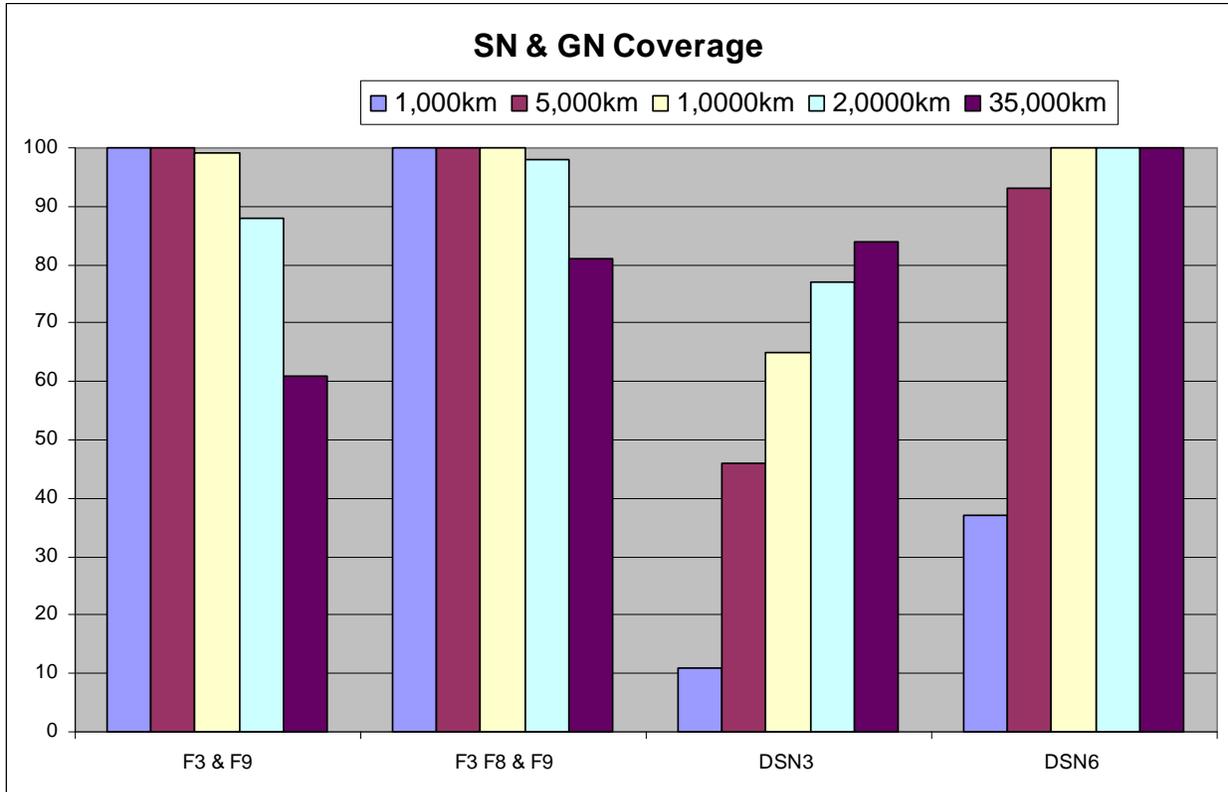


Figure 9. Summary of aerial coverages of the Space Networks and the Ground Networks for different altitudes

IV. Preliminary Flight System Findings

At the time of the study⁵, a number of low-mass and low-cost SN compatible transmitters are available in the market for micro-satellite applications. We reviewed the transmitter specifications⁶, discussed with the vendors, and summarized the relevant information of the candidate transmitters in Table 1 (partially populated).

⁵ This study starts circa March 2007 and ends circa September 2007.

⁶ When available.

SN Compatible Options	LCT2	T710/T710A/T710B	T709	T719	EW C12 0000
Manufacturer	Wallops	L3	L3	L3	TES (Thales)
Transmitter Mass (Kg)	< 0.9 Kg	< 2.0 Kg	< 4.7 kg	< 2.3 Kg	< 0.5 Kg
Transceiver Mass (Kg)					< 1.0 Kg
Transmitter Dimensions (cm)	10.16 x 12.70 x 3.81	13.34 x 17.78 x 6.60	17.78 x 17.78 x 18.42	13.97 x 19.18 x 10.16 (from phone)	12.7 x 16.00 x 2.03
# of antenna ports			2	2	1?
Spacecraft Bus IF	RS232		RS422		RS422
Approximate Lead time		18 - 24 months.			
Parts and Grades		NASA B+			
Testing and Certification Process					Space Qualified
Approximate Cost		Depends on acceptance test plan and part selection requirements.			Low Cost
Previous Users	ELV Sounding Rocket	T710-ATLAS LV T710B CloudSat spacecraft T710A ATLAS-V LV	Marshall ordered for CLV - Ares-1 Upper Stage S-Band Txr	NPP satellites (NPOESS Preparatory Project Spacecraft).	Deep Impact, Rosetta
Other Notes	No radiation harden or tolerate.	Rugged Version of T719, no digital processing slide for spread spectrum.	Very similar to T710, but with higher power output.	Digital dual mode	
Tx/Rv Mode	Tx only. DG2	Tx only. DG2	Can support TDRSS MA when user provides PN spreading code.	DG1: mode2, DG1:mode3, DG2	QPSK, Rate 1/2 coding
Ocillator		TCXO reference	TCXO reference		OCXO
Frequency	2200 - 2300	2200 - 2300	2200-2300	2200-2300	2200-2290
RF Power Output	10W (20W in development)	3W min. (total), 1.5W min (each)	26 W	7 W	2 W (5W option Available)
DC Power Input	< 69W	< 35W	145 W	37 W	10 W, 2.5 W standby

Table 1

SN Compatible Options, GN Mode Only	AeroAstro	SDST Modifications	MST-765		
Manufacturer	AeroAstro	General Dynamics	L3		
Transmitter Mass (Kg)	0.6 Kg	0.72 Kg increase	0.80 Kg		
Transceiver Mass (Kg)	0.9 Kg	0.72 Kg increase	1.60 Kg		
Transmitter Dimensions (cm)	8.89 x 5.08 x 2.79	52.26 cm ² increase in footprint	15.24 x 12.7 x 3.81		
# of antenna ports				1	1
Spacecraft Bus IF	RS422	Same as SDS1	RS-422		
Approximate Lead time					
Parts and Grades		Same as SDST			
Testing and Certification Process					
Approximate Cost					
Previous Users	MOST Mission, Canada	Heritage from SDST	ROCSAT III		
Other Notes	10 kRads (Si)	Same as SDST	100 kRads		
Tx/Rv Mode	Tx only. PCM/PSK/PM PCM/PM (DG2?)	Tx only. PCM/PSK/PM PCM/PM (DG2?)			
Ocillator					
Frequency	2200 - 2300	2200 - 2300	2200-2300		
RF Power Output	5W	5W	5W (6.5W available)		
DC Power Input	~32W	26.3W	35W		

Table 1 (Continued)

V. Preliminary List of Cost Drivers

To implement the proposed architecture of integrating a separate SN transmitter onto the spacecraft and subscribing the SN to track the spacecraft during the immediate launch vehicle separation phase, we identify the following items to be the major cost drivers from an end-to-end life cycle point of view:

- 1) **Telecom hardware** – this includes cost of hardware items like transmitter, antenna, coaxial cables, switches, etc.
- 2) **Flight system accommodation** – this includes the additional development cost to provide power, thermal, command-and-data-handling (C&DH), cable routing, switching, flight software, and structure support to integrate the SN transmitter onto the spacecraft. This also includes the cost of developing additional logics for spacecraft fault-tolerant. On the spacecraft system engineering and review process point of view, this architecture requires additional effort (thus cost) to identify decision points in mission planning life cycle as on whether the post-separation transmitter is needed or not.
- 3) **Assembly, Test, Launch, and Operation (ATLO) and test** – this includes the cost of additional Electromagnetic Compatibility (EMC) tests and Electromagnetic Interference (EMI) tests, and the certification of the SN transmitter.
- 4) **Mission Operation System (MOS) and Ground Data System (GDS)** – this include the additional cost on defining and developing additional telemetry channels on the flight system side, and the development and operation cost of interfacing the SN with project’s MOS/GDS .
- 5) **Network** – this includes the development and operation cost to coordinate SN support for launch support.

VI. Results Summary and Forward Plan

This paper summarizes the results of the preliminary architecture study on telemetry coverage during the immediate post-separation phase. We found that at low altitude (< 10000 km) when most post-separation critical operations are executed, Earth-based networks can only provide limited coverage, whereas space-based network likes SN can provide continuous coverage almost anywhere on Earth. Also COTS SN compatible transmitters are available for small satellite applications, and we identify a number of promising candidates.

Based on the aforementioned preliminary results, we propose the following forward plan to conduct an in-depth architecture study:

- 1) As this study was concluded in September 2007, new and improved SN transmitters have been introduced into the market. We propose to re-assess the transmitter options and compile an updated list to support further down-selection processes.
- 2) Investigate detailed nominal and off-nominal post-separation scenarios. This include
 - a. Assessing range, range rate, and attitude profiles for different scenarios to derive distance, Doppler, and Doppler rate statistics.
 - b. Assessing detailed spacecraft activities from separation till solar panel deployment and primary telecom subsystem activation to estimate critical telemetry generation rate.
- 3) Based on the above analysis, derive detailed coverage and communication performance requirements.
- 4) Assess upgrade required for each option to meet the requirements.
- 5) Evaluation each option against the FOMs.
- 6) Provide recommendation on transmitter option and other RF hardware.
- 7) Investigate other enhancements and new architecture capabilities like uplink commanding, position tracking, and SN ground station upgrades.

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Reference herein to any specific commercial product, process, or service by trade name, trademark, manufacturer, or otherwise, does not constitute or imply its endorsement by the United States Government or the Jet Propulsion Laboratory, California Institute of Technology.

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