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Very-Small-Satellite Design for Distributed Space Missions

David J. Barnhart,* Tanya Vladimirova,† and Martin N. Sweeting‡
University of Surrey, Guildford, GU2 7XH England, United Kingdom

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A new class of remote sensing and scientific distributed space missions is emerging that requires hundreds to thousands of satellites for simultaneous multipoint sensing. These missions, stymied by the lack of a low-cost mass-producible sensor node, can become reality by merging the concepts of distributed satellite systems and terrestrial wireless sensor networks. A novel, subkilogram, very-small-satellite design can potentially enable these missions. Existing technologies are first investigated, such as standardized picosatellites and microengineered aerospace systems. Two new alternatives are then presented that focus on a low-cost approach by leveraging existing commercial mass-production capabilities: a satellite on a chip (SpaceChip) and a satellite on a printed circuit board. Preliminary results indicate that SpaceChip and a satellite on a printed circuit board offer an order of magnitude of cost savings over existing approaches.

Nomenclature

A_{sa}	=	required solar array area, m^2
a	=	semimajor axis, m
C	=	capacitance, F
C_r	=	battery capacity required, A · hr
c	=	speed of light, 3×10^8 m · s^{-1}
F_p	=	flat-plate view factor
f	=	frequency, Hz
G_r	=	receiver gain
G_s	=	solar flux, 1418 to 1326 W · m^{-2}
G_t	=	transmitter gain
h	=	altitude, m
I_d	=	inherent degradation
K_a	=	spherical view factor
k	=	Boltzmann's constant, 1.381×10^{-23} J · K^{-1}
L_s	=	free-space loss
n	=	transmission efficiency between battery and load
P	=	period, s
P_{BOL}	=	beginning-of-life solar array power output, W · m^{-2}
P_e	=	power required in eclipse, W
P_s	=	power required in sun, W
P_{sa}	=	power required from solar array, W
P_t	=	transmitter power, W
q_I	=	Earth's infrared flux, 237 ± 21 W · m^{-2}
R	=	data rate, bits per second
$R_\oplus$	=	Earth's radius, 6,378,136 m
S	=	range, m
T_e	=	time in eclipse, s
T_s	=	time in sun, s
T_{sys}	=	system noise, dB · K
v	=	voltage, V
w	=	energy, J
X_e	=	power-transfer efficiency in eclipse
X_s	=	power-transfer efficiency in sun
α_{Si}	=	absorptivity of silicon, 0.48
ε_{Si}	=	emissivity of silicon, 0.46
η	=	solar cell efficiency
θ	=	incidence angle, deg
λ	=	wavelength, m
$\mu_\oplus$	=	Earth's gravitational parameter, 3.986×10^5 km^3 · s^{-2}
ρ	=	angular radius, deg
σ	=	Stefan–Boltzmann constant, 5.67×10^{-8} W · m^{-2} · K^{-4}

I. Introduction

VERY small satellites, defined here as having a subkilogram mass, have the potential to enable a new class of distributed space missions by merging the concepts of distributed satellite systems [1] and terrestrial wireless sensor networks [2]. Many new distributed space mission concepts require hundreds to thousands of satellites for simultaneous multipoint sensing to accomplish advanced remote sensing and science objectives.

Current very-small-satellite research and development efforts are based on labor-intensive or custom-manufacturing processes with inherently high unit costs. Two potential alternatives are presented that focus on a low-cost approach by leveraging existing commercial mass-production capabilities: a satellite on a chip (SpaceChip) [3–5] and a satellite on a printed circuit board (PCBSat) [6].

This paper reviews distributed satellite systems and introduces novel concepts and solutions. The motivation for this work is given in the first section, with a discussion on distributed space missions in which a reference mission is considered for application. Current and emerging very-small-satellite design concepts are then reviewed. Finally, the design and results of initial satellite-on-a-chip and PCBSat concepts are presented and evaluated for mission suitability.

II. Distributed Space Missions

The interchangeable terms *distributed satellite system* and *distributed space system* evoke the promise of realizing missions that have not been previously possible, whereas the term *constellation* is typically associated with a simpler form of the concept. Jilla et al. [7] defined a distributed satellite system as "a system of multiple satellites designed to work in a coordinated fashion to perform a mission." Burns et al. [8] expanded the definition to "an end-to-end system including two or more space vehicles and a cooperative infrastructure for science measurement, data acquisition, processing, analysis, and distribution." Shaw et al. [1] offered the most complete definition, identifying two formal types. The first relates to system implementations in which multiple satellites are sparsely distributed in a traditional constellation to meet mission requirements. Constellation scenarios do not typically require precise orientation between spacecraft but may optionally require propulsive stationkeeping. Satellites in a constellation are linked via ground relays and systems, with the rare exception of crosslinks or intersatellite links.

The second distributed satellite system type classified by Shaw et al. [1] introduces the concept of a local cluster, in which satellites

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*Major, U.S. Air Force, Ph.D. Student, Surrey Space Centre; david.barnhart@ieee.org. Senior Member AIAA.
†Senior Lecturer, Surrey Space Centre.
‡Director, Surrey Space Centre. Senior Member AIAA.

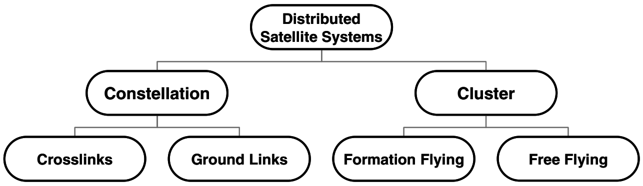

Fig. 1 Distributed satellite systems.

are intentionally placed close together in the same orbit to train on a common target. Optionally, this cluster of satellite nodes may have a more complex instantiation, commonly referred to as a formation. Formation flying requires that satellites in a cluster maintain precise spacing and orientation relative to each other, with the level of precision based on mission requirements. An ideally placed formation would only briefly exist before orbital perturbations disturbed the arrangement. This requirement directly implies that the spacecraft must have precise real-time location knowledge of all nodes and a propulsion system to maintain the formation. The motivation for formation flying is to synthesize a virtual aperture, antenna, or other sensor to attain mission performance levels that currently cannot be achieved by a monolithic satellite. Most aspects of this concept have been widely studied, but the first implementation has yet to be realized, with the exception of a few initial experiments.

A distributed satellite system taxonomy is shown in Fig. 1, with a discussion of current and planned systems in Secs. II.A and II.B. In Sec. II.C, a candidate distributed space mission is presented as a common reference for comparison of very-small-satellite technologies.

A. Current Distributed Satellite Systems

Table 1 presents a selection of current distributed satellite systems, grouped in the four typical mission categories: communications, navigation, remote sensing, and science. The first, largest, and best example of a distributed communications system is the $5 billion Iridium global mobile telephone network launched in 1997 [9]. Iridium is the only commercial constellation that employs crosslinks [10].

Currently, there are two distributed navigation systems: the global positioning system (GPS) and the Russian equivalent, GLONASS [9]. The GPS constellation is composed of 24 satellites in semisynchronous orbits, placed evenly in six planes to provide position and timing information to users on land, sea, air, and space.

Small satellites have recently entered the Earth-observation market. For example, the Disaster Monitoring Constellation (DMC) is the first commercial Earth-imaging constellation [11]. It offers an unprecedented revisit time of 24 h, versus days or weeks available from other systems.

The Cluster mission launched in 2000 is arguably the first satellite cluster to gather scientific data on the magnetosphere in three dimensions. Cluster is a maintained constellation of four satellites that forms a tetrahedron of various geometries on a periodic basis [12]. Similarly, the Earth-observation system (EOS) is a coordinated collection of 17 satellites performing various types of remote sensing and science missions. The segment of the EOS most interesting to this research is referred to as the A-train, which is a set of six closely spaced satellites in the same orbit, with the smallest distance being 100 km between CALIPSO and CloudSat [13]. The recent launches of ST5, FORMOSAT-3/COSMIC, and THEMIS indicate a growing interest in distributed science missions. However, system costs are still well out of reach for many scientific programs.

B. Emerging Distributed Satellite Systems

There has been a recent literary explosion of distributed mission topics. For example, the terms distributed satellite systems, satellite formation flying, and satellite cluster have become popular in AIAA publications, as highlighted in Table 2. Before 1996, "satellite cluster" describes close spacing in geostationary orbit (GEO).

Considering communication missions first, Ashford [14] noted that current realities fall short of previous predictions of a boom in low Earth orbit (LEO)-based distributed communication missions. For example, a large-scale system that never materialized was Teledesic. With conceptual designs ranging up to 840 satellites costing $5 million each, Teledesic was to provide the first global "internet in the sky." The Teledesic mission was abandoned after witnessing the technical successes and economic struggles of the Iridium, Globalstar, and ORBCOMM constellations. Fortunately, renewed interest in these existing constellations has been fueled by emerging applications, encouraging investors to replenish these constellations. Norris [15] proposed that clusters of small satellites operating in LEO will eventually be used to "virtually" replace larger monolithic telecommunication satellites. This may become reality as the GEO belt fills up, especially over the most populated areas of the Earth. Another variant of this idea, put forth by Edery-Guirardo et al. [16], is to augment larger satellite missions with a constellation of smaller communication relay satellites. However, large satellites in

Table 1 Selected distributed satellite systems [9–13]

Mission type	System	Satellites	Type	Satellite mass, kg	System cost, million USD
Communication	Iridium	66	Constellation with crosslinks	689	~5000
	Globalstar	24	Constellation with ground links	222	Unknown
	ORBCOMM	26	Constellation with ground links	22	~330
Navigation	GPS	24	Constellation with ground links	989–1077	>2000
	GLONASS	24	Constellation with ground links	~1400	Unknown
Remote sensing	DMC	5	Constellation with ground links	166	40
Science	Cluster	4	Free-flying cluster	1200	315
	EOS	17	Constellation with ground links	Varied	Unknown
	ST5	3	Constellation with ground links	25	130
	COSMIC	6	Constellation with ground links	69	55
	THEMIS	5	Constellation with ground links	128	200

Table 2 Distributed satellite system terms in AIAA publications as of April 2007

Year	Distributed satellite systems	Satellite formation flying	Satellite cluster
Before 1991	0	0	19
1991–1995	0	0	10
1996–2000	27	16	23
2001–2005	41	82	80
2006–present	6	19	19

GEO appear to be the mainstay of high-bandwidth global communications for the near term.

GPS, GLONASS, and the upcoming Galileo mission have already been categorized as constellations using ground links. Neither crosslinks nor clusters have been proposed for distributed navigation systems. Instead, the current focus is on their vulnerability to jamming [17]. For the GPS system in particular, next-generation systems will mitigate this vulnerability with the combination of higher-power radio frequency (RF) signals and other antijam technologies, which will cause the satellite mass to rise from 1000 to over 1500 kg. The threat of jamming will likely grow, requiring larger systems with increased RF power.

There are numerous envisioned distributed remote sensing systems; however, very few of them have gone beyond the conceptual or experimental phase. A short list of constellation-based mission examples is presented, which require simultaneous multipoint sensing: 1) volcano, fire, or earthquake preemptive warning and detection; 2) treaty monitoring (Kyoto protocol, RF, nuclear, and others); 3) distress-beacon monitoring; 4) space control, signals intelligence, and other military missions [18]; 5) particular imaging with frequent temporal repeats and high spatial resolution; 6) constellation sharing in which contributing members access the services of the entire group; and 7) disposable, short-lived, rapid-response sensor networks for use in LEO and the upper atmosphere [19]

Reconfigurability of the satellite nodes would be required for more advanced missions such as 1) beam forming to remotely sense a particular location at optical or radio wavelengths and 2) minimizing power expenditure by dynamically optimizing RF links.

Satellite formation flying is a promising but unrealized system architecture for missions such as space-based radar, as suggested by Das et al. [20]. Clusters of co-orbiting inspectors of larger satellites, the space shuttle, or the International Space Station are suggested by Macke et al. [21] for distributed field measurements.

Science and exploration missions have traditionally been dominated by single-spacecraft or interplanetary probe architectures due to typically limited science budgets and resources. New simultaneous multipoint sensing missions are being considered based on small satellites, such as 1) magnetotail behavioral studies, solar wind variations, and other geospace science [22]; 2) interplanetary exploration based on a satellite on a chip [23]; 3) monitoring and warning of large-area space phenomena, mainly space weather, including plasma and radiation density [24]; 4) monitoring wide-area highly time-dependent phenomena such as atmospheric drag or aurora in LEO; 5) detailed characterization of environments to support interplanetary exploration, such as Mars, asteroids, or other planets; and 6) upper-atmosphere monitoring (e.g., CO_2 levels at 60–250 km).

Reconfigurability of satellite nodes would also be required for more advanced missions such as 1) measuring ion- or electron-scale space weather events and effects within the magnetosphere and 2) compensating for interference from other sources such as radiation [lightning, trapped radiation (e.g., the south Atlantic anomaly), and stray electromagnetic fields] by frequency hopping.

The Terrestrial Planet Finder (TPF) mission is one of the few serious formation-flying proposals for science and exploration that is currently under study [25]. TPF will employ a formation-flying cluster at one of the sun–Earth libration points to synthesize a very large aperture to see further in the universe than ever before. A simple cluster mission proposed by Herrero et al. [22] would measure magnetic field variations around spacecraft or perform visual inspections of its exterior. Asteroid mapping and in-flight calibration of a communications beam pattern was also proposed. The Orion–Emerald mission was proposed as a formation-flying demonstration [26].

Carpenter et al. [27] outlined the challenges associated with formation flight, because it is a very complex, multifaceted problem. Consequently, no complete formation-flying missions have been implemented beyond a few initial experiments. Once enough of the formation-flying problems are resolved, perhaps very small satellites, reviewed in Sec. III, could support these missions as well.

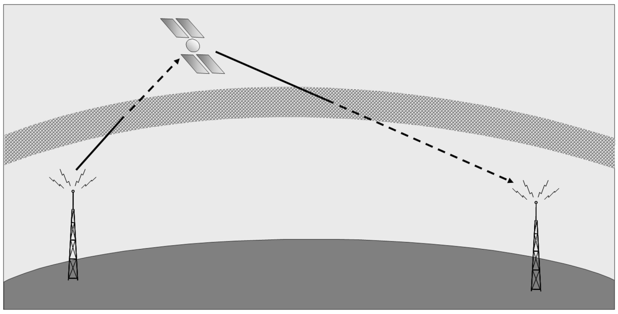

Fig. 2 Satellite communication signal scintillation.

C. Reference Distributed Space Mission

As previously discussed, there is an unrealized family of space weather missions for simultaneous measurements of phenomena over a large volume. One interesting mission is the detection, mapping, and study of ionospheric plasma depletions, otherwise known as plasma bubbles [24]. This phenomenon, which typically occurs in LEO at low latitudes after sunset, is believed to cause communication and navigation satellite signal outages by scintillating the signal, as depicted in Fig. 2 [24]. The understanding gained by such a mission would positively impact private, commercial, government, and military sectors, which depend on satellite communication and navigation for commerce, political stability, and military operations. This mission is used as a common reference to compare the supporting technologies to follow.

To demystify the phenomenon, simultaneous distributed observations, on the order of hundreds to thousands, would be required from a nonmaintained constellation. This implies that a low-cost mass-producible sensor node would have to be developed to accomplish the mission. In support of this goal, a mass of 1 kg is set as the upper limit of options being considered. At a minimum, each sensor node must be able to measure the plasma activity, stamp the data with the time and location taken, and then relay the data. Because of the proposed small mass, it is unlikely that each satellite would relay its data to the ground. Alternatively, the sensor nodes would implement a short-range mesh network, supported by a co-orbiting master relay satellite. Before a complete architecture can be developed for such a mission, a better understanding is required of the cost and performance of potentially enabling very-small-satellite designs.

III. Very Small Satellites

Since the dawn of the space age in 1957, increasing mission requirements have driven up satellite mass from Sputnik's 84 kg to over 6000 kg for some systems today. Consequently, cost, complexity, program timelines, and management overhead have grown considerably.

Reversing this trend, a fast-growing small satellite industry, rooted in academia, has enabled increasingly capable and cost-effective space missions. Focusing on sub-500-kg satellites, their success is based on embracing sensibly reduced requirements and leveraging commercial technology. To compare the capabilities of satellites, the space community generally agrees on the mass classification shown in Table 3.[§] Approximate mission costs are also listed. The preponderance of missions has been in the minisatellite and microsatellite ranges, as shown in Fig. 3. The focus of this research is on the downward trend from nanosatellites to picosatellites and potentially femtosatellites.

A. Current Very-Small-Satellite Technologies

Nanosatellites are not considered very small satellites here, but it is important to note that this mass range is currently considered the lowest practical end at which full satellite functionality can be

[§]Data available online at http://centaur.sstl.co.uk/SSHP/nano/index.html [retrieved 20 Feb. 2006].

Table 3 Satellite categories by mass and approximate costs

Category	Mass, kg	Cost, USD
Large satellite	>1000	0.1–2 B
Medium satellite	500–1000	50–100 M
Minisatellite	100–500	10–50 M
Microsatellite	10–100	2–10 M
Nanosatellite	1–10	0.2–2 M
Picosatellite	0.1–1	20–200 K
Femtosatellite	<0.1	0.1–20 K

achieved. For example, the $2 million, 6.5-kg SNAP-1 mission was the first nanosatellite to demonstrate a complete set of satellite capabilities typically found in larger satellites, including full attitude and orbit determination and control [28]. Other ongoing developments are exploring the use of nanosatellites for distributed mission applications [29,30].

Twenty picosatellites have flown since 2000, as summarized in Table 4 [31–33]. The first picosatellite mission was hosted by the Orbiting Picosatellite Activated Launcher (OPAL) on a Minotaur launch vehicle. Six custom-built picosatellites were deployed, but only Picosat 1A/1B was functional. The tethered pair, considered one object, carried a few experiments powered by a primary battery and transmitted their data to Earth. Later in 2000, Picosat 1C/1D, similar to 1A/1B, separated from the MightySat 2.1 host satellite and was also a success.

The 13 remaining picosatellite missions were developed using the CubeSat educational satellite standard, defined by Stanford University and the California Polytechnic Institute [34]. CubeSat has addressed some important issues by reducing the complexity of satellite design, especially the launch-vehicle interface. The design concept is essentially a scaled-down version of larger satellite designs using miniaturized modules and a standard form factor of $10 \times 10 \times 10$ cm or increments thereof. CubeSats are now available as a commercial kit that gives developers a basic structure and flight computer for $5000. The payload and required subsystems must be developed or purchased, in addition to launch costs of about $40,000 [35].

In 2003, the Eurockot launch deployed the first five subkilogram CubeSats, but only two were declared successful [31]. In 2005, the Student Space Exploration and Technology Initiative (SSETI) mission deployed three more subkilogram CubeSats from a Kosmos-3M launch vehicle, with two being successful. Unfortunately, 14 CubeSat systems were destroyed by a launch vehicle failure in July 2006 [31]. Five more subkilogram CubeSats were launched in

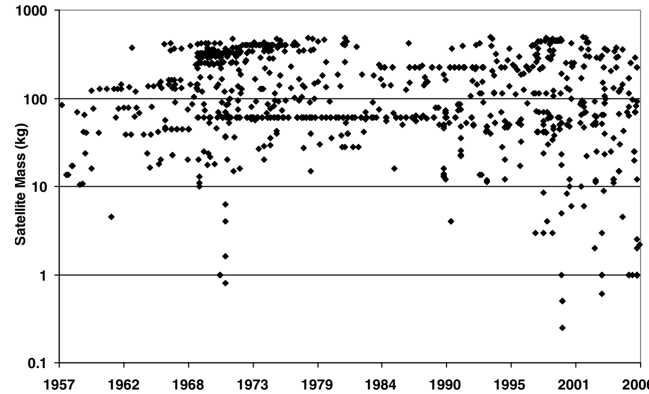

Fig. 3 Satellite mass histogram below 500 kg.

April 2007, with more launches planned. Currently, picosatellites have a reported success rate of 45% overall. Based on the results of the successful CubeSat missions, this technology may be able to support the reference distributed science mission.

No femtosatellites have flown in space, other than experimental test structures [36]. Reference [37] presented a femtosatellite design for spacecraft inspection based on a glass/ceramic structure and microelectromechanical systems (MEMS) sensors in 2002. Cyrospace, Inc. has announced a femtosatellite design; however, it is advertised as having a 500-g mass.

B. Emerging Very-Small-Satellite Technologies

Since 1998, Helvajian and Janson [38–41] have pioneered microengineering of aerospace systems based on MEMS and microfabrication. One of the earliest concepts was an "all silicon" approach, in which satellites are mass-produced by stacking up payloads and subsystems built on silicon wafers [42]. Xuwen et al. [43] and Shul et al. [44] published similar concepts, but no follow-up to their work has emerged. Integrating MEMS with complementary metal-on-silicon (CMOS) technology, which is the most common integrated-circuit (IC) fabrication technology, is highlighted in all these efforts as an essential development for very small satellites.

Huang et al. [37] also developed the co-orbiting satellite assistant (COSA) concept, proposed initially with a 100-g-mass baseline. COSA is intended for a week-long satellite-inspection mission, with most of the design effort going into the propulsion and structural subsystems. The concept has been recently revised, targeting a 1-kg configuration [45]. According to [45], seven wafers made of Foturan$^{\mathrm{TM}}$, which is a glass/ceramic material that is restructured

Table 4 Summary of picosatellite missions as of April 2007 [31–33]

Mission	Satellite	Bus	Mass, kg	Status
OPAL 2000	Picosat 1A/1B	Custom	0.275 each	Success
	Thelma	Custom	0.5	Never operational
	Louise	Custom	0.5	Never operational
	JAK	Custom	0.5	Never operational
	Stensat	Custom	0.23	Never operational
	MASat 1	Custom	0.5	Never operational
MightySat 2.1 2000	Picosat 1C/1D	Custom	0.275 each	Success
Eurockot 2003	CubeSat-XI-iV	CubeSat	1	Success
	DTUSat	CubeSat	1	Never operational
	CUTE-I	CubeSat	1	Success
	CanX-1	CubeSat	1	Never operational
	AAU CubeSat	CubeSat	1	Premature failure
SSETI 2005	CubeSat XI-V	CubeSat	1	Success
	UWE-1	CubeSat	1	Success
	Ncube-2	CubeSat	1	Never operational
DNEPR 2007	AeroCube-2	CubeSat	1	Premature failure
	CAPE1	CubeSat	1	Success
	CP3	CubeSat	1	Never operational
	CP4	CubeSat	1	Success
	LIBERTAD-1	CubeSat	1	Success

using a laser, require 20 h of processing each, totaling 140 h for one satellite. However, production times are predicted to approach 10 min per wafer. A conventional PCB is used for the supporting electronics. Startup costs are initially estimated at $600,000, with a unit cost of $30,000.[¶] These costs are for the structural and propulsion subsystems alone, which dominate the mass fraction of the satellite. Because COSA is primarily designed for in-orbit maneuvers and spacecraft inspection, it may not be well suited for a large-scale distributed science mission, in which the emphasis is on the payload, electrical power, and communication capabilities.

In parallel with the introduction of microengineered aerospace systems, the concept of multifunctional structures and architectures was introduced, also backing the idea of low-cost mass production of satellites [46]. This approach fostered the responsive space movement, which proposes that satellites be built and rapidly deployed using streamlined manufacturing processes and modular technologies [47,48]. Similarly, Bruhn and Stenmark [29] led an effort to reduce satellite mass and volume by orders of magnitude with multichip module (MCM) technology. Their approach is based on a proprietary architecture called Multifunctional Micro Systems (MMS). A third party recently licensed the MMS technology with the goal of mass-producing nanosatellites for $4 million and picosatellites for $2 million that have capabilities of much larger satellites.[**]

Two new very-small-satellite concepts are under investigation, focused on low cost and mass producibility, using current commercial practices. SpaceChip is a proposed monolithic satellite on a chip based on commercial CMOS, with a projected unit cost of $1000. The basic idea behind a satellite on a chip is to put the entire functionality of a satellite on a chip, which is typically a thumbnail-sized IC. Perhaps the first mention of a satellite on a chip can be attributed to an interview [49] in 1994. Since then, many have proposed satellite-on-a-chip technology as the ultimate goal for spacecraft miniaturization, but none have published any significant results [3–5]. The preliminary design of SpaceChip is further explored in Sec. IV.

The satellite-on-a-PCB approach was originally aimed at developing a prototype to guide the SpaceChip architecture [6]. The goal of PCBSat now is to determine if the reference distributed science mission can be supported with a 100-g sensor node built with commercial components and fabrication technologies. A unit cost of $500 may be possible, based on initial results. The design of the PCBSat prototype is presented in Sec. V.

IV. Satellite-on-a-Chip Preliminary Design

At this stage of research, SpaceChip is a preliminary design based on Wertz and Larson's [50] space mission analysis and design (SMAD) principles. Considering the reference science mission, a fleet of SpaceChips would be deployed from a host small satellite in LEO. The mission will be performed for as long as the fleet of SpaceChips remains in communication range of the host satellite, which would naturally drift apart due to the effects of differential drag. The host satellite would then relay the space network data to the ground.

A satellite is typically composed of a payload and a set of supporting subsystems including structural, electrical power (EPS), data handling (DH), communications (Comm), attitude/orbit control (AOCS), and thermal control (TCS), depending on mission requirements. The rest of this section maps each subsystem from the mission requirements to a conceptual design. A notional configuration of a satellite on a chip is illustrated in Fig. 4, and a summary of the mission requirements and outcomes is given in Table 5.

[¶]Private communications with Henry Helvajian, The Aerospace Corporation.

[**]Data available online at www.globeinvestor.com/servlet/ArticleNews/story/CNW/20051125/C9967 [retrieved 1 Nov. 2006].

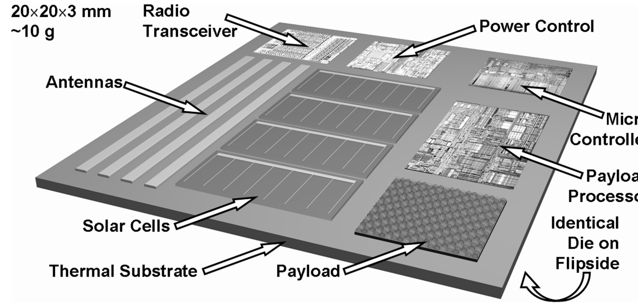

Fig. 4 Notional SpaceChip configuration.

A. SpaceChip Configuration and Structure

At the beginning of the satellite-on-a-chip work, a true monolithic system-on-a-chip (SoC) implementation was viewed as truly encompassing the spirit of the idea. However, a monolithic approach, which does not allow the attachment of discrete components or the merging of various elements into a hybrid assembly, imposes considerable limitations. Most notably, the design cannot exceed the reticle size, which is an area limit imposed by the photolithography process used in the particular semiconductor process line. This caps the maximum circuit area to approximately 400 mm^2 (20×20 mm) for modern CMOS processes. Assuming a silicon density of 2330 kg/m^3 and wafer thickness of 0.75 mm, the die mass would be approximately 1 g.

In 1967, a technique called wafer-scale integration (WSI) was proposed to overcome the reticle limit [51]. WSI allows multiple reticle-sized designs to be colocated on the same wafer and then connected using various interconnection techniques. This would allow a final product that, in theory, could be as large as the entire wafer, which could be as large as 300 mm in diameter. Unfortunately, inherent defects in the semiconductor manufacturing process have prevented WSI from becoming widely adopted [52].

MCM technology eventually replaced WSI for designs requiring more area [52]. MCMs integrate unpackaged "known good die" on a range of substrates such as PCBs, thin films, and ceramics using fine-line interconnects. MCM technology, including three-dimensional variants, has already been used in satellite applications [53]. MCMs or other system-in-package (SIP) techniques are typically used in applications in which integrated density or performance is essential [54]. For less-demanding applications, advancements in IC packaging make traditional PCBs a logical choice.

Despite a growing number of packaging alternatives, SoC technology is rapidly advancing. In fact, high-profile MCM-based miniaturization projects, such as the Smart Dust wireless sensor node, are now investigating SoC [55]. CMOS technology is the most widely used microelectronics fabrication technology, due to its low cost at high volume. Currently, feature sizes of 45 nm are common, which will only shrink in time. CMOS technology options have broadened over the past few years with the introduction of processes optimized for RF, optical sensors, and integration of bipolar transistors (SiGe-BiCMOS). Nonvolatile memory can also be integrated. A full-reticle prototype design using a multiproject vendor such as MOSIS or EUROPRACTICE can cost as much as $2400 per die, whereas a production run would cost less than $500 each.

B. SpaceChip Payload

The chosen SoC approach greatly limits payload options. Considering the reference mission, no sensors on a chip scale are possible at this time to detect plasma-bubble phenomenon, due to the physical geometries required.[††] For demonstration purposes, a visible CMOS imager is being considered as the payload for the satellite-on-a-chip feasibility study after a survey of potential payloads [56]. In addition to CMOS imagers, other monolithic

[††]Private communications with Lon C. Enloe, U.S. Air Force Academy, Department of Physics.

Table 5 SpaceChip system requirements and verification

System	Requirements	Outcomes
Top level	SpaceChip shall be implemented on a commercial CMOS process, suitable for integration of digital, analog, and RF components.	austriamicrosystems SiGe-BiCMOS 0.35 μm
	SpaceChip shall meet all mission objectives and support the operations concept.	
Payload	The payload shall detect the space weather phenomenon of interest.	Limited options
	A simple demonstration payload shall be considered.	CMOS imager
Orbit	SpaceChip shall operate in an orbit to support the mission.	~500-km altitude Low inclination
Configuration and structure	Configuration shall be a monolithic "satellite on a chip."	Deviations required
	Size shall not exceed typical CMOS process reticle limit.	20×20 mm maximum
	Mass shall be less than 10 g.	~10-g package
	The design shall incorporate a launch-vehicle interface.	TBD
EPS	Power source shall be solar energy via integrated photovoltaic cells.	~1-mW budget
	Secondary power storage shall be investigated.	No monolithic option
DH	Shall be based on a low-power reduced-instruction-set microcontroller.	Design-hardened asynchronous microcontroller
	Nonvolatile memory technologies shall be investigated.	
	Design shall withstand natural radiation environment.	
Comm	2.4-GHz unlicensed ISM band shall be used.	1-μW RF, 1-km range External antenna required
AOCS	Attitude determination shall not be required.	Passive ADCS
	Orbit-determination options shall be investigated.	No orbit determination possible
Propulsion	Propulsion shall not be required but options shall be investigated.	No monolithic option
Thermal	Passive control shall be used.	Thermal substrate

CMOS sensor technologies were investigated; for example, infrared, magnetic field, radiation, and pressure sensors have been demonstrated [57].

More interesting sensors are possible due to the emergence of CMOS MEMS technology, in which MEMS is monolithically integrated with CMOS on a limited basis [58]. Pressure, chemical, thermal, tactile, proximity, flow, force, neural, vacuum, acceleration, gyroscopic, and sound sensors are now possible. CMOS MEMS applications require custom pre-, front-end, and back-end processing. Because of the growing popularity of this approach, a few commercial foundries now offer limited CMOS MEMS processing, such as X-FAB, Inc.

Table 6 SpaceChip power budget

System	Typical [50]	Design	Units
Payload	40%	80	μW
EPS	20%	40	μW
DH	10%	20	μW
Comm	30%	1	mW
ADCS	0%	0	
Propulsion	0%	0	
Thermal	0%	0	
Structure	0%	0	
Total	100%	1.14	mW

C. SpaceChip Electrical Power Subsystem

Power distribution, regulation, and control aspects of an EPS can be met with the basic wiring, switching, and regulation circuitry that are routinely implemented in CMOS [59]. However, power generation via integrated solar cells on CMOS presents the greatest challenge. Typically, solar cells are fabricated with optimized silicon or gallium arsenide processes, distinctly different from commercial CMOS. Integrating solar power with digital circuitry has not been of interest until recently. Again, using Smart Dust as an example, the design first featured an integrated battery, then used MCM integration to incorporate solar cells, and finally demonstrated a monolithic solution using a custom silicon-on-insulator (SOI) process [60]. Although SOI is growing in popularity, it is not yet widely available.

A few monolithic self-powered devices on bulk CMOS have been demonstrated. Three such examples are a sensor network node [61], a retinal implant [62], and an experiment to provide local power at each logic gate [63]. Of these efforts, the best efficiency achieved is 1%.[‡‡] Bermejo et al. [64] explain that the CMOS process imposes some restrictions that drastically reduce the efficiency of integrated solar cells, indicating why a SIP approach is typically used.

Using a baseline value of 80 μW for a CMOS imager payload [56], Table 6 presents the power budget, which totals 1.14 mW, dominated by the communication subsystem, described in Sec. IV.E. All other subsystem power requirements are based on the typical minimum values for small satellites [50].

With an initial power budget, the EPS sizing process is straightforward, based on SMAD [50] equations. All parameters, intermediate results, and final results are shown in Table 7. Equation (1) is first used to calculate an orbital period of 94.6 min,

Table 7 SpaceChip electrical power subsystem sizing

Parameter	Assumed value [50]	Result	Equation
h	500 km		
$R_\oplus$	6378 km		
a	6878 km		$a = h + R_\oplus$
$\mu_\oplus$	$3.986 \times 10^5\ \mathrm{km}^3 \cdot \mathrm{s}^{-2}$		
P		94.6 min	(1)
ρ		68 deg (1.187 rad)	(2)
T_e		35.7 min	(3)
T_s		58.9 min	$T_s = P - T_e$
P_e	100 μW		
w		214 mWs or mJ	$w = P_e T_e$
v	2.5 V		
C		68.5 mF	(4)
P_s	1.14 mW		
X_s	0.85		
P_e	0		
P_{sa}		1.34 mW	(5)
θ	45 deg		
G_s	$1326\ \mathrm{W} \cdot \mathrm{m}^{-2}$		
η	1%		
I_d	100%		
P_{BOL}		$9.4\ \mathrm{W} \cdot \mathrm{m}^{-2}$	(6)
A_{sa}		11.9×11.9 mm	$A_{sa} = P_{sa}/P_{BOL}$

[‡‡]Private communications with David Blaaw, University of Michigan, Department of Electrical Engineering.

and Eq. (2) gives an Earth angular radius of 68 deg, based on a notional altitude of 500 km. From the results of Eqs. (1) and (2), Eq. (3) gives a time in eclipse of 35.7 min. Subtracting this value from the calculated period gives a sunlit time of 58.9 min.

A capacitor is assumed to be the only possible method of monolithic power storage. Using a 10% duty cycle for all systems during eclipse ($\sim$100 μW), a total power-storage requirement of 214 mJ is found from the product of the eclipse power requirement and time in eclipse. Equation (4) gives an integrated capacitance requirement of 68.5 mF. Even using the high-capacitance option of 4.8 $\mathrm{fF} \cdot \mu\mathrm{m}^2$ in SiGe-BiCMOS, this would require an area of 40,000 times the maximum reticle area, which conclusively rules out integrated power storage. An external thin-film battery could be considered, if required.

To determine the required solar array area, an average solar array output power requirement of 1.34 mW is found with Eq. (5), assuming no eclipse operations. Finally, Eq. (6) reveals a beginning-of-life areal power output of 9.4 $\mathrm{W} \cdot \mathrm{m}^{-2}$. The results from Eqs. (5) and (6) give an array size of 11.9×11.9 mm, which is 35% of the maximum reticle area. This is a promising result, even with an efficiency of 1%.

$$P = 2\pi \sqrt{\frac{a^3}{\mu_\oplus}} \tag{1}$$

$$\rho = \sin^{-1}\left(\frac{R_\oplus}{R_\oplus + h}\right) \tag{2}$$

$$T_e = \frac{2\rho}{360\,\mathrm{deg}} P \tag{3}$$

$$w = \tfrac{1}{2} C v^2 \tag{4}$$

$$P_{\mathrm{sa}} = \left(\frac{P_s T_s}{X_s} + \frac{P_e T_e}{X_e}\right) \Big/ T_s \tag{5}$$

$$P_{\mathrm{BOL}} = G_s \eta I_d \cos\theta \tag{6}$$

One broad-scope issue that complicates the puritan satellite-on-a-chip idea is that the resulting design is inherently two-dimensional, using only one side of the wafer. Such a configuration is unacceptable, because the system could go long periods without power if the inactive side faces the sun. Because of these physical constraints, a proposed deviation from the pure satellite-on-a-chip definition should be considered. SpaceChip could be composed of two identical 20×20 mm die sandwiched together, with the active sides facing outward. No die interconnects would be required, because only one side at a time will be active due to solar illumination.

D. SpaceChip Data Handling Subsystem

The DH subsystem provides a range of onboard computing services. It receives, validates, decodes, and distributes commands from the ground, payload, or subsystem to other spacecraft subsystems. It also gathers, processes, and formats spacecraft housekeeping and mission data for downlink or use onboard. DH subsystems are usually the most difficult to define early in the design, due to the initially vague hardware and software requirements of the payload and subsystems.

At a minimum, the DH subsystem is composed of a central processing unit (CPU) and supporting memory elements. The difficult part of the design is the hardware interface to the other systems, typically using a digital data bus and analog-to-digital converters (ADC). For SpaceChip, a small reduced-instruction-set (RISC) CPU design is all that can be supported by the available power. Some introductory thought has already been given to miniaturizing flight-computer components to a single chip, reflecting a growing trend in SoC development [65].

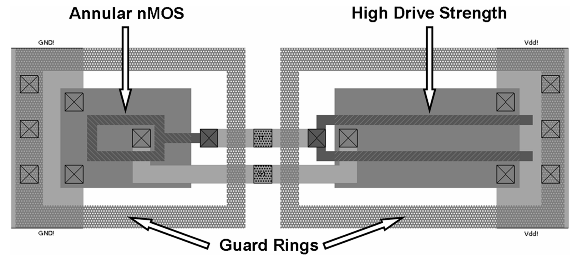

Fig. 5 Design-hardened inverter.

A reset function and external clock are normally required to power up and run a CPU. To provide the reset signal, an on-chip pull-up resistor can be used. For this application, a clock source can be generated on the chip with a ring oscillator. To further reduce power, asynchronous logic is proposed, which will also eliminate dependence on an external oscillator [66].

One issue that plagues data handling systems operating in space is the natural radiation environment. Radiation effects include gradual system degradation caused by the total-ionizing-dose and single-event phenomena induced by high-energy particles such as electrons, protons, and heavy ions [67]. The problem can be solved at the integrated-circuit level using a CMOS device layout technique called *design hardening* to mitigate both total-ionizing-dose and single-event phenomena, as illustrated in Fig. 5 [67]. The technique is not without fault, because there are power and area penalties. By laying out the n-type transistors in an annular shape, the mechanisms that cause transistor leakage from ionizing radiation are eliminated. Increasing the drive strength (width) of the transistors increases the threshold at which single-event upsets occur from high-energy particle strikes. Finally, adding $p+$ and $n+$ (highly doped material used in the CMOS fabrication process) guard rings around the transistor areas prevent single-event latch-up. The increased power requirement can be offset by using asynchronous logic and sacrificing area, which has already been demonstrated for space applications [66].

E. SpaceChip Communications Subsystem

An obvious challenge for a satellite on a chip is the communications link between the ground and the satellite. Because of its limited size, the onboard RF transmit power must be significant enough for an effective downlink. Initial calculations revealed that the corresponding electrical power to generate the minimum downlink RF power would require an integrated solar array area of at least 50 cm^2, which is much greater than the 400 mm^2 area of the largest die. Tracking is another challenge, because the ground station must know exactly where the satellite is to avoid pointing losses with its required high-gain antennas. Because of the very small size of a satellite on a chip, it is unlikely that space surveillance networks could detect it.

The strategy to meeting these challenges is to avoid them altogether. One potential architecture would exploit a supporting satellite in orbit that could serve as the master relay to the ground station. Table 8 summarizes the communication subsystem performance for this architecture. SpaceChips could be massively distributed in that orbit with a maximum separation of 1 km, similar to terrestrial wireless sensor nodes. Equation (7) gives a free-space loss of -100 dB for this distance using a 2.4-GHz signal in the unlicensed instrumentation, scientific, and measurement (ISM) band. Assuming no line, atmospheric, rain, or polarization losses and a solid-state electrical-to-RF conversion efficiency of 1% [50], a low-power on-chip digital radio [68] would give a 19.6-dB link margin at 582 bits per second (bps), as found with Eq. (8).

Table 8 SpaceChip communication subsystem sizing

Parameter	Assume value [50]	Result	Equation
P_t	1 μW		
G_t	0 dB		
f	2.4 GHz		
c	3×10^8 m · s^{-1}		
λ		0.125 m	$\lambda = c/f$
S	1 km		
L_s	−100 dB		(7)
G_r	0 dB		
k	1.381×10^{-23} J · K^{-1}		
T_{sys}	21.3 dB · K		
Modulation	Binary phase shift		
Scheme	Keying (BPSK)		
Minimum E_b/N_o	9.6 dB		
Desired E_b/N_o	>19.6 dB		
R		582 bps	(8)

$$L_s = \left(\frac{\lambda}{4\pi S}\right)^2 \tag{7}$$

$$\frac{E_b}{N_0} = \frac{P_t G_t L_s G_r}{k T_{\mathrm{sys}} R} \tag{8}$$

Despite previously mentioned integrated solar cell inefficiency, the communication subsystem is the key factor that limits satellite-on-a-chip applicability. Even if sufficient solar power can be generated to produce the corresponding RF power, the effectiveness of integrated antennas on CMOS die becomes the issue. The maximum range achieved is approximately 5 m, as recently demonstrated by Lin et al. [69] and O et al. [70], which is almost two orders of magnitude less than the 1-km goal. Consequently, an external antenna must be considered, although the 582-bps data rate and subkilometer range greatly limits SpaceChip's application.

F. SpaceChip Attitude and Orbit Determination and Control Subsystem

Although no stressing attitude control requirements exist for the reference science mission, minimized body rates are ideal. Attitude control requirements can be met by using active and/or passive means. The magnetorquer/magnetometer combination is especially advantageous for very small satellites [71]. MEMS gyroscopes were also considered, but they cannot be effectively integrated with complete CMOS designs [58]. All other sensors and actuators currently used in small satellites are still major subcomponents, much larger than the size of a CMOS chip.

To stabilize attitude on this simple mission, two methods of passive control can be employed together. First, an on-chip inductor would serve as a passive magnetorquer in the x axis, which is in the plane of the die. Then, a passive aerodynamic "drag tail" would be employed on one edge. This approach is essentially a miniaturized version of that proposed in [72].

Orbit determination is very important to some missions, including the reference science mission. GPS has been acknowledged as an independent and reliable method for determining spacecraft position and velocity for small satellites. Single-chip solutions have been demonstrated, but they require numerous external passive components and over 24 mW of power [73].

G. SpaceChip Propulsion Subsystem

Orbit control is not possible without propulsion. Much work has been focused on propulsion for very small satellites. The most promising technology that may eventually be applicable is the digital micropropulsion effort [74]. However, this technology requires a high activation voltage, has difficulty delivering symmetric thrust, and cannot be integrated monolithically with CMOS.

H. SpaceChip Thermal Subsystem

The temperature extremes that a satellite on a chip would experience are estimated in Table 9. Using the previously calculated Earth angular radius found in Eq. (2), the flat plate over a spherical Earth model gives the corresponding view factors using Eqs. (9) and (10). Assuming the worst-case conditions for both cases, Eq. (11) gives a maximum temperature of 96°C and Eq. (12) gives a minimum temperature of −72°C. This temperature range is not unreasonable when compared with the operating range of industrial grade electronics (−40 to +85°C). Further laboratory verification is needed and any problems can most likely be addressed with a simple phase-changing thermal management substrate such as paraffin.

$$F_P = \sin^2\rho \tag{9}$$

$$K_a = 0.664 + 0.521\rho - 0.203\rho^2 \tag{10}$$

$$T_{\max(\mathrm{SA})} = \left[\frac{\alpha_t G_S + \varepsilon_b q_I F_P + \alpha_b a G_S K_a F_P - G_S \eta}{\sigma(\varepsilon_b + \varepsilon_t)}\right]^{1/4} \tag{11}$$

$$T_{\min(\mathrm{SA})} = \left[\frac{\varepsilon_b q_I F_P}{\sigma(\varepsilon_b + \varepsilon_t)}\right]^{1/4} \tag{12}$$

I. SpaceChip Technology Assessment

The concept of a satellite on a chip has been assessed by the notional design of SpaceChip already presented. The main advantage of this approach is the potential very low cost of under $1000 per satellite node in volume quantities, requiring two die each. Deploying 1000 SpaceChips would cost $1 million plus launch costs of $50,000 for the 10-kg mass, assuming a midrange launch cost of $5000/kg. The key performance parameters of the SoC approach are a 5-m communication range, 582-bps data rate, 1% solar cell efficiency, no eclipse operations, no orbit determination, and finally, the lack of a meaningful payload for the reference ionospheric-plasma-depletion mission. Although SpaceChip cannot currently support the reference mission, it is still potentially suitable for simple missions such as dosimetry, but any mission architecture will require a co-orbiting relay satellite. As SoC technology improves, perhaps SpaceChip will be able to support a wider range of interesting missions.

V. Satellite-on-a-PCB Prototype Design

Similar to SpaceChip, SMAD [50] principles are used throughout the design of PCBSat. However, the PCBSat design is a "bottoms-up" approach, in which a finite set of payload and subsystem components, constrained by commercial off-the-shelf

Table 9 SpaceChip thermal subsystem sizing

Parameter	Assume value [50][a]	Result	Equation
α_{Si}	0.48		
$\varepsilon_{\mathrm{Si}}$	0.46		
ρ		68 deg (1.187 rad)	(2)
F_p		0.86	(9)
K_a		0.99	(10)
G_s (hot)	1418 W · m^{-2}		
Albedo (hot)	0.35		
q_I (hot)	258 W · m^{-2}		
σ	5.67×10^{-8} W · m^{-2} · K^{-4}		
$T_{\max}$		96°C	(11)
q_I (cold)	216 W · m^{-2}		
$T_{\min}$		−72°C	(12)

[a]Data available online at http://www.gps.caltech.edu/genesis/Thermal-Coll.html [retrieved 1 Nov. 2006].

Table 10 PCBSat system configuration and test results

System	Requirement	Key components	Expected results	Actual results
Top Level	All COTS parts		\$300/prototype cost	\$285
	Commercial PCB			
Payload	CMOS imager	ST VS6502 640 × 480	99 mW	99 mW
	Plasma sensor	MESA	200 mW	TBD
Orbit	LEO		∼500 km	TBD
Configuration and structure	∼100-g mass		100–200 g	70 g, less structure
	PC104 PCB		9.0 × 9.5 cm	9.0 × 9.5 cm
	LV interface		P-POD	TBD
EPS	7-cell solar array	RS 276-124	788-mW peak power	500-mW peak power
	Regulated 3.3 V	MAX604	92% regulator η	92%
	PPT and BCR	MAX856/982	80% PPT/BCR η	82.7%
	Li-ion battery	Olympus Li-30B	645 mAh	432 mAh needed
	V&I telemetry	MAX4072		
DH	3.3-V RISC CPU	Atmel Mega128L	3.6864-MHz clock	3.6864 MHz
	128-K flash memory	Acroname USB	5-mA current draw	5 mA
	USB umbilical	AVRISP	Software size TBD	19 kB, 15% used
	ISP			
Comm	2.4-GHz ISM	MaxStream XBeePro	181-mW RX	170 mW
	RSSI telemetry		706-mW TX	700 mW
			1.3-km range	TBD
AOCS	Orbit determination	iTrax-03S GPS	GPS lock in 33 s	27 s
		Sarantel GeoH-SMP	100 mW	100 mW
	Active magnetic		33 mW	33 mW
	Passive aerodynamic	Structure TBD		TBD
Propulsion	None			
Thermal	2-channel telemetry	Thermistor	Functional	Functional

(COTS) parts availability, are integrated to determine the overall system capability, which in turn determines its range of applications. To keep part and manufacturing costs low, commodity surface-mount components are selected in the design. After two design revisions, the current system configuration is as shown in Fig. 6 and Table 10. Expected and actual results are also summarized here and explained briefly in the following sections.

A. PCBSat Configuration and Structure

The initial design of PCBSat has been purposely confined to a single PC104 form-factor configuration. The CAD tool selected for this project was EAGLE from CadSoft. The final Gerber PCB layer files were inspected for errors with Pentalogix Viewmate. The PCBs were then produced at a commercial facility in prototype quantities.

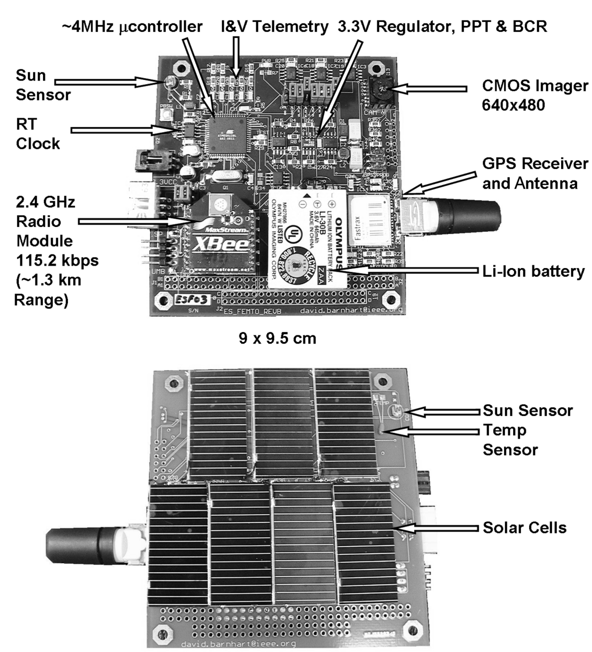

Fig. 6 Front and back views of PCBSat.

The final system configuration is envisioned to be a 10 × 10 × 2 cm block with a mass of 200 g, as shown in Fig. 7. It will be compatible with the existing Poly Picosat Orbital Deployer (P-POD) spacecraft separation mechanism, in which 15 PCBSats could be jettisoned from each P-POD [75]. Although aluminum is typically used, space-qualified plastic such as Delrin® will be considered.

B. PCBSat Payload

The demonstration payload is the STMicroelectronics VS6502 color CMOS imager with integrated lens. It has two-wire control (I^2C) and 5-bit data interfaces. The device snaps into a surface-mount socket mounted on the PCB.

The miniature electrostatic analyzer (MESA), shown in Fig. 8, has been identified as a candidate payload for the reference science mission [76]. MESA was designed, built, and tested in plasma environments created in terrestrial test chambers in 2002. Currently, in a standalone configuration with supporting electronics, it has a mass of 150 g, measures 5 × 5 cm, requires 300 mW of power, and has a low data-rate requirement. The sensor alone has a mass less than 50 g and requires about 200 mW of power.

C. PCBSat Electrical Power Subsystem

A spacecraft EPS is typically composed of four basic functions: power source, energy storage, power distribution, and power

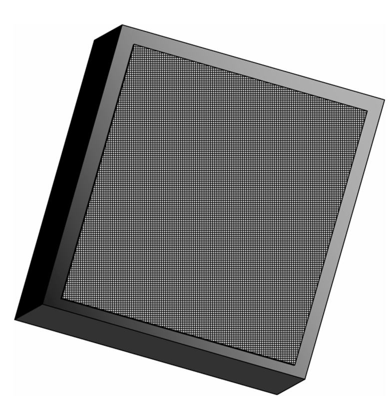

Fig. 7 Conceptual final design of PCBSat.

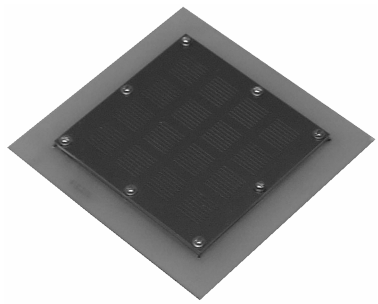

Fig. 8 MESA plasma sensor.

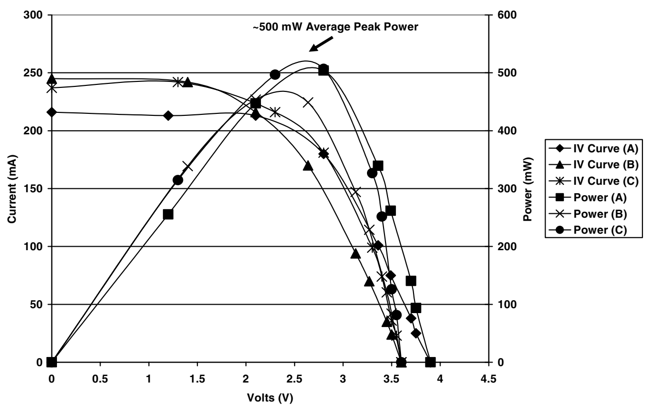

Fig. 9 Initial PCBSat solar array 4 curve results.

regulation and control. The basic EPS design of PCBSat is to use primary solar power, secondary rechargeable batteries, and deliver 3.3-V regulated power.

Solar energy was the obvious choice for the primary power source. Solar array peak-power tracking (PPT) was selected over the less complex, but inefficient, direct-energy-transfer option. This method places a smart interface between the solar array and battery to extract the maximum amount of power out of the solar array over a range of solar conditions. The measured efficiency of the PPT circuit is 82.7%. The PCBSat solar array delivers an average of 500-mW peak power, based on three test articles, as illustrated in Fig. 9. This result is significantly less than the expected 788 mW, based on the cell manufacturer's specifications. The tests were performed at noon on a clear day at 19°C, and so these results should only be considered as an initial estimate. Work is underway to evaluate the solar array in a calibrated solar simulator. Recalling Eq. (6), the average power output in orbit would then be 353 mW, assuming an average incidence angle of 45 deg.

Another important feature of a good EPS design is battery-charge regulation (BCR). The baseline design uses the MAX856/982 ICs from Maxim Integrated Products. Selecting a suitable battery was surprisingly difficult due to limited physical configurations. After evaluating several options, an Olympus Li-30b lithium-ion pack was chosen based on its form factor, mass, 645-mAh capacity, and 3.6-V operating point. Voltage regulation was achieved with a linear regulator (MAX604), which gives an efficiency of 92% when stepping down from 3.6 to 3.3 V.

A preliminary power budget is shown in Table 11 for PCBSat with the MESA payload. Recalling the nature of ionospheric plasma depletions, the phenomenon only occurs between dusk and dawn, which corresponds to orbital eclipse. Therefore, MESA and GPS (for time and location stamping) will only need to operate during this time. This gives a sunlit power requirement of 223 mW and an eclipse power requirement of 523 mW.

Using the previous orbit scenario for SpaceChip in Table 7, a measured sunlit-power-transfer efficiency of 76%, an estimated eclipse-power-transfer efficiency of 60%, and the preceding power results, Eq. (5) gives a solar array power requirement of 821 mW. Considering the average solar array power output of 353 mW discussed previously, more investigation is required to reduce power requirements and increase the solar array output. Considering the battery, a total required capacity for a single battery of 432 mAh is found using Eq. (13), assuming a depth of discharge (DOD) of 80% for LEO and a transmission efficiency of 90% between the battery and the load. This shows that the selected battery, having a capacity of 645 mAh, will be more than ample for the mission.

$$C_r = \frac{P_e T_e}{(\mathrm{DOD})n} \tag{13}$$

Finally, the PCBSat EPS monitors the following telemetry points: voltage and current of the solar array, battery, and the 3.3-V regulated power supply. Six ADC channels, part of the data handling subsystem discussed next, are used to measure these points, along with resistor-based voltage dividers and current sensing ICs (MAX4072).

D. PCBSat Data Handling Subsystem and Software Development

The chosen core of the DH subsystem is the Atmel Mega128L 8-bit AVR® low-power microcontroller, which is in-system programmable (ISP) via a six-wire programming interface to an AVRISP® dongle to a PC. It also has a boot-loader option, which is essential for updating software once deployed. CodeVisionAVR was selected as the software development environment.

The software for the project is currently at 6800 lines of code written in the ANSI C language. The compiled binary file is 19 kB in size, taking 15% of the available code space. Hardware interrupts signal-data collection and transmission. A significant amount of the code is required for the GPS and CMOS imager modules.

Radiation and charged particles, for which the fluence greatly varies with altitude, are two of the main problems addressed when flying COTS components in space. Long-term exposure to radiation causes a degradation of performance and increased power draw due to the total-ionizing-dose effect, which will not be a concern for short-lived missions such as those that PCBSat will support. Four to eight mil of cover glass for the solar cells and a small amount of aluminum spot shielding on the ICs can mitigate the accumulation of total dose. However, single-event effects will have to be tolerated and handled with software and a watchdog timer.

Table 11 PCBSat power budget

System	Typical, % [50]	Measured max, mW	Sunlit duty cycle, %	Sunlit power use, mW	Eclipse duty cycle, %	Eclipse power use, mW
Payload	40	200	0	0	100	200
EPS	20	385				
DH	10	18	100	18	100	18
Comm RX	30	170	100	170	100	170
Comm TX		700	5	35	5	35
AOCS	0	100	0	0	100	100
Propulsion	0	0				
Thermal	0	0				
Structure	0	0				
Total	100%	1603 mW		223 mW		523 mW

E. PCBSat Communications Subsystem

The PCBSat Communications subsystem is one of the key elements of the design, because the communication range is a significant parameter to trade off in distributed satellite system constellations. The main drivers are low-power, long-range communications and unlicensed operations. Similar to SpaceChip, the 2.4-GHz ISM band was chosen to eliminate licensing requirements.

A single-chip RF solution was sought, mirroring the SoC goal of SpaceChip. The Atmel ATR2406 ISM transceiver was used on the first revision of the PCB. However, due to its limited range and software requirements, the MaxStream XBeePro module, which is ZigBee/IEEE802.15.4-compliant, was selected. It has an electrical-to-RF transmission efficiency of 6.7% and a range of about 1335 m. It is connected to the one of the microcontroller's serial ports, configured at 115.2 kbps. A radio with more range will be needed when the mission architecture is better defined, such as the MaxStream XTend transceiver, which has a maximum range of 65 km with 25% RF efficiency.

F. PCBSat Attitude and Orbit Determination and Control Subsystem

No attitude control is required, but reducing the body rates is desired. Because of the mass, size, and power constraints, very limited AOCS options are available for PCBSat. For attitude determination, two cadmium sulfide (CdS) sensors, one on the front and the other on the back, are used to tell which face of the PCB is illuminated. For attitude control, a solid-state relay is used to power a magnetorquer, which is activated by one of the microcontroller's pulse-width modulated channels. Passive aerodynamic control will be investigated [72].

For orbit determination, the Fastrax iTrax-03S GPS module and Sarantel GeoHelix-SMP passive antenna were chosen. It is connected to the microcontroller via the second serial port, configured at 4800 bps. The module has a one-pulse-per-second output, which is connected to one of the Mega128's external interrupt ports for precise time and location stamping of payload data. For actual space application, terrestrial GPS receivers cannot be used as is. Because of the orbital velocity of about 7.5 km/s in low Earth orbit, the receiver firmware must be modified [77].

G. PCBSat Thermal Control Subsystem

The thermal environment is one of the most challenging issues in spacecraft systems engineering. The vacuum of space introduces unique thermal control challenges, because the convective heat transfer with the air in the terrestrial environment mitigates nonspace system thermal problems. The spacecraft structure, in addition to ensuring that the satellite survives launch, can be purposely designed to ensure a tolerable thermal environment. At this stage of research, only a minimal TCS is implemented via battery and solar array temperature telemetry.

H. PCBSat Technology Assessment

Two revisions of PCBSat have been completed, with a promising prototype cost of $285 per node. Before the ionospheric-plasma-depletion study mission was considered, the design focus was on determining the maximum capabilities of a 100-g satellite. With a mission and payload now identified, the detailed mission requirements and architecture must now be developed in concert with a PCBSat flight model. Many of the space environment hazards that were neglected during exploratory prototyping must now be addressed and properly tested, such as debris, radiation, vacuum, atmospheric drag in LEO, free fall, and the launch environment. The mass is expected to rise to 200 g per node and the power budget will be balanced. The mission architecture will be a tradeoff of the scientific requirements, flight model capabilities, and constellation design, including a relay satellite. Another important parameter is the mission lifetime, because deploying large numbers of very small satellites must be properly considered to avoid space debris concerns. The cost in volume is expected to remain below $500 (half of SpaceChip). This cost is very attractive, because deploying 1000 PCBSats would cost $500,000 and launch costs of $1 million, excluding the relay satellite.

VI. Conclusions

Numerous envisioned distributed space missions with great benefit to society are awaiting technical solutions to be developed. Most of the academic excitement currently surrounds a few missions that require clusters of formation-flying satellites, with many challenging problems yet to solve. In contrast, there are several beneficial missions, such as the ionospheric-plasma-depletion reference mission, that rely on less-complex architectures based on traditional satellite constellations to achieve simultaneous multipoint sensing.

Existing subkilogram technologies were considered for such a mission, including traditional picosatellites and advanced micro-engineered aerospace systems. The standardized picosatellite approach (i.e., CubeSat) can most likely support a distributed science mission at relatively low cost. Microengineered aerospace systems, such as the co-orbiting satellite assistant, appear to be more focused on advanced technology development for specific purposes, instead of large-scale distributed missions.

Satellite-on-a-chip and satellite-on-a-PCB concepts, both currently in development, are presented as new very-low-cost alternatives. In the context of a distributed science mission, SpaceChip offers an incredibly low mass solution of 10 g, but several performance parameters limit its applicability to the simplest of missions. Work is currently underway to improve on the solar cell efficiency and communication range. Perhaps as system-on-a-chip technology matures, SpaceChip will become a viable option. PCBSat, with an estimated mass of 200 g and compatibility with existing deployment mechanisms, has shown to be a technically viable and cost-effective solution for the reference distributed science mission. However, more work is required to fully develop a system architecture and mission-ready flight model. Both SpaceChip and PCBSat have the potential to support a limited range of distributed space missions with at least an order of magnitude cost savings over other very-small-satellite approaches.

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C. McLaughlin
Associate Editor