Primordial Structure investigation (1'S1): A Low-Cost Space Mission to Image the Ir~tcr]]~ctliatc-Scale Cosmic Background Anisotropy

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ABSTRACT

Preliminary results are presented for the Primordial Structure Investigation mission study. PSI is a concept for a high-sensitivity, low-cost mission to image the Cosmic Microwave Background on intermediate angular-scales (0.5° - 10°). The mission objective is to measure the anisotropy of the CMB with sufficient sensitivity, and over a large enough solid angle, to unlock the wealth of evidence that the anisotropy holds for the origin of structure and the nature of the universe. PSI consists of an 80-cm aperture fed by 32 radiometers (eight at each of four frequencies), carried into heliocentric orbit on a spinning spacecraft. Total power radiometers, based on current HEMT (High Electron Mobility Transistor) and MMIC (Monolithic Microwave Integrated Circuit) technologies, provide low noise, stable gain, and low power dissipation to allow passive cooling. The spacecraft is constrained to fit the projected launch capabilities of a Taurus vehicle, has a mass of 213 kg, consumes 230 w, and is designed for a 5-year lifetime. The mission will image ~10^3 pixels to a sensitivity of 1 μK, or ~10^4 pixels to a sensitivity of 3 μK, at each frequency.

This study supports the feasibility of a low-cost space approach to the problem of determining the intermediate angular-scale anisotropies. Costing of our baseline design has not been completed, but we anticipate a mission within the guidelines of the anticipated NASA Mid-sized Explorer Program. However, further work is needed in the areas of HEMT development and characterization, optics design, and the multi-frequency strategy for the removal of galactic foreground emission to fully justify the present Concept.

INTRODUCTION

Following the discoveries made by the COsmic Background Explorer (COBE), the measurement of the it~tcrt~]c(li:i[c.-scalc (().5° to 10") Cosmic Microwave Background (CMB) structure has become perhaps the most important unrealized goal of observational cosmology. Knowledge of this structure relates directly, and uniquely, to fundamental and well-founded questions about the origin of the universe: the formation of present structure; actual parameters of the "Big Bang" standard model such as density and dark matter content; and fundamental physics in the very early universe such as that needed to support the inflationary paradigm. Measurement of the detailed nature and statistical proper-tics of the intermediate-scale structure will provide an essential test of theories about the origin of the universe and the nature of fundamental physics.

The PSI mission was conceived as a low- cost space approach to image the CMB on intermediate angular scales, and is supported as an internal study at the Jet Propulsion Laboratory. The goals of the mission are. 1) to determine the primeval spectrum of spatial structure from COBE scales (≥ 10°) down through the horizon scale (-10, to provide the complete and possibly definitive complement of evidence for the origin of structure and the necessary extensions to the standard models for physics and cosmology, and 2) to determine the properties of the causal turnover spectrum around 1" to determine fundamental properties of the universe, and to refine our observational link to the origin of present-day structure. The wealth and breadth of the.sc results and the statistical nature of the interpretations required to obtain them implies the need for extensive and precise measurements of anisotropics down to scales -0.5°. In particular, unambiguous images are needed of the CMB at this resolution, with high signal-to-noise ratio, that contain at least many hundreds of contiguous pixels.

THE 'S1 BASELINE MISSION

A baseline mission was developed as our initial estimate of the most cost-effective combination of mission, spacecraft, and instrument to carry out the desired imaging of the CMB. 'l"his baseline tests [he hypothesis that the imaging may be achieved using a low-cost launch vehicle, a small, spin-stabilized spacecraft with maximum heritage from existing spacecraft, and a simple instrument design. We believe that a detailed mission design is premature at (his time because of uncertainties that remain in receiver technology, optical design, and foreground removal strategy. We have made reasonable assumptions whet-c needed to arrive at the baseline design. 'l'he additional work needed to justify these assumptions is summarized in the final section of this paper. The costing of the baseline mission has not been completed, but is anticipated to be consistent with the expected NASA Mid-sized Explorer (Midex) limit of $60M.

Mission Overview

The parameters of the baseline mission are given in 'l'able 1. The spacecraft, shown in Figure 1, is placed well beyond the Earth-Moon system (we have chosen a heliocentric orbit for illustration), which minimizes systematic errors and consequently simplifies the experiment design. The launch vehicle was taken to be the Taurus for the purpose of this study because of its low projected launch cost vs. either the Delta 11 or Titan IIG - approximately $20M vs. $35-50M (Bayer et al., 1993). The size of the spacecraft is limited by the Taurus launch shroud, which we take as a primary constraint for the design of the spacecraft and its instrumentation. As shown in Table 11, the net spacecraft mass for this design comfortably allows an Earth-escape launch trajectory.

Table 1

Fig. 1

Table 1

The spacecraft carries a passively-cooled, multi-frequency microwave radiometer that is always shaded from the sun. Its -0.5° antenna beams are directed 2.5-5° off the spin axis of the spacecraft so that on snort time scales the radiometer views all points on a circle of ≤ 10° diameter centered on the spacecraft spin axis direction. Redirection of the spin axis two or three times a day over a period of approximately 3 months allows the imaging of a selected area about 100 square degrees in the sky. The spin axis is slewed approximately every 3 months so that a new area may be imaged and the radiometers may continue to view away from the sun.

Instrument

The instrument comprises an array of microwave receivers at the focus of a clear-aperture antenna. In the conceptual design, thirty-two radiometers (eight at each of four frequencies) are connected to an array of eight feedhorns in the focal plane. The 80-cm diameter offset-parabolic antenna provides a half-power beamwidth of 0.5° at approximately 60 GHz. A schematic of the radiometers connected to one feed horn is shown in Fig. 2. Each radiometer consists of a low noise HEMT amplifier passively cooled to approximately 100 K, followed by a MMIC amplifier to raise the signal to the level required for detection, A temperature-controlled noise diode is included for calibration. One radiometer channel, through the digitization electronics, is contained in a single, small physical unit that dissipates approximately 100 mw of power. "This estimate is based on existing components configured in an optimum design.

Fig

MMIC and HEMT technologies are rapidly developing in the frequency range extending as high as 140 GHz. The frequency range below 60 or 70 GHz, where dust mission can be ignored, is one of the ranges where the cosmic background structure measurements are most promising, so the match to the new technology is excellent. "Table III lists conservative performance parameters for the instrument, assuming a typical range of frequencies from 30 to 120 GHz. The exact choice of frequencies remains to be determined, but is of relatively minor impact on the spacecraft design.

Table I

Spacecraft

The spacecraft is a simple spinner with no articulating elements and three one-time deployable fixed solar arrays. Its principal heritage is from SAMPEX, a Small Explorer spacecraft built with similar reliability goals and approach and launched in 1993. We have carried out a preliminary design to a level that includes detailed specification of all the major subsystems listed in Table 11. 'This design assumes a five-year heliocentric mission, but could be adapted for stationkeeping at the L2 Lagrangian point with minor modifications.

The attitude control subsystem determines the position of the spinning, spacecraft in all three axes and triggers the thruster firing commands that re-position the spin axis. Spacecraft position is determined by off-the-shelf sun sensors and a star scanner with an accuracy of 0.02°, sufficient to meet a ground reconstruction requirement of ().10 for antenna pointing knowledge. A passive nutation damper, which is an energy-dissipating fluid mechanical device, is included to damp out the motion induced by the thrusters. The propulsion subsystem carries 25 kg of hydrazine to control the magnitude and direction of spacecraft spin for at least S years.

The command and data subsystem controls the spacecraft operation in its preprogrammed profile of attitude changes. It accepts engineering data from all on-board subsystems as well as instrument science data, stores these data and delivers them to the telecommunications subsystem for transmission to the ground as often as once a day. The instrument can be limited to data rates less than 100 bits per second by compacting the raw integrations using data averaging on scan pixels combined with 10 W-1CVC1 statistical analysis. Engineering data of up to 40 bits per second is allowed for. The most stringent processing requirement is to accept data from the attitude sensors and determine 3-axis attitude. This is estimated to require less than ().S million instructions per second (MIPS) computer processing capability. The data storage and processing requirements can be met by many available, computer systems. The one chosen for the baseline study has a processing capability greater than 1 MIPS, provides analog to digital conversion for the attitude sensors, and accepts solid state memory cards with 52 Mbits capacity each.

The telecommunications subsystem accepts real time and recorded data from the command and data subsystem, convolutionally encodes the data and provides modulated radio-frequency transmission to 34-m ground stations. It also accepts uplink commands and transfers them [o the command and data subsystem or other spacecraft subsystems. The baseline design system operates in S-band with an integrated transponder/command detector/telemetry modulation unit. The transponder output is amplified by a solid state power amplifier to 20 w and transmitted over either the low gain or medium gain antenna. A low gain antenna is located on one of the, deployable solar arrays and supports a downlink at 1200 bits per second without rc-orientation of the spacecraft during the first year of the mission. A medium gain antenna is included, which would be used by reorienting the spacecraft (o place the earth "within its ± 15" field of view. This mode will support the baseline data rate at the end of a 5-year heliocentric mission, when the spacecraft-earih distance is expected to increase to 0.5 AU.

The power subsystem supplies electrical power to all spacecraft equipment from just before launch through a.scent, separation and all in-flight operations. The most severe requirement is to support the 163 w peak load of the transponder and power amplifier during data transmission to the ground. A battery is included to provide power during launch and a.scent until the shroud is separated and the solar arrays are deployed. The three solar arrays use gallium arsenide on germanium solar cells on both sides, and arc sized and oriented so that the arrays receive enough illumination to fully power the spacecraft, including science instrument and downlink, for all sun orientations and spin rates. The large power margin results from making the solar arrays as big as possible while still fitting in the launch vehicle. Their size could be reduced by almost half and still provide comfortable margin. Alternatively, the power could be used with a more powerful transmitter to send more downlink data,

The thermal subsystem provides temperature control of all spacecraft bus electronics assemblies and allows for (he cooling of the microwave radiometer cold plate and antennas. The instrument is mounted on a cold plate that is thermally isolated from the spacecraft bus, and a simple sun shade shields the inside of the microwave radiometer from solar illumination. Thermal isolation is provided by fiberglass tube structural supports, low conductance cables and multi layer insulation. The elements of the science instrument see only cold space. All cold elements have high emissivity finishes, and the 3.2 w dissipated within the instrument leads to an instrument temperature < 100K. Temperature control of the. spacecraft electronics is provided primarily by surface finishes, insulation, heaters and louvers on the spacecraft bus.

NEED FOR FURTHER WORK

The PSI baseline mission supports the contention that the intermediate-scale imaging of the CMB can be accomplished with a small space mission. However, work is needed in several areas to produce a fully credible experiment design.

Mission Optimization

Trade-off studies may lead to an improvement over the baseline design, For example, the choice of a heliocentric orbit over the L2 option simplifies the spacecraft by eliminating the need for a propulsion system for midcourse correction and stationkeeping, but places an extra burden on the communication system. We believe both options are viable, but further study may show the 1.2 option to be more desirable.

The choice of passive cooling for the radiometer front ends leads to the ultimate. simplicity in instrument design, However, a significant improvement in sensitivity can be obtained by reducing HEMT temperatures from the ~100 K possible with radiative cooling to much lower temperat ures by using active coolers. '1 'he trade-off between instrument complexity using space cryocoolers relative to the improved performance remains to be studied. Continuing improvements in cryocoolers may make this possibility more attractive.

HEMT Development

The current generation of low-noise, cooled HEMTs is based on ambient-temperature transistor designs whose. performance improves as their temperature is reduced. A modest effort to design transistors specifically optimized for low temperature performance can lead to perhaps a factor of two improvement over the performance estimates given in 'J'able 111. Futher effort is needed in any case to understand and improve the stability of HEMT amplifiers. The PSI sensitivity is compromised by amplifier instabilities if the random component of signal drift in one spacecraft rotation period approaches the same magnitude as the sing] c-scan pixel measurement uncertainty due to radiometer noise. For the PSI parameters (his implies a requirement for HEMT gain stability $\Delta G/G \leq 10^{-5}$ in 10 seconds. A measurement of an existing HEMT amplifier has indicated the presence of intrinsic instabilities $\Delta G/G \sim 10^{-4}$ in 10 seconds (Jarosik et al., 1993), which is unacceptable. The necessai y stability must be demonstrated to justify the simplicity of the. present total-power approach.

Frequency Selection

Measurements at multiple frequencies are necessary to allow the separation of the structure due to galactic foreground components from that of the CMB, taking advantage of their different frequency dependences. This separation has been accomplished for the large-sca]c measurements (Bennett et al., 1992), but has not been thoroughly explored for smaller scales. Brandt et al. (1994) argue from the most quantitative analysis done to elate that a definitive, pixel by pixel separation may require 1 μK accuracy at four or more frequencies, which wc use as a conservative criterion for our study. The optimum choice of frequencies requires further efforts in modeling, the availability of improved maps of low-level galactic signals (particularly in the microwave region), and mission trade studies that realistically take into account the frequency dependence of various performance factors such as sensitivity, bandwidth, and spatial resolution.

Optics

The objective of the optical design is to support a maximum number of focal-plane feeds while maintaining low sidelobes for all beams on the sky. Wc have assumed that this may be accomplished with a clear aperture, dual shaped-reflector approach using multi-frequency, dua]-polarization feeds, and have used a successful dual-beam antenna design as a model for sizing our optical system (Cha, 1983). To justify the PSI concept, however, it is necessary to carry out a detailed design complete with beamshape and sidelobe calculations that enable the modeling of systematic errors due to beamshape differences and off-axis galactic sources.

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TABLES

Table I

Primordial Structure investigation

<u>General Characteristics</u>

Launch Vehicle:	Taurus
Mission lifetime	5 yr
Orbit:	Heliocentric (L2 possible with small modifications)
Spacecraft Heritage:	SAMPEX "
Total Mass:	213 kg
Total Power:	230 w
Attitude Control:	Spin-stabilized at 6 rpm
	Optical tracking and pointing reconstruction
	Spin redirection thrusters /passive nutation damping
	<u>Instrument</u>
Radiometer Characteristics:	Total power, direct detection
	HEMT preamp followed by MMIC receiver
	0.1 w power consumption/channel
	Radiatively cooled to< 100 K
	4 frequencies to be chosen in range 30-120 GHz
	8 radiometer channels at each frequency
Optical Design:	Single 80-cm off.set paraboloid main antenna
	Focal plane array of dual-polarization, multi-frequency feeds
	Typical beam axis offset 2.5”-5° from spacecraft spin axis

Table 11

Spacecraft Mass and Power Summary

Subsystem	Mass	Power
Antenna	32 kg	0 w
Microwave instrument	17	9
Command & Data	9	15
Attitude Control	8	8
Telecommunications (transmit)	13	163
Power	48	10
Propulsion (incl. 25 kg propellant)	38	5
Struc., Mech., Thermal	43	20
Launch	5	Q
System Total	213 kg	230 w
Launch Vehicle Capability	280 kg	
Solar Array Capability		543 w

Table 1 [I

Radiometer Performance Estimates*

f [GHz]	B [GHz]	HPB w [deg]	T_s [K]	No. pixels measured to 3 μK in5 yr	No. pixels measured to 1 μK in 5 yr
30	5	1.1	43	30300	3360
60	7.5	0.6	80	13100	1460
90	10	0.4	110	9300	1030
120	12	0.3	140	6900	760

* The frequencies given in the 'l'able are chosen to illustrate llliMT-based radiometer performance, and are not optimum for the subtraction of galactic foregrounds.

FIGURE CAPTIONS

Figure 1: The PSI Spacecraft

Figure 2: Schematic of one radiometer/feed combination and a typical radiometer block diagram. A set of four such identical radimnc(cr/feed combinations provides eight channels at each of two frequencies, and a frequency-scaled set (i.e., eight feedhorns total) provides the balance of channels and frequencies.

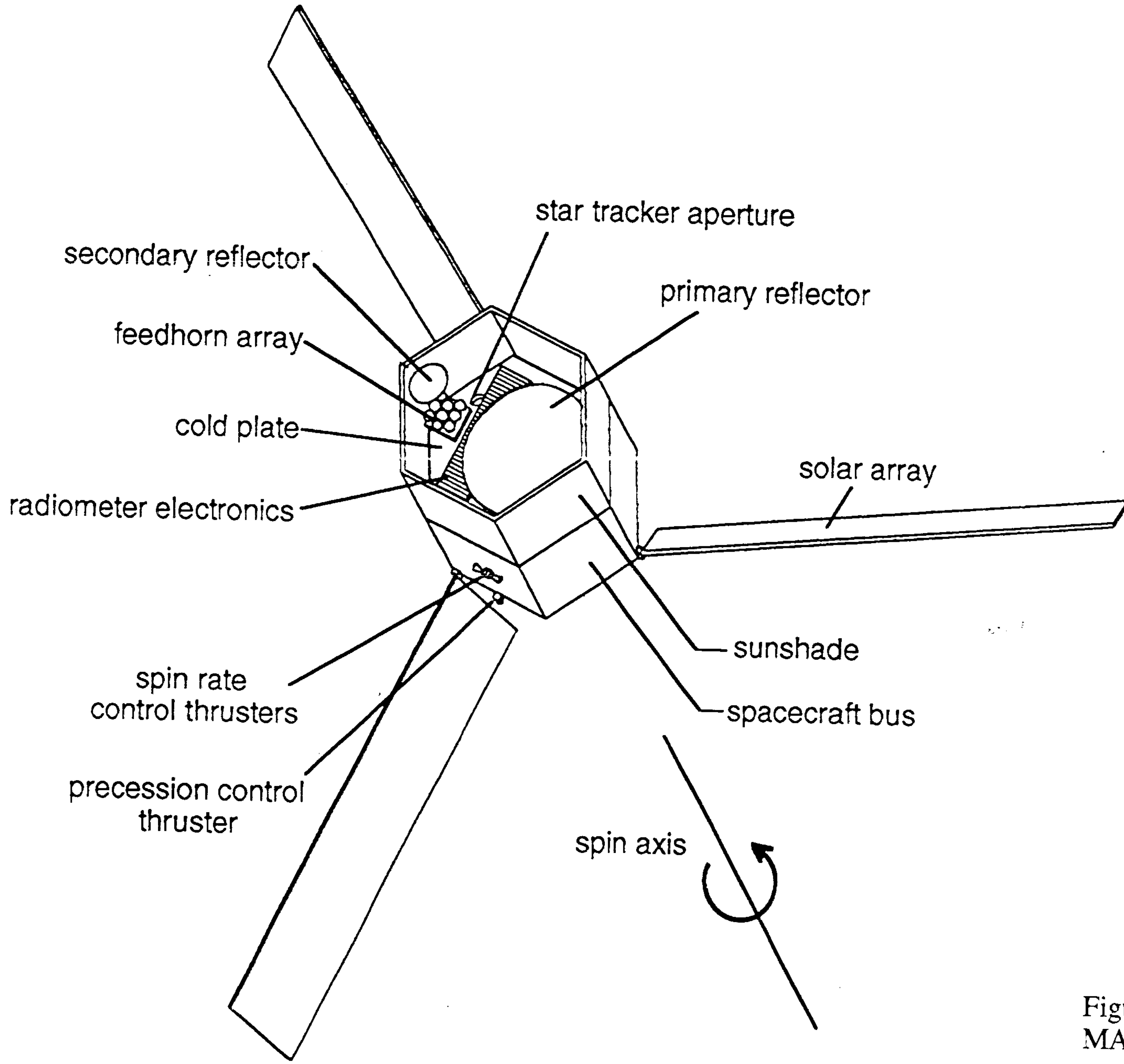
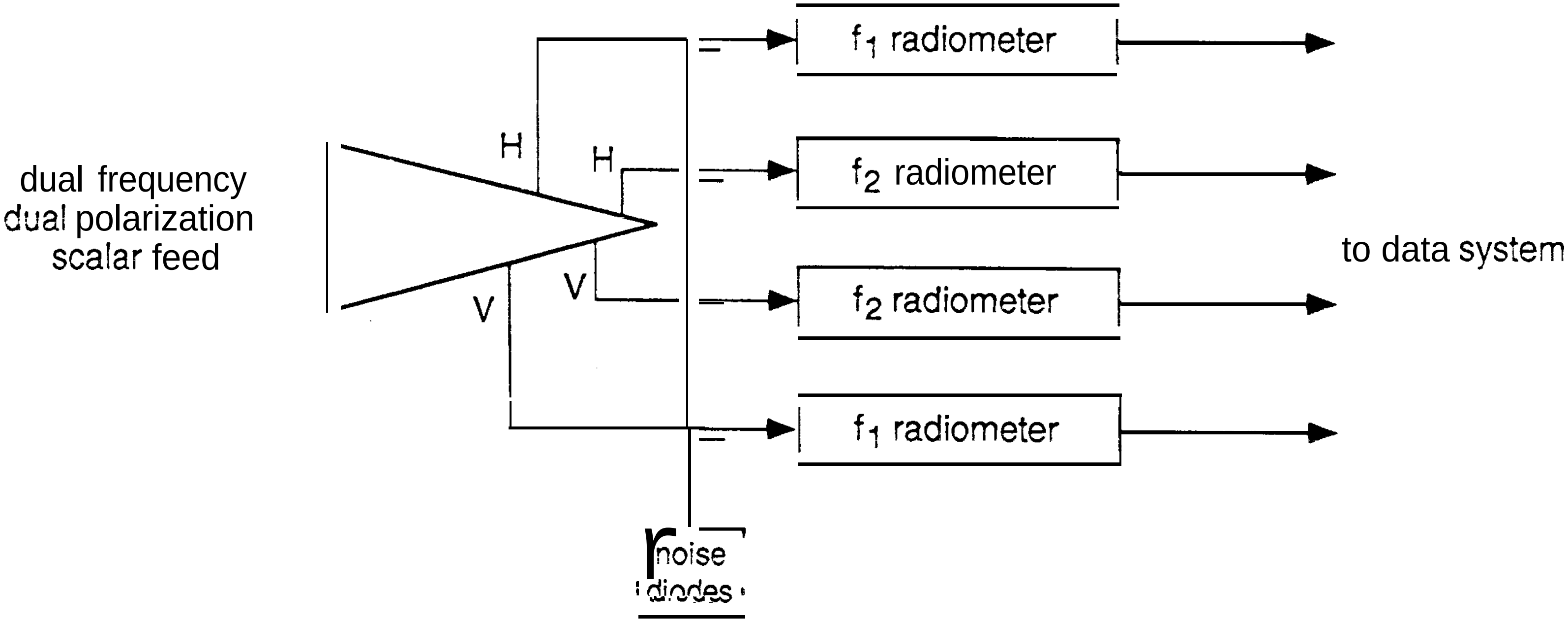
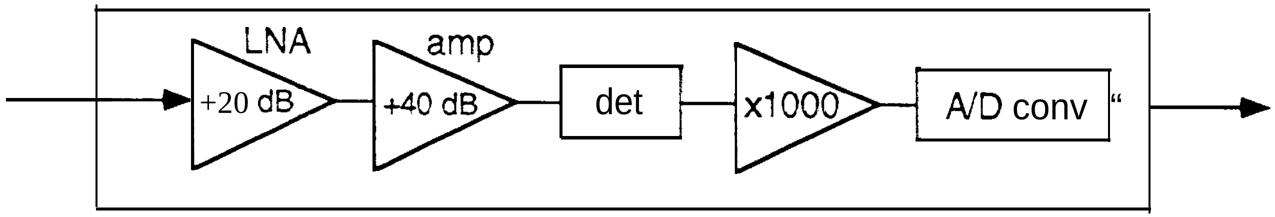


Figure 1
MAJ 12/30/93

Radiometer/Feed Combination



Radiometer Block Diagram



Power = 100 mW

Figure 2
MAJ 12/30/93