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SCD1 THERMAL DESIGN AND TEST RESULT ANALYSIS

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ABSTRACT

The SCD1 (Satelite de Coleta de Dados 1) is a spin stabilized low-Earth orbit satellite dedicated to the collection and distribution of environmental data. It was completely developed at the Brazilian Institute for Space Research (INPE) and is scheduled to be launched in 1992.

The SCD1 passive thermal control design configuration is presented and the thermal analysis results are compared with the temperatures obtained from a Thermal Balance Test. The correlation between the analytical and experimental results is considered very good.

Numerical flight simulations show that the thermal control design can keep all the subsystem temperatures within their specified temperature range.

INTRODUCTION

The Data Collecting Satellite 1 (SCD1), is the first satellite completely designed and built in Brazil. It is part of an ambitious program that intends to enable Brazilian scientists and industry to develop and fabricate launchers, and more suitable satellites for Brazilian needs. SCD1 is an experimental satellite and its mission is to collect meteorological data sent from automatic ground stations, and relay them to a central ground station. As of now the satellite is being assembled, and it is scheduled to be launched in 1992.

The SCD1 is a small satellite weighing 115 kg, its shape is a right octagonal prism, whose base fits in a 1 m diameter circle and is 67 cm high.

Its specified orbit is circular at 750 km altitude, inclined 25 degrees with respect to the equator.

The electric power (around 80w) is supplied by nine solar panels installed on the eight lateral panels and on the upper octagonal one. The equipment are located on the lower (internal face) central (both faces) and upper (internal face) panels.

The satellite is spin stabilized and its attitude control is such that during its life time (1 year) the sun rays will never strike the lower panel.

Three critical cases were selected for thermal control design purposes: two cold cases and one hot case.

A passive thermal design concept was developed and the equipment operating temperature ranges were obtained through a mathematical model and validated by a Thermal Balance Test (TBT).

SATELLITE DESCRIPTION

The SCD1 primary structure consists of a central tube, two octagonal panels, and four stiffening bars. Eight rectangular solar panels and one octagonal top, close the two enclosures where the equipment are installed. Figure 1 shows an exploded view of the spacecraft with all the electronic subsystems.

The electronic boxes are installed on the internal face of lower panel, both faces of central panel and inner face of upper panel. On the lower panel there are four electronic boxes (Battery, PCU, Magnetometer Electronics, and Magnetometer Sensor), and two circular holes where the two Shunt Dissipators are fixed. On the lower face of the central panel, seven electronic boxes are settled (Decoder, PDU, DC/DC Converter, Sun Sensor £ 1 and £ 2, DCP Transponder, and UPC). The last six boxes (TMTC Transponders £ 1 and £ 2, Encoder, Diplex £ 1 and £ 2, and UPD/C) are installed on the central panel upper face. A Nutation Damper and a Torque Coil are fixed on the inner face of the upper panel.

DESIGN CONSTRAINTS AND REQUIREMENTS

The complete mission is divided in six phases: Count-Down; Launch; Acquisition; Acceptance; Routine; Attitude Change.

The Count-Down phase is scheduled to last 5 days and consists of the final system test before launch. During this time the spacecraft will be powered via umbilical connector.

The Launch phase lasts 11 minutes corresponding to the time period between lift-off and separation. At 100 seconds from lift-off the aerodynamic fairing will be jettisoned and the satellite exposed to solar radiation and aerodynamic heating.

The acquisition phase involves the orbit and attitude determination. The orbit determination lasts 11h and 40 min (7 complete orbits) while the attitude determination lasts 38 hours more.

The acceptance phase starts just after the attitude acquisition phase and lasts one month. During this period in-orbit tests are effected and the collected data quality verified.

The routine phase, in which the SCD1 will fulfill its data collecting mission, lasts 11 months.

The attitude change phase occurs whenever the satellite attitude is out of specification. Attitude manoeuvres are accomplished by telecommand activation of an on board magnetic torque coil.

The combination of the orbit and attitude parameters during SCD1 lifetime, creates some severe thermal constraints to its operation.

Its relatively low altitude (750 km), makes the Earth and Albedo radiation effects significant when compared to the direct Sun radiation. The orbit inclination (25 deg) imposes an angle variation between sun vector and orbit plane from 1.5 to 48.5 degrees. This variation causes an eclipse period variation from 35 to 29 minutes for an orbit period of 100 minutes.

The attitude control obtained by spin stabilization and nutation damper action, and manoeuvres done by the torque coil imposes a solar aspect angle variation from 0 to 90 deg which means an absorbed heat load variation of about 37% during one year.

In general, the electronic equipment present low dissipation values, and the total power dissipated by the spacecraft (not including the Shunt Dissipators) varies from 22.9 W to 37.6 W according to the mission phase. Table 1 shows the equipment average dissipation values for minimum and maximum total heat dissipation operating modes, and the ones for normal operating mode. Shunt Dissipators are the most dissipative subsystem, which average power can vary from 2.2 W to 11.7 W with a peak of 47 W.

The operating temperature range specified for each equipment is shown in Table 2. Two equipment are considered critical: the Battery that has the narrower temperature range (-5 to 25 C) and the DCP Transponder, whose lower temperature limit is the highest (0 C).

The thermal control design shall keep all the subsystem temperatures within the specified range for all the satellite operating modes during the routine phase, and it shall warrant the subsystem integrities during non operating modes and other mission phases.

The thermal control subsystem shall be based only on passive techniques.

THERMAL CONTROL DESIGN

Due to the satellite orbit and attitude, one can select three critical cases for the thermal control design. The first one, so called AGEOMIN, corresponds to the situation in which the angle between the sun vector and the orbit plane, and the angle between the sun vector and the spin axis, are simultaneously zero. The second one, so called LATMIN, corresponds to the case where the angle between the sun vector and the orbit plane is zero, and the angle between the sun the vector and the spin axis is equal 90 degrees. The third one, so called AGEOMAX, corresponds to the case where the angle between the sun vector and the orbit plane is maximum (48.5 deg), and the angle between the sun vector and the spin axis is zero.

In the AGEOMIN case, the sun rays strike perpendicularly the upper panel, and the eclipse period is maximum. It causes simultaneously the minimum temperature on the lower panel, and the maximum temperature on the upper panel. Therefore, the largest temperature gradient through the satellite structure occurs in this case.

In the LATMIN case the sun vector is perpendicular to the spin axis, and the eclipse period is maximum. This case corresponds to the minimum spacecraft heat load, and the minimum temperature of the central panel.

In the AGEOMAX case the maximum heat load occurs, bringing all the electronic equipment of central and lower panels to their maximum flight temperatures.

These conditions impose three different main heat path throughout the satellite structure. The first one occurs in the AGEOMIN case when the heat flux comes from the upper panel, mainly by radiation. Part of the heat is reradiated by the lateral panels and the rest reaches the lower panel. The second one occurs in the LATMIN case, when the heat flux coming from the lateral panels heats nearly uniformly all the equipment, and is reradiated by all the satellite external surfaces. In the AGEOMAX case, which presents an intermediate situation, the heat flux comes from the lateral and the upper panels.

Based on this observations, the thermal control strategy was defined. The three main points of this strategy are:

- since in the AGEOMIN case, the lower panel temperature tends to be low, it is necessary to minimize the heat losses through the lateral and lower panels,

- in the LATMIN case the central panel equipment temperatures tend to be low. It is necessary to warrant that enough heat flux from the lateral panels reaches this panel, keeping it above the minimum specified temperature levels,

- in the AGEOMAX case the whole satellite temperature becomes high. It is necessary to minimize the absorbed heat loads to decrease the temperature level.

It can be seen that some points of the thermal control strategy are in conflict with each other, thus suggesting that a compromise solution is necessary.

To accomplish the strategy described above, the following measures are adopted:

- the lower panel external surface is covered with a low emissivity and low solar absortivity coating;

- the available areas for thermal control on the upper panel are covered with high emissivity and low solar absortivity coating;

- the available areas for thermal control on the lateral panels are covered by low emissivity and low solar absortivity coating;

- the inner surfaces of the lateral panels, in the region between the lower and central panels, are also covered with low emissivity coating;

- the lateral panels are insulated from the octagonal panels by fiberglass washers;

- the Battery is completely covered by a low emissivity coating, its base is bonded to the lower panel, and the external satellite surface underneath the battery is covered with a mosaic to adjust its temperature level;

- the DCP Transponder is completely covered with a low emissivity coating, and it is conductively insulated from the central panel by means of fiberglass washers;

- PCU, DC/DC Converter, and TMTC Transponders £1 and £2, are thermally bonded to their panels;

- the upper face of the Encoder box is covered with low emissivity coating;

- the upper face TMTC Transponder £1 and £2, and their lateral faces that see the lateral panels are covered with low emissivity coating;

- the Shunt Dissipator is insulated from the lower panel by stainless steel washers, and its external surface is covered with high emissivity and low solar absortivity coating;

- the rest of the satellite inner surface is covered with high emissivity coating.

THERMAL ANALYSIS

To define the thermo-optical properties and to verify the performance of the thermal control design, a 111 node finite differences model was developed. Figures 2 to 5 show the nodal breakdown. This model was generated and solved by a computer program developed at INPE, based on the TMG and SINDA programs.

To perform the thermal analysis the minimum internal heat dissipation mode was associated with the two cold cases (AGEOMIN and LATMIN), and the maximum internal heat dissipation was associated with the hot case (AGEOMAX). After some iterations a suitable thermal control design was achieved. Figure 6 shows the resulting thermal control configuration.

THERMAL BALANCE TEST

In order to validate the mathematical thermal model and to qualify the thermal control design concept, a full scale Thermal Model (TM) was built and underwent to a Thermal Balance Test (TBT).

The TBT was performed in a 3m x 3m Thermal-Vacuum Chamber. The space environment was simulated by a black shroud cooled by LN2. The heat loads were simulated by skin heaters fixed on all the TM external surfaces and inside the electronic box mock-ups. The vacuum level was below 1E-5 Torr.

The complete TBT was divided in five phases: two static and four dynamic. It started with the STATIC AGEOMAX case, followed by DYNAMIC AGEOMAX, DYNAMIC LATMIN, DYNAMIC AGEOMIN, and finally STATIC AGEOMIN. It lasts nearly 62 hours and was accomplished successfully.

By comparison of experimental and the predicted temperatures, it was identified discrepancies caused by either out of specification conditions in the TBT, or by problems in the mathematical model. The most important discrepancies were:

- the actual heat loads were 5 to 10% lower than the specified ones,

- the emissivity of the central tube extension was above its normal condition due to the application of unspecified tape over it, to hold heater wiring,

- the heat capacitance of the lateral and upper panels were underestimated,

- joint conductances were underestimated,

After eliminating these discrepancies, the mathematical thermal model was adjusted, and a new temperature distribution was calculated. Table 3 shows the average differences between the predicted and the measured temperatures, and the standard deviation before and after the model adjustment. Figures 7 to 11 show the equipment temperature difference histograms for the adjusted model.

FLIGHT PREDICTIONS

After the mathematical thermal model adjustment, all the critical flight conditions were simulated to verify the thermal control subsystem performance. Figure 12 shows the predicted in flight normal operating equipment temperature ranges versus the specified ones.

CONCLUSION

TBT results showed a good correlation with the analytical predictions. The flight simulations showed that the equipment temperatures for the normal satellite operating mode will be kept in their specified range. For the other mission phases and operating modes it was verified that the equipment temperatures never exceed their acceptance limits.

TABLE 1.- AVERAGE EQUIPMENT HEAT DISSIPATION

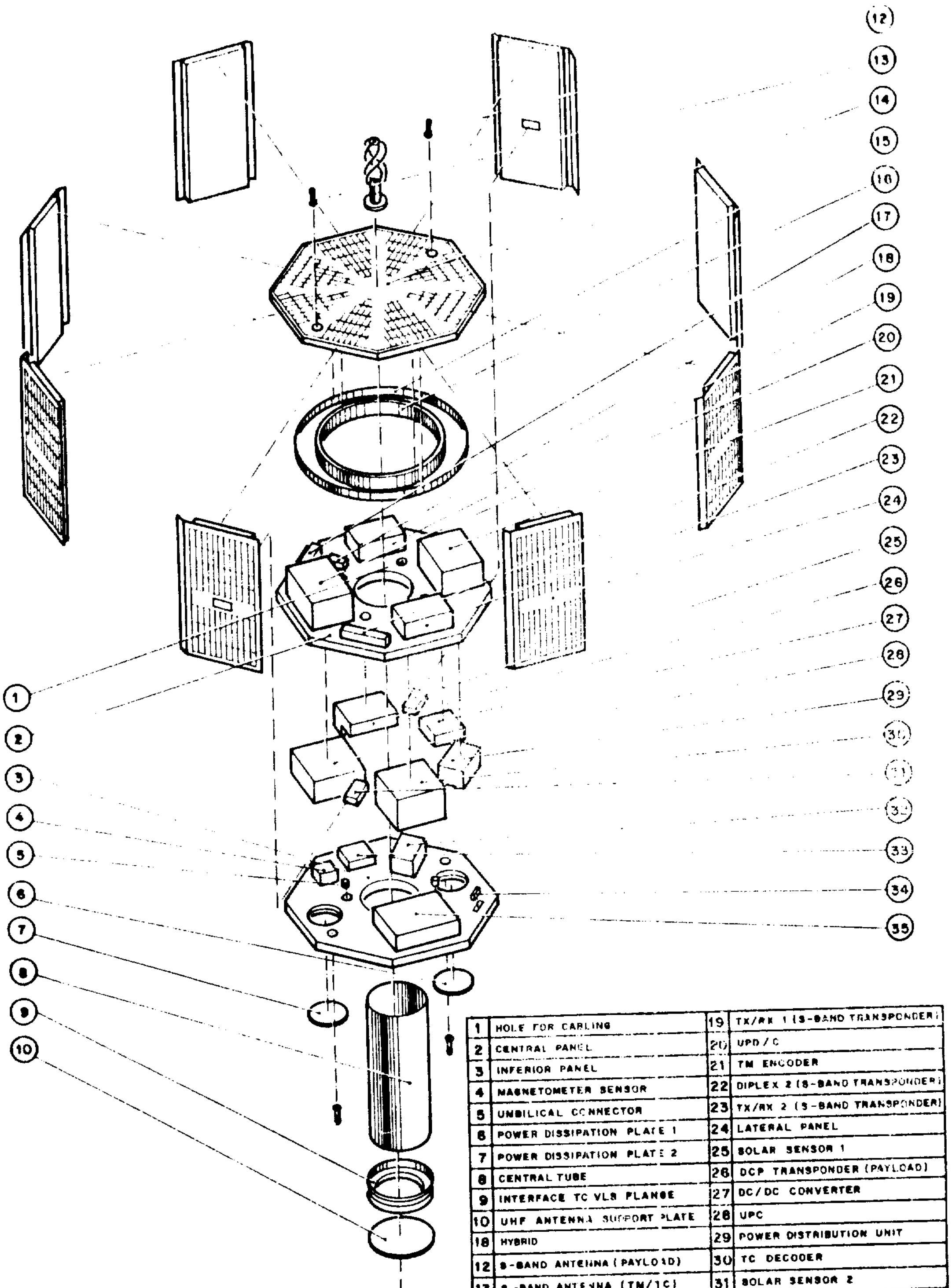
EQUIPMENT	Qmin (W)	Qmax (W)	Qnormal (W)
BATTERY	1.1	2.2	1.4
MAGNET. ELECTRONICS	0	.4	.10
PCU	4.9	8.0	5.9
DC/DC CONVERTER	3.9	6.9	4.3
SOLAR SENSOR	0	.2	.10
DCP TRANSPONDER	0	.42	.42
TC DECODER	5.5	5.5	5.5
UPC	0	2.5	2.5
UPD/C	0	3.5	0
TM ENCODER	0	0	.28
TR/TX 2	3.75	3.75	3.8
TR/TX 1	3.75	4.26	4.3
TORQUE COIL	0	0	0

TABLE 2.- OPERATING TEMPERATURE RANGE

EQUIPMENT	Tmin (C)	Tmax (C)	START UP (C)
TOP SOLAR CELLS	-60	80	-80
LATERAL SOLAR CELLS	-60	45	-80
BATTERY	-5	25	-15
MAGNET. SENSOR	-20	60	-30
MAGNET. ELECTRONICS	-20	60	-30
PCU	-10	50	-20
SHUNT DISSIPATOR	-10	60	-20
DC/DC CONVERTER	-10	50	-20
SOLAR SENSOR	-30	60	-40
SOLAR SENSOR ELETR.	-20	60	-30
DCP TRANSPONDER	0	40	-10
TC DECODER	-10	40	-20
UPC	-10	40	-20
UPD/C	-10	40	-20
TM ENCODER	-10	40	-20
TR/TX	-10	40	-20
DIPLEXER	-10	40	-20
HYBRID	-40	100	-40
PDU	-10	50	-20
NUTATION DAMPER	-50	75	-60
TORQUE COIL	-60	75	-70
DIODE PLATE	-40	60	-50

TABLE 3.- AVERAGE TEMPERATURE DIFFERENCES BEFORE AND AFTER MODEL ADJUSTMENT

PHASE	BEFORE ADJUSTMENT		AFTER ADJUSTMENT	
	Tmeas-Tpred (C)	(C)	Tmeas-Tpred (C)	(C)
STATIC AGEOMAX	-4.0	2.5	-.30	1.6
DYNAMIC AGEOMAX	-4,3	2.5	-.90	1.9
DYNAMIC LATMIN	-3,8	4.8	-.10	2.4
DYNAMIC AGEOMIN	-4.1	3.7	-1.3	2.7
STATIC AGEOMIN	-2.8	3.4	.50	2.2



1	HOLE FOR CABLING	19	TX/RX 1 (S-BAND TRANSPONDER)
2	CENTRAL PANEL	20	UPD / C
3	INFERIOR PANEL	21	TM ENCODER
4	MAGNETOMETER SENSOR	22	DIPLEX 2 (S-BAND TRANSPONDER)
5	UMBILICAL CONNECTOR	23	TX/RX 2 (S-BAND TRANSPONDER)
6	POWER DISSIPATION PLATE 1	24	LATERAL PANEL
7	POWER DISSIPATION PLATE 2	25	SOLAR SENSOR 1
8	CENTRAL TUBE	26	DCP TRANSPONDER (PAYLOAD)
9	INTERFACE TC VLS FLANGE	27	DC/DC CONVERTER
10	UHF ANTENNA SUPPORT PLATE	28	UPC
18	HYBRID	29	POWER DISTRIBUTION UNIT
12	S-BAND ANTENNA (PAYLOAD)	30	TC DECODER
13	S-BAND ANTENNA (TM/TC)	31	SOLAR SENSOR 2
14	SUPERIOR PANEL	32	BATTERY
15	TORQUE COIL	33	MAGNETOMETER ELECTRONICS
16	NUTATION DAMPER	34	INTEGRATION CONNECTOR
17	DIPLEX 1 (S-BAND TRANSPONDER)	35	POWER CONDITIONING UNIT

FIGURE 1.- SCD1 EXPLODED VIEW

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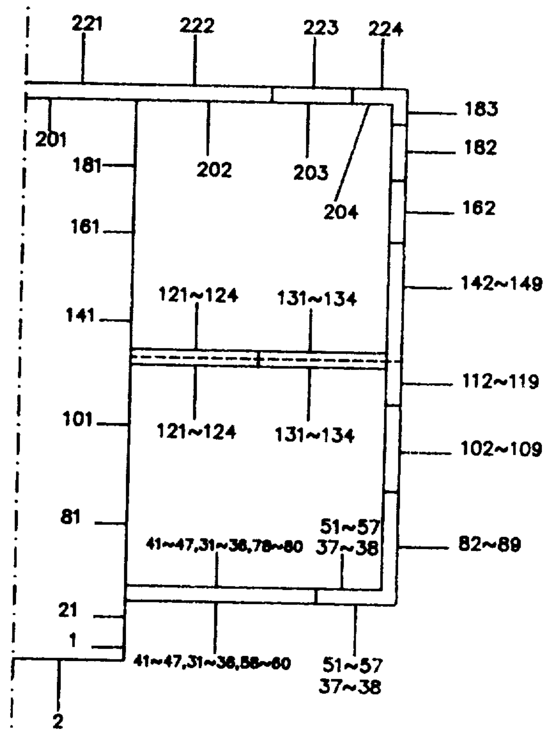

FIGURE 2.- SCD1 NODAL BREAK DOWN : CROSS SECTION

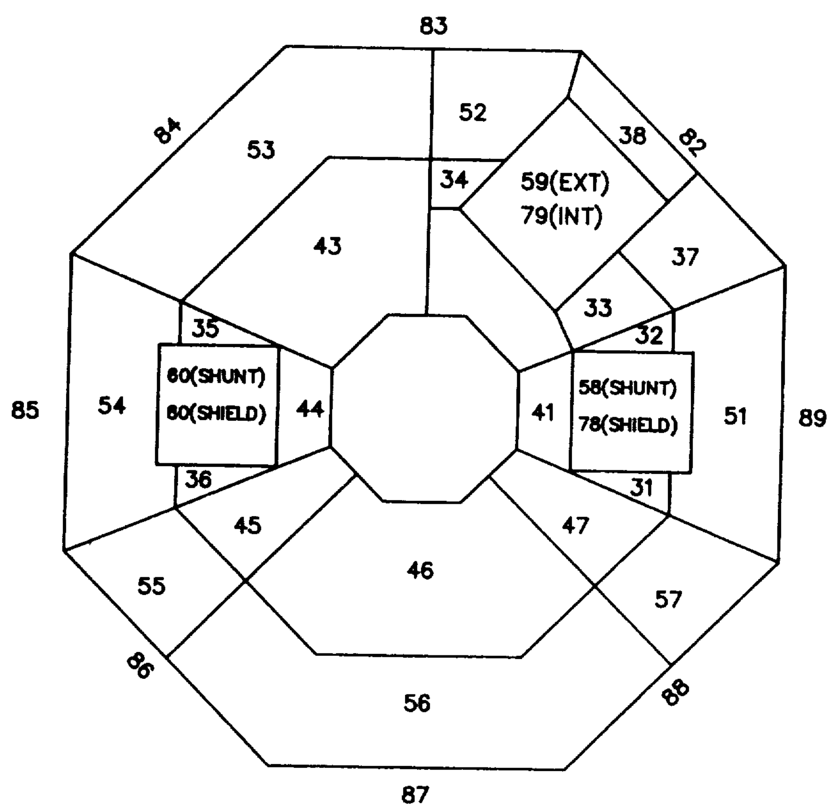

FIGURE 3.- SCD1 NODAL BREAK DOWN : LOWER PANEL

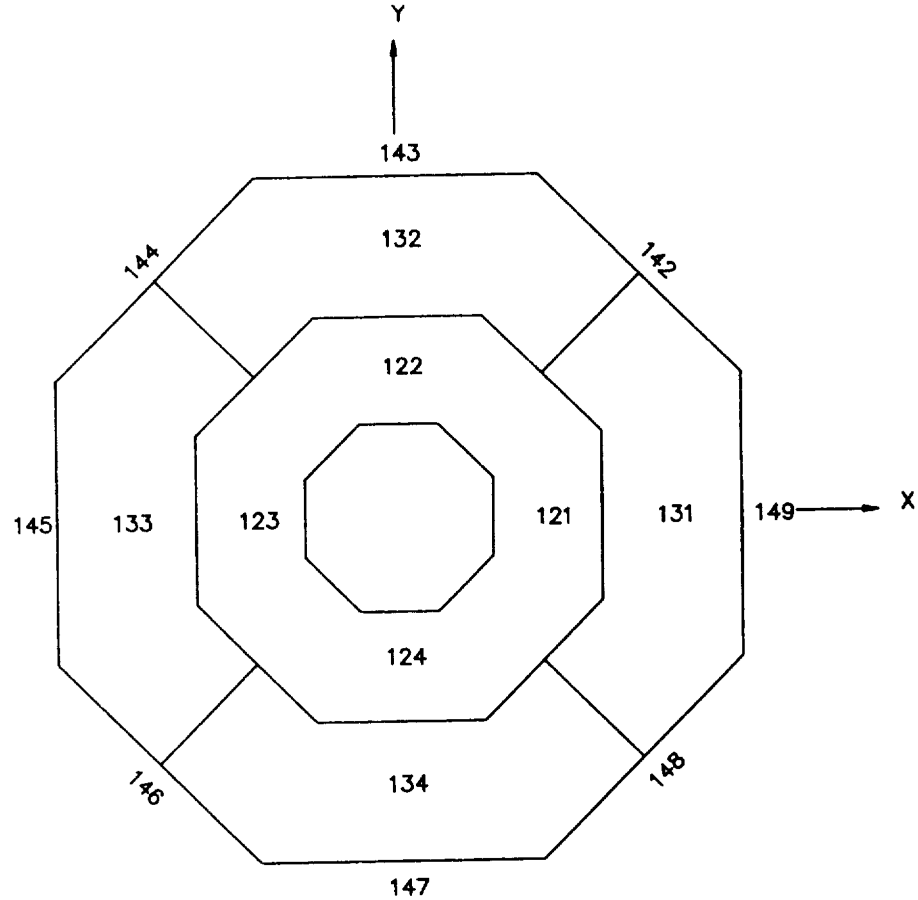

FIGURE 4.- SCD1 NODAL BREAK DOWN : CENTRAL PANEL

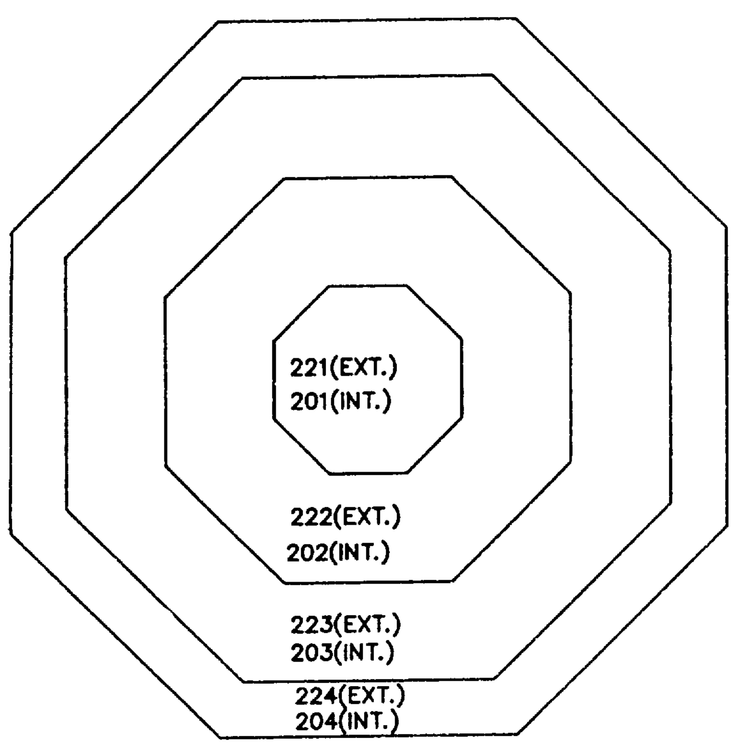

FIGURE 5.- SCD1 NODAL BREAK DOWN : UPPER PANEL

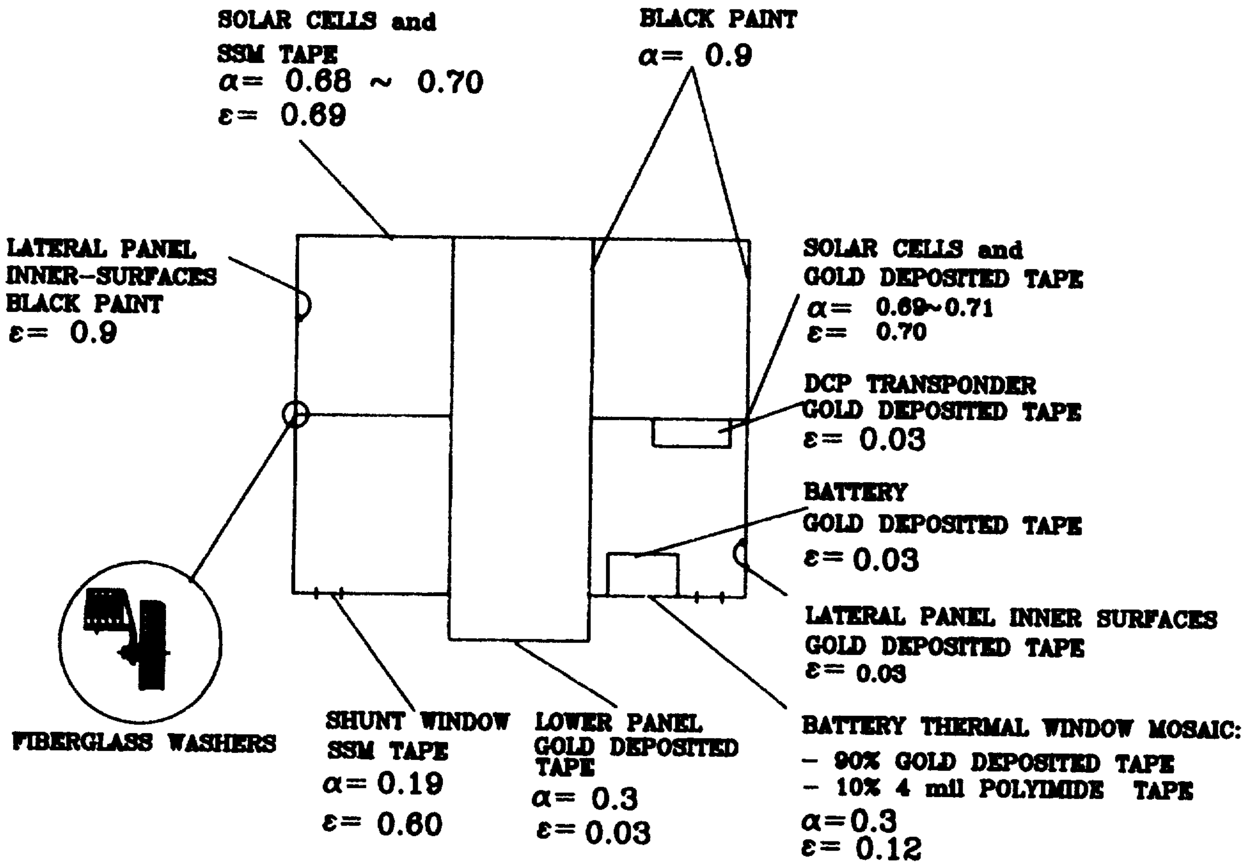

FIGURE 6.- SCD1 THERMAL CONTROL SUBSYSTEM DESCRIPTION

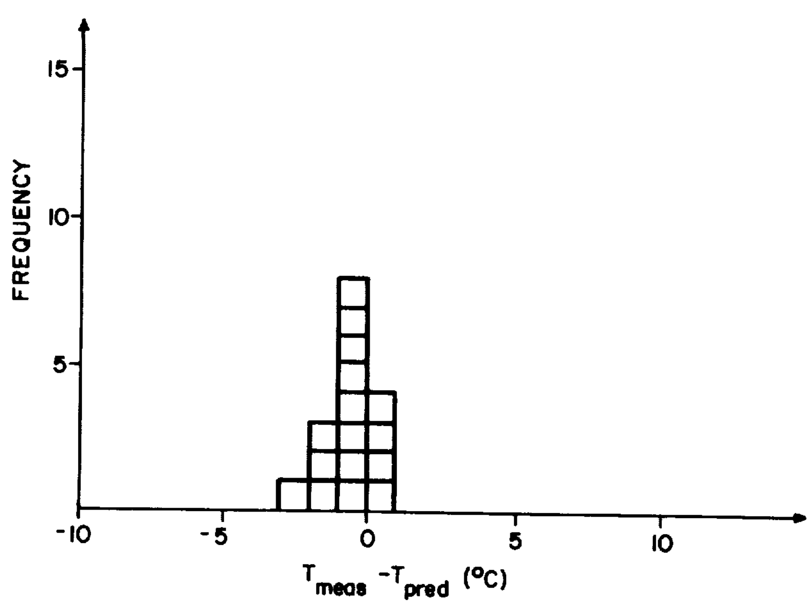

FIGURE 7.- MEASURED-PREDICTED TEMPERATURES (STA. AGEOMAX)

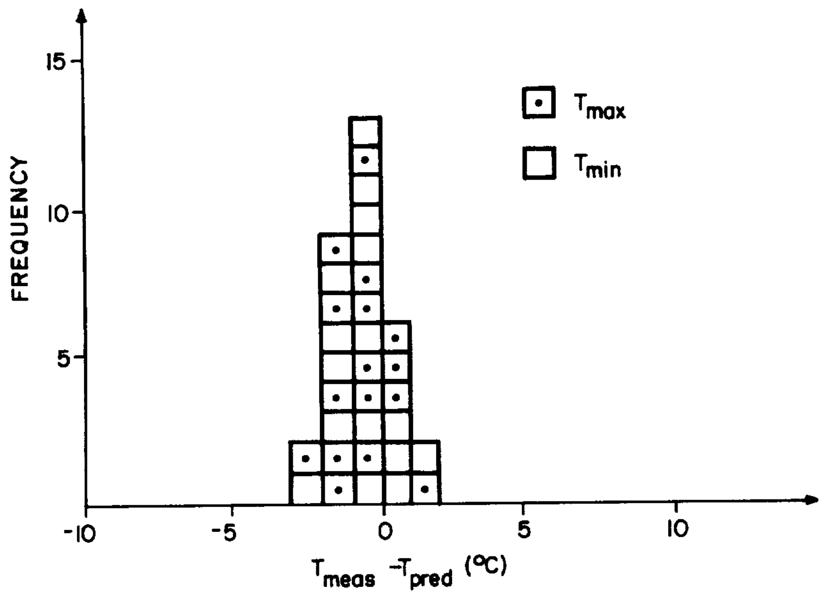

FIGURE 8.- MEASURED-PREDICTED TEMPERATURES (DYN. AGEOMAX)

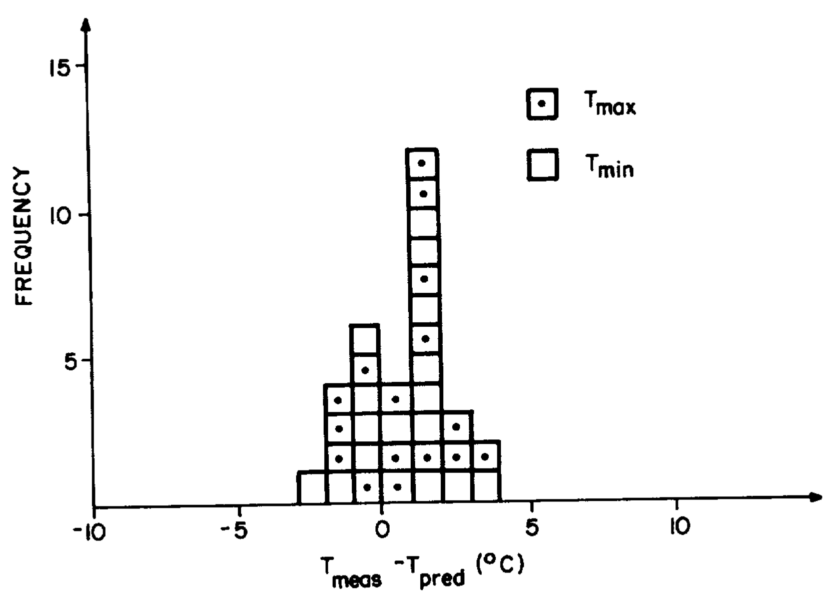

FIGURE 9.- MEASURED-PREDICTED TEMPERATURES (DYN. LATMIN)

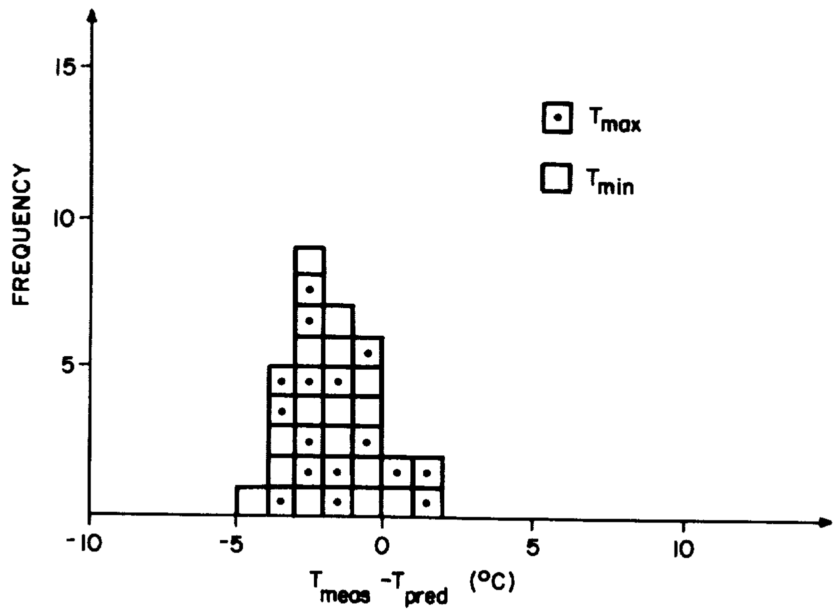

FIGURE 10.- MEASURED-PREDICTED TEMPERATURES (DYN. AGEOMIN)

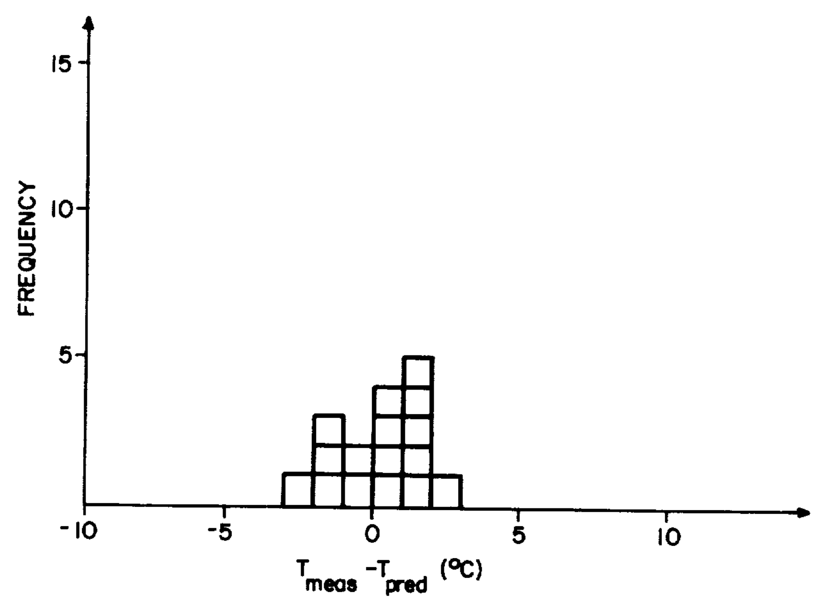

FIGURE 11.- MEASURED-PREDICTED TEMPERATURES (STA. AGEOMIN)

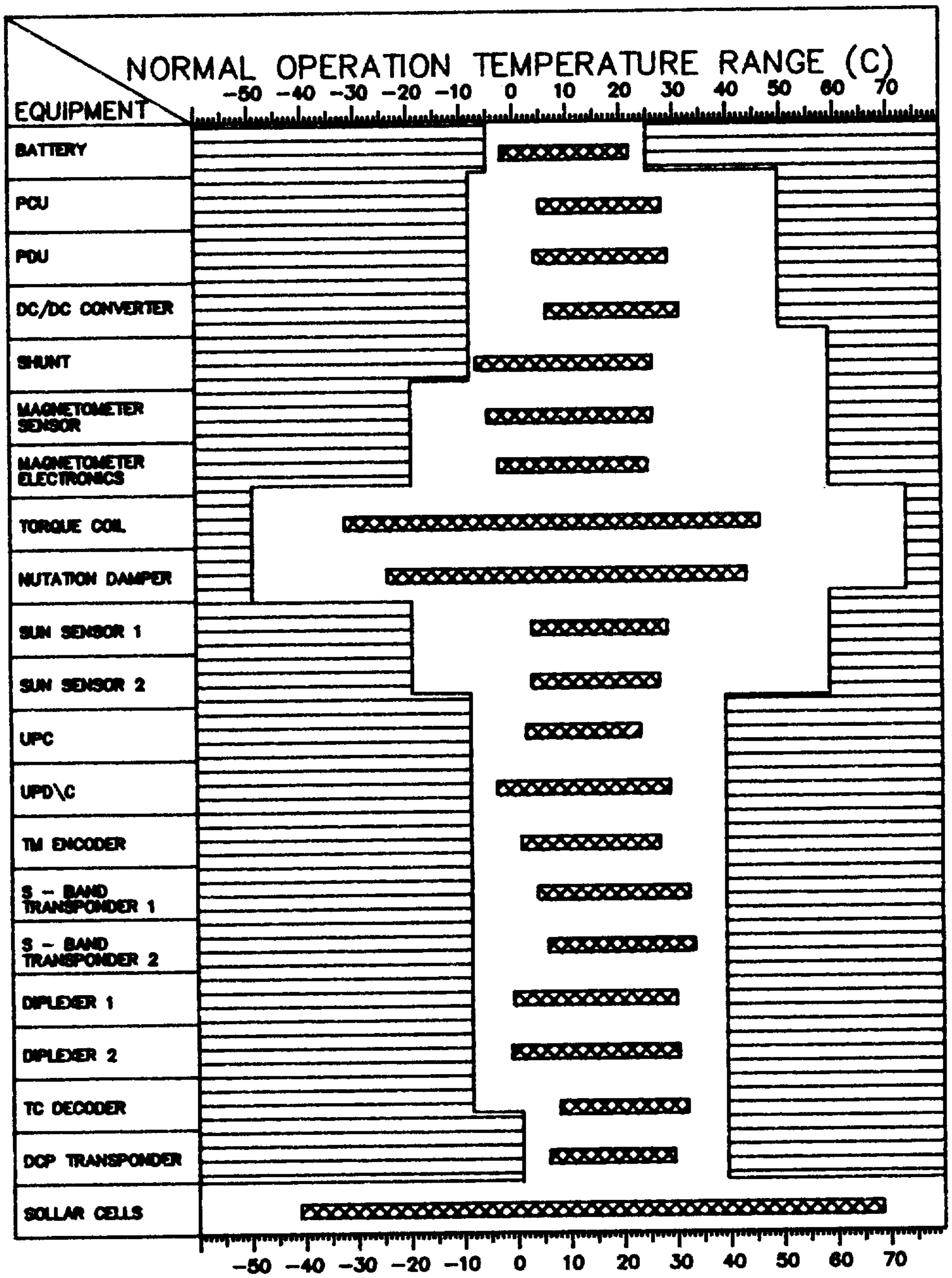

FIGURE 12.- FLIGHT PREDICTION VERSUS SPECIFIED TEMPERATURE RANGE

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