SPAR-R.1113 ISSUE A

REFERENCE PARAMETERS AND
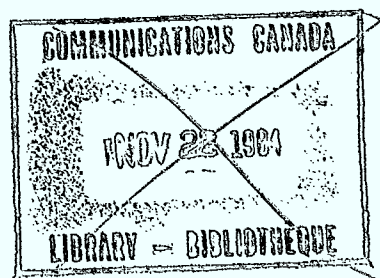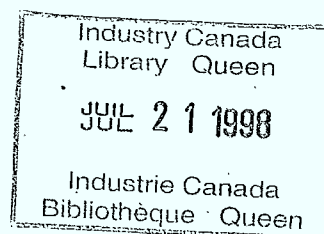# CONTROL SYSTEM PERFORMANCE
REQUIREMENTS FOR THIRD
GENERATION SPACECRAFT

SPAR

SPAR-R.1113   ISSUE A

REFERENCE PARAMETERS AND
CONTROL SYSTEM PERFORMANCE
REQUIREMENTS FOR THIRD
GENERATION SPACECRAFT

Prepared for:

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Reference: DSS Contract
             15ST.36100-1-0102

October 1982

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SPAR **Space & Electronics Group**

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SPACE PROGRAM

TITLE: REFERENCE PARAMETERS AND CONTROL SYSTEM PERFORMANCE
REQUIREMENTS FOR THIRD GENERATION SPACECRAFT

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REFERENCE PARAMETERS AND CONTROL SYSTEM PERFORMANCE

REQUIREMENTS FOR THIRD GENERATION SPACECRAFT

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1.0 INTRODUCTION

This document is prepared under the DSS Contract: "Design of an Attitude and Communications Beam Control System for Third Generation Spacecraft" (DSS F/N 15ST.36100-1-0102). The objective of this report is to identify a baseline spacecraft configuration and parameter set together with the control system performance requirements. Such a data base will form a common reference point for future studies of control system technologies for third generation spacecraft.

2.0 SYSTEM CONFIGURATION

The third generation spacecraft will generally be characterized by non-symmetric configurations, large distributed masses, flexible structures and tight beam pointing requirements. A typical candidate is represented by the Mobile Communications Satellite (M-SAT) shown in Fig. 2-1 which is currently under plan as a joint Canada/U.S.A. spacecraft. The projected time frame of operation is in the late 1980's for the demonstration (Canada coverage) spacecraft and the mid-1990's for the operational (North America coverage) spacecraft. For this reason, there is a definite element of timeliness in adopting M-SAT as the baseline spacecraft for the current design study.

2.1 Background

M-SAT is a geostationary communications spacecraft designed to provide radio telephony service for either Canada alone or together with the 50 United States including up to 200 miles of coastal waters. Such a satellite is intended to complement the urban-based terrestrial cellular mobile radio systems by extending coverage to the rural areas.

UHF Reflector

Aperture

RCS Propulsion Pack

Feed Array

Bus Structure

Astromasts

Solar Array
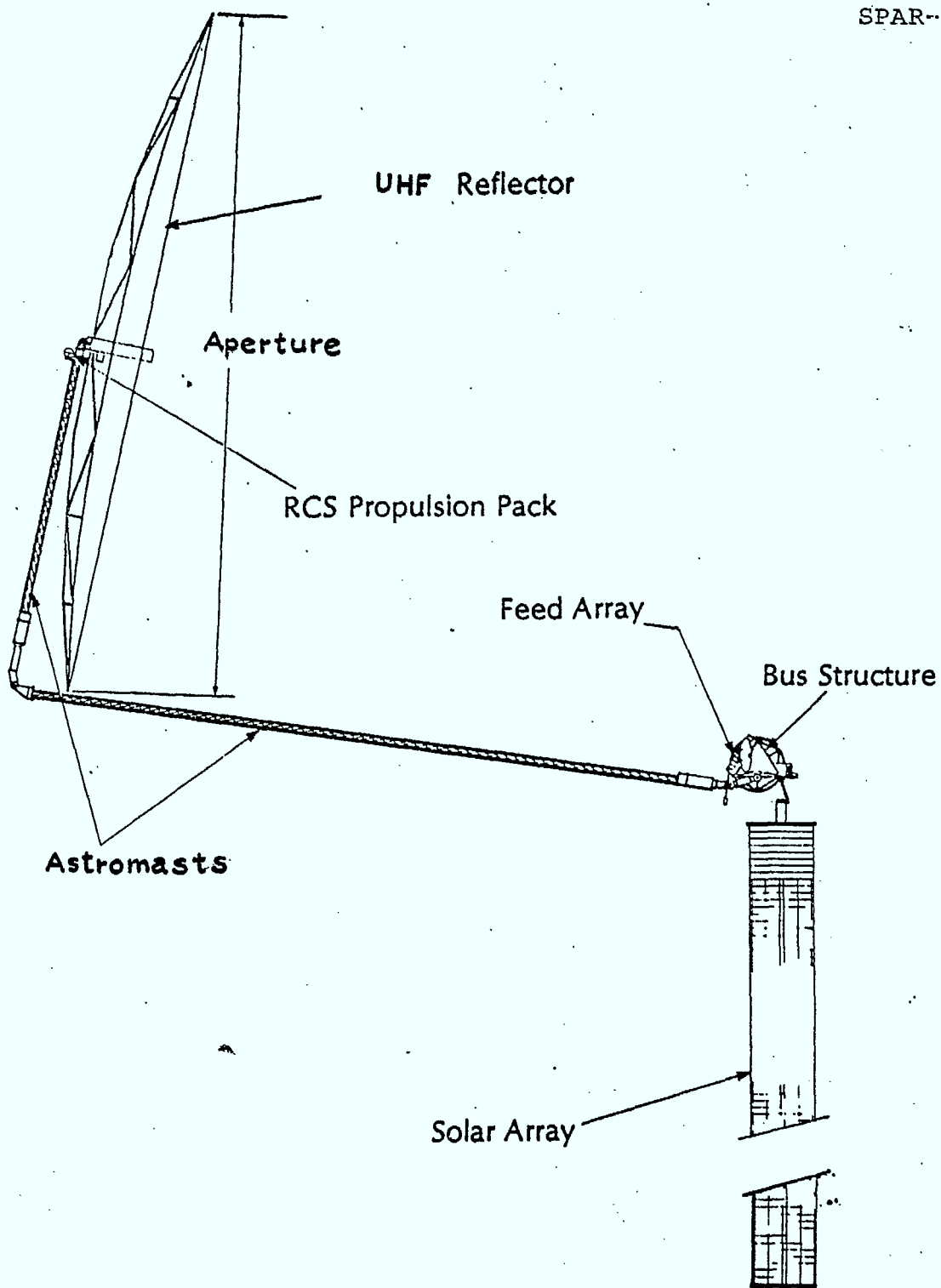
FIGURE 2-1   M-SAT CONCEPTUAL LAYOUT

2.1 Background (cond.)

Communication between the satellite and the base
stations in the cellular radio systems will be in
the 12/14 GHz bands whilst that between the satellite
and the mobile units will be in the 820/860 MHz range.
The total service area will be covered in multiple
contiguous spot beams in the UHF bands via the main
reflector (cf. Fig.2-2); a separate antenna will provide
backhaul beam coverage in the K-band (12/14 GHz).
Detailed trade-off analyses and sizing of the payload
system for M-SAT are documented in [1] and 2 .

2.2 Configuration Breakdown

From the dynamics standpoint, M-SAT may be broken down
into five major components as shown in Fig. 2-3:

(1) main body
(2) solar array
(3) boom
(4) tower
(5) main reflector

The main body comprises the bus structure on which is
located the bulk of the communications payload as well
as the various payload support subsystems such as
attitude control, reaction control, telemetry, tracking
and command, power (minus the solar array) and thermal
control. The boom and tower masts are taken to be Astro-
masts of appropriate dimensions. The mast hinges at the
main body/boom and the boom/tower interfaces are assumed
to be fixed once the spacecraft is deployed. The possibility
of actuating these joint hinges for attitude control will
not be considered in this study. However, antenna pointing
is achieved with a two-degree-of-freedom gimbal mechanism
located at the tower/reflector interface. This mechanism
will be considered as part of the reflector component in
the configuration breakdown.

2.3 Sensor and Actuator Selection

An appropriate complement of sensors and actuators have
been selected for the baseline configuration; they are
outlined in Table 2-1. The selection was based on a pre-
liminary analysis of the control system requirements as
outlined in [3]. More will be said about this in Sec. 6.
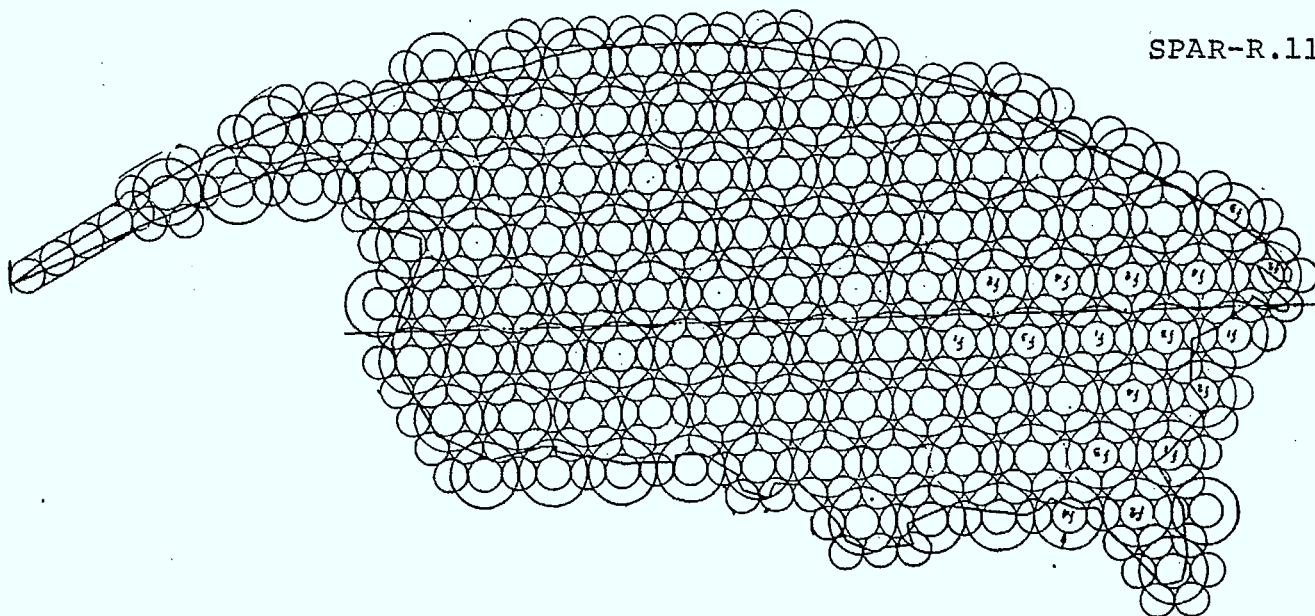
A)   NORTH AMERICA
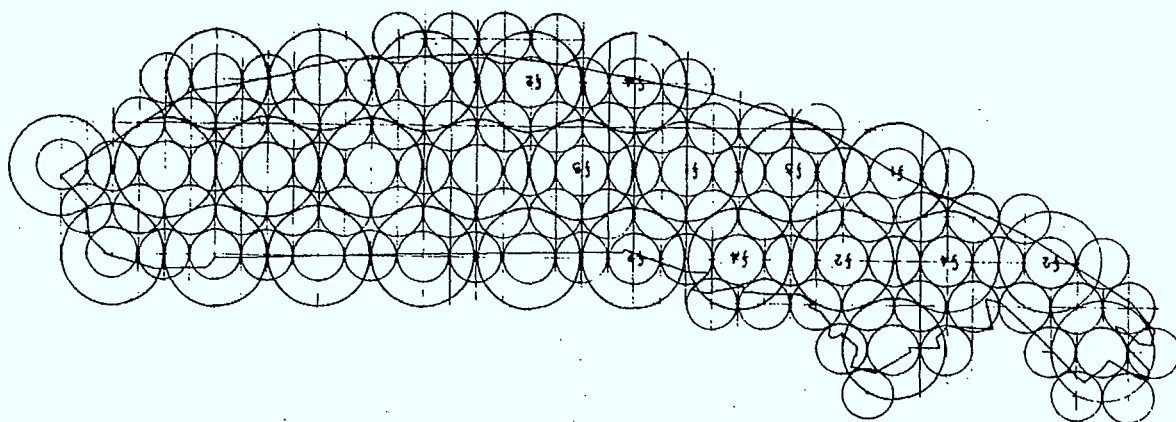
B)   CANADA

FIGURE 2-2   BEAM COVERAGE FROM 109° W LONGITUDE IN ORBIT

REFLECTOR

TOWER

BOOM

MAIN

BODY

SOLAR

ARRAY
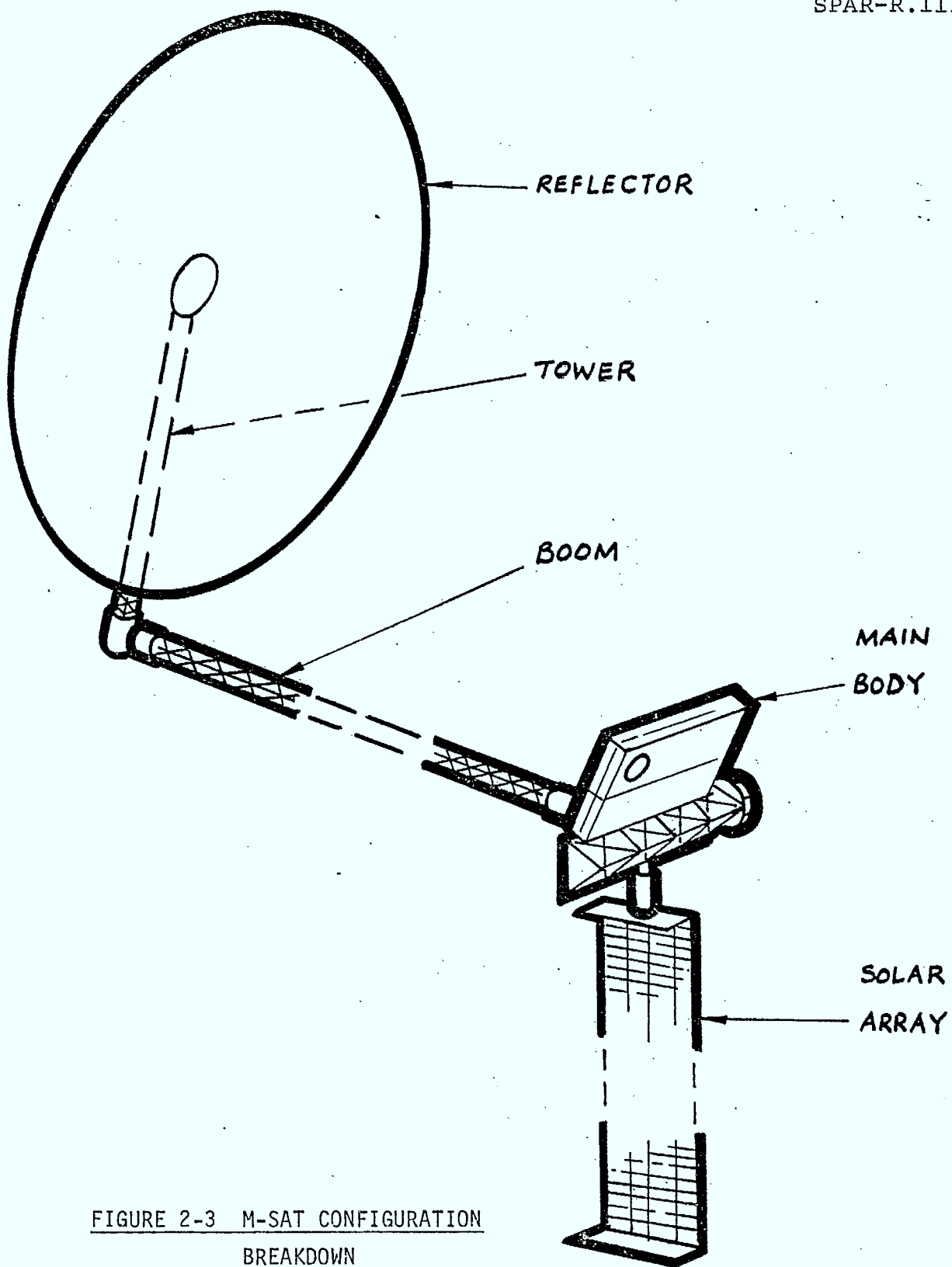
FIGURE 2-3 M-SAT CONFIGURATION
BREAKDOWN

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COMPONENTS	ACTUATORS	SENSORS
		- EARTH SENSORS
MAIN BODY	- REACTION WHEELS	- RF SENSOR
	- THRUSTERS	- SUN SENSORS
		- STAR TRACKERS
	- SOLAR ARRAY DRIVE	- GYROS
		- TACHO
		- LASER RANGE FINDER
		- ARRAY DRIVE ENCODERS
SOLAR ARRAY	- MAGNETIC TORQUER	- SUN SENSOR
		- LOCAL DISPLACEMENT/ VELOCITY SENSORS
BOOM TOWER	- LOCAL FORCE/TORQUE ACTUATORS	- LOCAL DISPLACEMENT/ VELOCITY SENSORS
	- TWO AXIS GIMBAL DRIVE	- ENCODERS
REFLECTOR	- MAGNETIC TORQUER	- TACHO
	- THRUSTERS	- STAR TRACKERS
	- LOCAL FORCE/TORQUE ACTUATORS	- LOCAL DISPLACEMENT/ VELOCITY SENSORS

2.3 Sensor and Actuator Selection (cond.)

The locations for the sensors and actuators are indicated in Fig. 2-4 and Fig. 2-5, respectively. It should be pointed out that neither the locations nor the selections are immutable and that both are subject to further design optimization.

3.0 BASELINE SYSTEM PARAMETERS

The baseline system selected for study is essentially the North American Operational satellite previously defined during the M-SAT program [1]. Figure 3-1 is a conceptual representation of the configuration. The diagram is not to scale. It includes dimensions of the key elements and the co-ordinates of a number of points of primary interest. The frame of reference used is body fixed and such that the +X-axis is along the orbital velocity vector, the +Y-axis is the southward pointing normal to the equatorial plane and the +Z-axis is towards the nadir for a correctly pointing satellite in an accurately equatorial circular orbit.

For the purpose of structural mass modelling, seven component elements are identified as follows:

1) Bus Structure + Feed Arrays
2) Solar Array
3) Boom
4) Tower
5) Reflector
6) Reflector Gimbal
7) Elbow

The mass properties are summarized in Table 3-1. Of the seven elements, the solar array, the boom, the tower and the reflector are assumed to be flexible; the other elements are assumed to be rigid. The flexibility of the boom and the tower are specified in Table 3-2 by two translational and one rotational stiffness value. The solar array flexibility is described in Table 3-3 in terms of the modal frequencies for the first out-of-plane, first in-plane and first torsional flexible modes. The flexibility of the reflector dish is specified in Table 3-3 by the first natural frequencies of two rocking modes and one torsional mode. Values for the modal damping coefficients are provided in Table 3-4 for each flexible mode.

STAR
TRACKERS

ENCODERS
TACHOS,

RF SIGNALS

RF
SENSOR

STAR
TRACKERS

LASER
BEAMS

SUN
SENSORS

EARTH
SENSORS

LASER
RANGE
FINDER

GYROS
TACHOS.

SUN
SENSOR

ENCODER

LOCAL
DISPL./VEL.
SENSORS
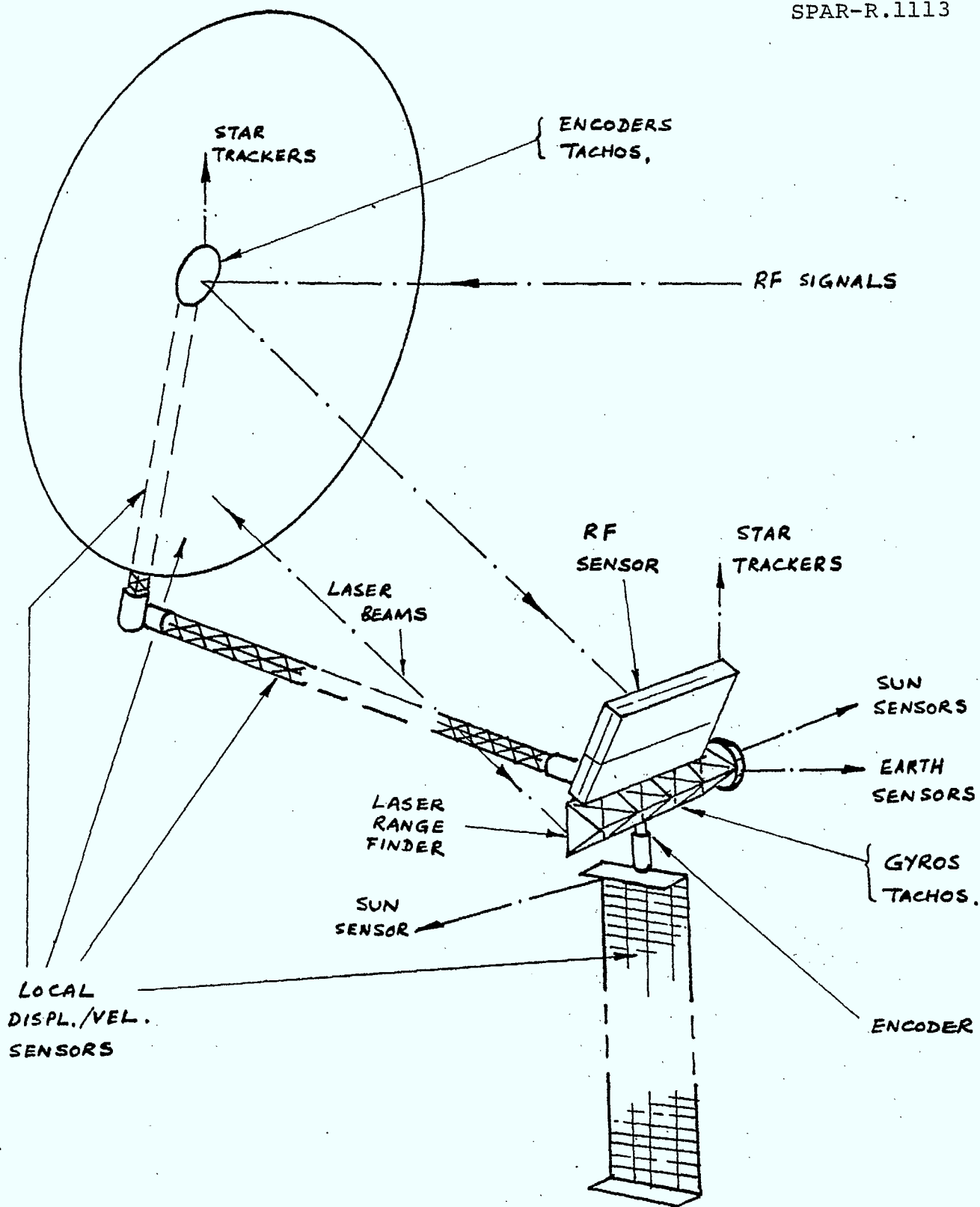
FIGURE 2-4 SENSOR LOCATIONS

AZ.

MAG. TORQUER

GIMBAL
DRIVE

EL.

THRUSTERS

LOCAL
FORCE / TORQUE
ACTUATORS

THRUSTERS

REACTION
WHEELS

MAG.
TORQUER

SOLAR
ARRAY
DRIVE
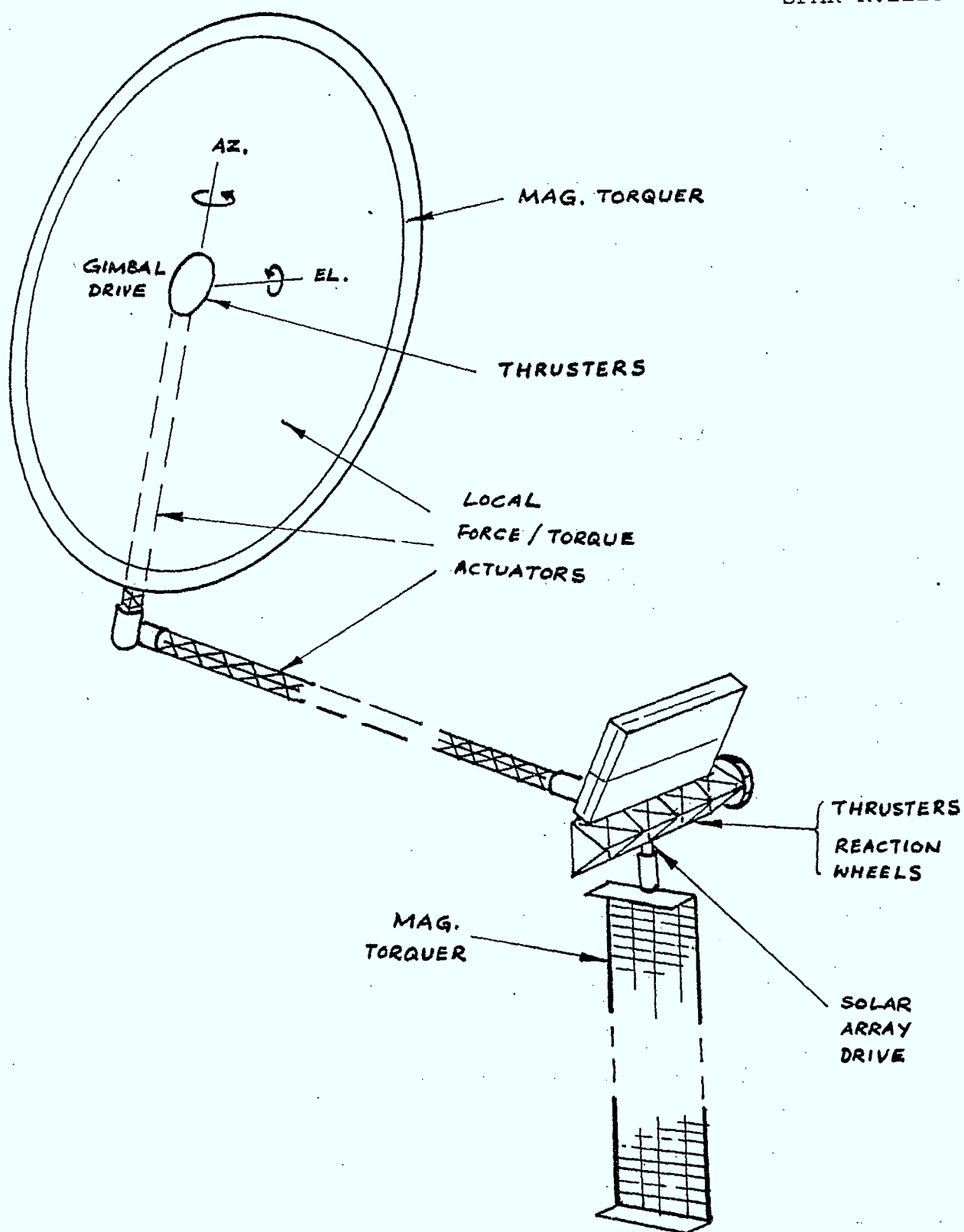
FIGURE 2-5 ACTUATOR LOCATIONS

Reflector
Gimbal
(0,-23.7,-43.7)

Lockheed Reflector
44.4 m Dia.

42.5 m Dia. Aperture

Tower
23.6 m

RCS Propulsion
Pack

Backhaul Feed

Feed Array 4.7 m × 9.2 m

PA

-Z   29°

(0,-0.58,-6.75)

⊕ CM

Bus Structure
(0,0,0)

(0,2.1,1.7)

Elbow
(0,-0.5,-47.8)

(0,1.8,-1)

Boom
46.9 m

+Y

0.5m Dia. Astromast

Solar Array
8.2 kW EOL

38 m
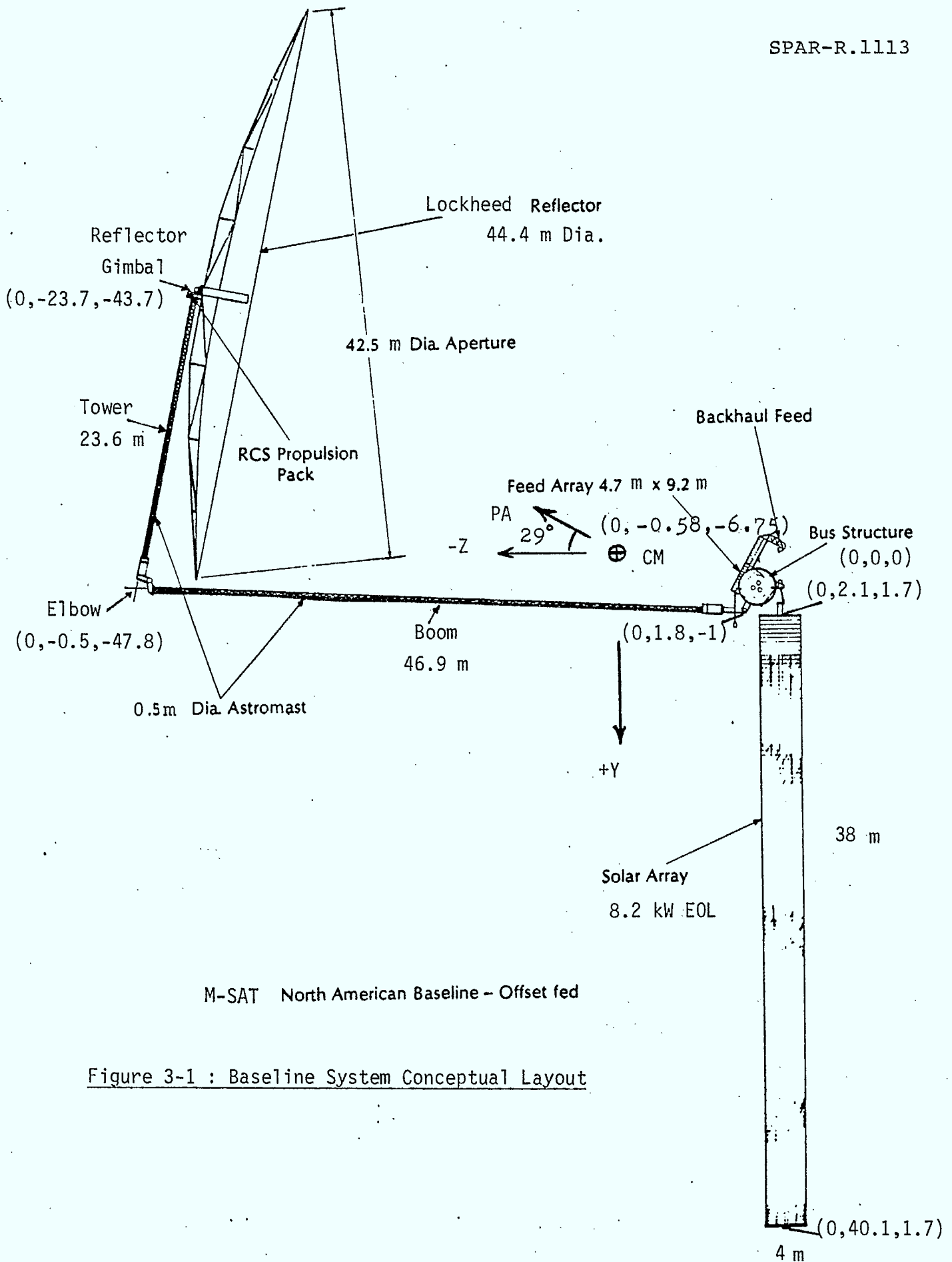
M-SAT   North American Baseline – Offset fed

Figure 3-1 : Baseline System Conceptual Layout

(0,40.1,1.7)

4 m

-10-

TABLE 3-1: BASELINE SYSTEM MASS PROPERTIES SUMMARY

ELEMENT	MASS (kg)	CM COORDINATES (m)			MOMENTS AND		PRODUCTS OF INERTIA (kg.m^2)			
		X	Y	Z	IXX	IYY	IZZ	IXY	IXZ	IYZ
1. BUS & FEED ARRAY	2644	0	-0.1	-0.8	3,614	18,325	18,925	0	0	-403
2. SOLAR ARRAY	320	0	21.10	1.7	57,132	297	56,865	0	0	0
3. BOOM	60	0	0.7	-25.4	9,108	9,075	35	0	0	0
4. TOWER	30	0	-13.1	-45.7	1,040	1,001	41	0	0	0
5. REFLECTOR	164	0	-23.5	-41.8	20,367	20,367	40,340	0	0	0
6. REFLECTOR GIMBAL	173	0	-23.57	-43.91	14	11	4	0	0	-6
7. ELBOW	144	0	-1.66	-47.43	46	3	42	0	0	12
PRINCIPAL AXIS TOTAL	3535	0	-0.58	-6.75	1,310,258	1,130,869	253,399	0	0	0

ROTATION FROM BODY FRAME TO PRINCIPAL AXIS FRAME = -28.87 DEGREES ABOUT X-AXIS

- MOI for elements (1) - (7) are about each element's CM in body-fixed reference frame.

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TABLE 3-2:   BASELINE SYSTEM BOOM AND TOWER STIFFNESS

	BOOM	TOWER
BENDING STIFFNESS $(N.m^2)$		
IN-PLANE	2.1×10^5 (XX-axis)	2.1×10^5 (XX-axis)
OUT-OF-PLANE	2.1×10^5 (YY-axis)	2.1×10^5 (ZZ-axis)
TORSIONAL STIFFNESS $(N.m^2)$	6.0×10^4 (ZZ-axis)	6.0×10^4 (YY-axis)
SHEAR STIFFNESS (N)	2.7×10^6	2.7×10^6

MODE	MODAL FREQUENCY (Hz)
	SOLAR ARRAY
1ST OUT-OF-PLANE BENDING	0.03
1ST IN-PLANE BENDING	0.03
1ST TORSIONAL	0.03
	REFLECTOR
1ST ELEVATION ROCKING	0.77
1ST AZIMUTH ROCKING	0.77
1ST TORSIONAL	0.39

TABLE 3-4: BASELINE SYSTEM FLEXIBLE MODE DAMPING COEFFICIENTS

MODE	DAMPING COEFFICIENT
	BOOM
IN-PLANE BENDING (XX-AXIS)	0.01
OUT-OF-PLANE BENDING (YY-AXIS)	0.01
TORSIONAL (ZZ-AXIS)	0.02
	TOWER
IN-PLANE BENDING (XX-AXIS)	0.01
OUT-OF-PLANE BENDING (ZZ-AXIS)	0.01
TORSIONAL (YY-AXIS)	0.02
	SOLAR ARRAY
OUT-OF-PLANE BENDING	0.01
IN-PLANE BEND ING	0.01
TORSIONAL	0.02
	REFLECTOR
ELEVATION ROCKING	0.01
AZIMUTH ROCKING	0.01
TORSIONAL	0.02

3.0   Baseline System Parameters (cond.)

The primary bus actuation system comprises a set of
roll, pitch and yaw reaction wheels with characteristics
as listed in Table 3-5. A complement of 22N bipropellant
thrusters is specified in Figure 3-2. They are employed
to perform the following manoeuvres:

1)   Momentum Dump   (MD)
2)   East-West Stationkeeping (EWSK)
3)   North-South Stationkeeping (NSSK)

The minimum on-time for each thruster is given as 10 ms.
Table 3-6 lists the thruster locations and orientations.
Table 3-7 lists the thruster combinations which will be
exercised to perform the various manoeuvres together with
the duty cycle and resulting torque components in the body
frame.

Solar array-mounted and reflector-mounted magnetic torquer
specifications are given in Table 3-8.

4.0   ALTERNATIVE SYSTEM PARAMETERS

The controller will be designed using the baseline system
parameters. The performance sensitivity to system para-
meters is a subject of some interest. A second set of
system parameters is therefore defined for use in system
test comparisons. The alternative system parameters
selected are those for the Canadian Operational satellite
defined during the M-SAT study program [2], and are pro-
vided below. Figure 4-1 is a conceptual representation of
the configuration and includes dimensions of the key
elements and the co-ordinates of points of primary interest.
The diagram is not to scale.

A body fixed frame of reference is used similar to that for
the baseline system, and the same seven component elements
are identified for the purpose of structural mass modelling.
The mass properties for the alternative system are summarized
in Table 4-1. As for the baseline system, the solar array,
the boom, the tower and the reflector are assumed to be flex-
ible and all other elements are assumed to be rigid. The
boom and tower flexibility values are given in Table 4-2.
The solar array flexibility is specified in Table 4-3. Values
for the modal damping coefficients are listed in Table 4-4.

TABLE 3-5: BASELINE SYSTEM REACTION WHEEL CHARACTERISTICS

SPAR

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PARAMETER	UNITS	VALUE
MAXIMUM WHEEL MOMENTUM	N.m.s.	±100
MAXIMUM WHEEL SPEED	rpm	±3000
MAXIMUM DRIVE TORQUE	N.m	±0.2
WHEEL TORQUE CONSTANT	N.m/A	0.056
WHEEL DRAG TORQUE AT MAXIMUM SPEED	N.m	0.001
WHEEL DRIVE VOLTAGE	V	28
MAXIMUM DRIVE POWER	W	100
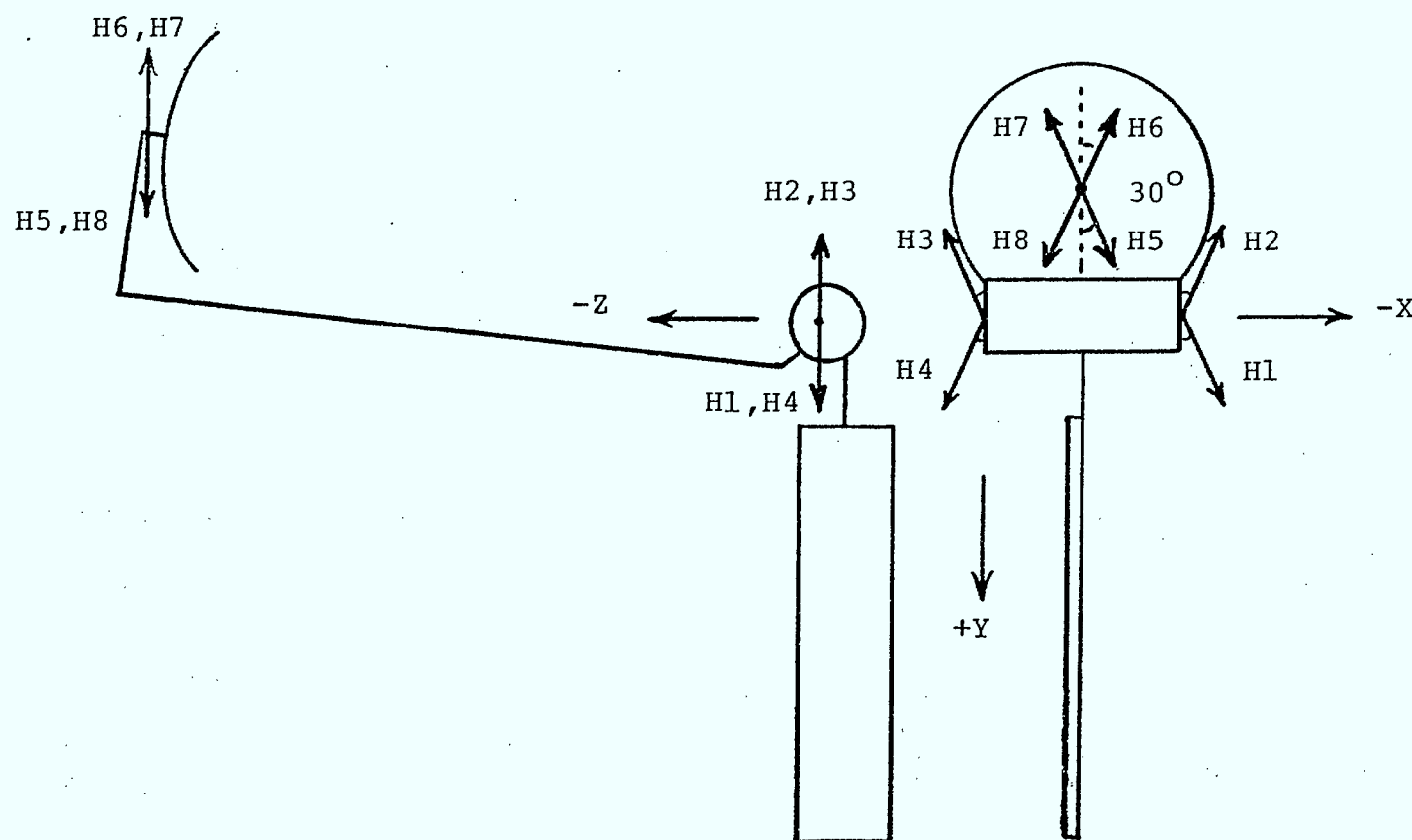
Figure 3-2: Baseline System Thruster Locations and Mass Expulsion Directions

Thruster Identification	Thruster Coordinates (m)			Reference Direction	Rotation to Thrust Direction	
	X	Y	Z		Axis	Angle(deg.)
H1	-6	0	0	-Y	Z	+30
H2	-6	0	0	+Y	Z	-30
H3	+6	0	0	+Y	Z	+30
H4	+6	0	0	-Y	Z	-30
H5	0	-23.7	-43.7	-Y	Z	+30
H6	0	-23.7	-43.7	+Y	Z	-30
H7	0	-23.7	-43.7	+Y	Z	+30
H8	0	-23.7	-43.7	-Y	Z	-30

CONTROL MODE	DUTY CYCLE								TORQUE COMPONENTS (N.m)		
	H1	H2	H3	H4	H5	H6	H7	H8	TX	TY	TZ
NSSK		1	1			0.183	0.183		0.45	0	0
EWSK			1	1			0.183	0.183	0	0.26	-80.3
+ ROLL						1	1		1408	0	0
- ROLL					1			1	-1408	0	0
+ PITCH	1							0.183	-0.226	148.6	61.4
- PITCH				1	0.183				-0.226	-148.6	-61.4
+ YAW	1				0.183				-0.226	-0.13	154.5
- YAW				1				0.183	-0.226	0.13	-154.5

TABLE 3-8:   BASELINE SYSTEM MAGNETIC TORQUER CHARACTERISTICS SPAR

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PARAMETER	UNITS	ARRAY-MOUNTED	REFLECTOR-MOUNTED
Area	m^2	150	1550
No. of turns	-		
Ampere-turns	A		
Resistance	Ω		
Reference Direction	-	+X/+Z	+Z
Rotation to coil normal			
- Axis		-	X
- Angle		0	-10

Reflector
Gimbal
(0,-15,-32.6)

Lockheed Reflector
31.9 m Dia.

30.5 m Dia. Aperture

Tower
15.9 m

RCS Propulsion
Pack

Feed Array 2.03 m x 5.79 m

PA

26°   (0, -0.72,-9.28)

-Z   ⊕ CM

Bus Structure
(0,0,0)

(0,2.3,1.4)

Elbow
(0,0.9,-33.8)

Boom
32.7 m

(0,1.1,1.1)

0.45 m Dia. Astromast

+Y

32.6 m

Solar Array

5 kW EOL

M-SAT Canadian   Baseline – Offset fed
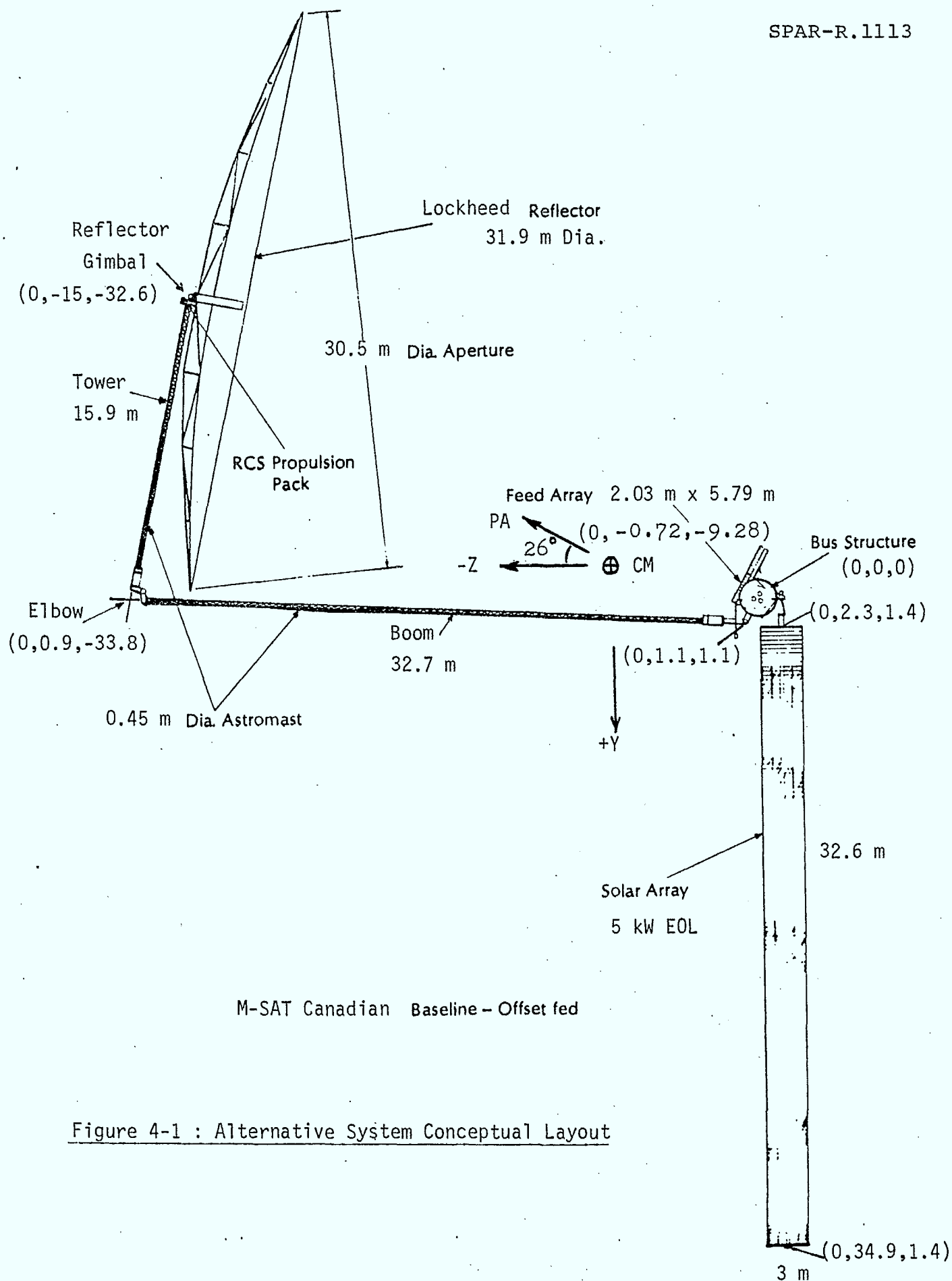
Figure 4-1 : Alternative System Conceptual Layout

(0,34.9,1.4)

3 m

TABLE 4-1: ALTERNATIVE SYSTEM MASS PROPERTIES SUMMARY

SPAR

Element	Mass (kg)	CM Coordinates (m)			Moments and Products of Inertia (kg.m^2)					
		X	Y	Z	IXX	IYY	IZZ	IXY	IXZ	IYZ
1. Bus + Feed Arrays	1056.5	0	0.05	- 0.12	542	1,182	918	0	0	-119
2. Solar Array	196	0	17	1.2	27,355	99	27,269	0	0	0
3. Boom	60	0	1	-18	4,501	4,501	1	0	0	0
4. Tower	30	0	- 8.3	-33.1	339	3	337	0	0	0
5. Reflector	170	0	-14.9	-31	9,559	9,559	19,768	0	0	0
6. Reflector Gimbal	126.5	0	-14.81	-32.84	13	8	5	0	0	-6
7. Elbow	157	0	- 0.49	-33.67	101	0	101	0	0	-6
Principal Axis Total	1,796	0	- 0.72	- 9.28	541,405	460,500	102,246	0	0	0

Rotation from Body Frame to Principal Axis Frame = -26.14 degrees about X-axis

- MOI for elements (1) - (7) are about each element's CM in body-fixed reference frame.

TABLE 4-2:   ALTERNATIVE SYSTEM BOOM AND TOWER STIFFNESS

	Boom	Tower
Bending Stiffness (N.m^2)		
In-plane	1.4×10^5 (XX-Axis)	1.4×10^5 (XX-axis)
Out-of-plane	1.4×10^5 (YY-axis)	1.4×10^5 (ZZ-axis)
Torsional Stiffness (N.m^2)	4.9×10^4 (ZZ-axis)	4.9×10^4 (YY-axis)
Shear Stiffness (N)	2.7×10^6	2.7×10^6

Mode	Modal Frequency (Hz)
	Solar Array
1st Out-of-Plane Bending	0.053
1st In-Plane Bending	0.053
1st Torsional	0.05
	Reflector
1st Elevation Rocking	1.25
1st Azimuth Rocking	1.25
1st Torsional	0.57

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Mode	Damping Coefficient
	Boom
In-Plane Bending (XX-axis) Out-of-Plane Bending (YY-axis) Torsional (ZZ-axis)	0.01 0.01 0.02
	Tower
In-Plane Bending (XX-axis) Out-of-Plane Bending (XX-axis) Torsional (YY-axis)	0.01 0.01 0.02
	Solar Array
Out-of-Plane Bending In-Plane Bending Torsional	0.01 0.01 0.02
	Reflector
Elevation Rocking Azimuth Rocking Torsional	0.01 0.01 0.02

4.0 Alternative System Parameters (cond)

As for the baseline system, the primary bus actuation system comprises a set of roll, pitch and yaw reaction wheels with characteristics as listed in Table 4-5. A complement of 22N bipropellant thrusters is specified in Figure 4-2. They are employed to perform the MD, EWSK and NSSK manoeuvres described for the baseline system. The minimum on-time for each thruster is given as 10 ms. Table 4-6 lists the thruster locations and orientations. Table 4-7 lists the thruster combinations which will be exercised to perform the various manoeuvres together with the duty cycle and resulting torque components in the body frame.

Solar array-mounted and reflector-mounted magnetic torquer specifications are given in Table 4-8.

TABLE 4-5:   ALTERNATIVE SYSTEM REACTION WHEEL CHARACTERISTICS   SPAR

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PARAMETER	UNITS	VALUE
MAXIMUM WHEEL MOMENTUM	N.m.s.	±100
MAXIMUM WHEEL SPEED	rpm	±3000
MAXIMUM DRIVE TORQUE	N.m	±0.2
WHEEL TORQUE CONSTANT	N.m/A	0.056
WHEEL DRAG TORQUE AT MAXIMUM SPEED	N.m	0.001
WHEEL DRIVE VOLTAGE	V	28
MAXIMUM DRIVE POWER	W	100
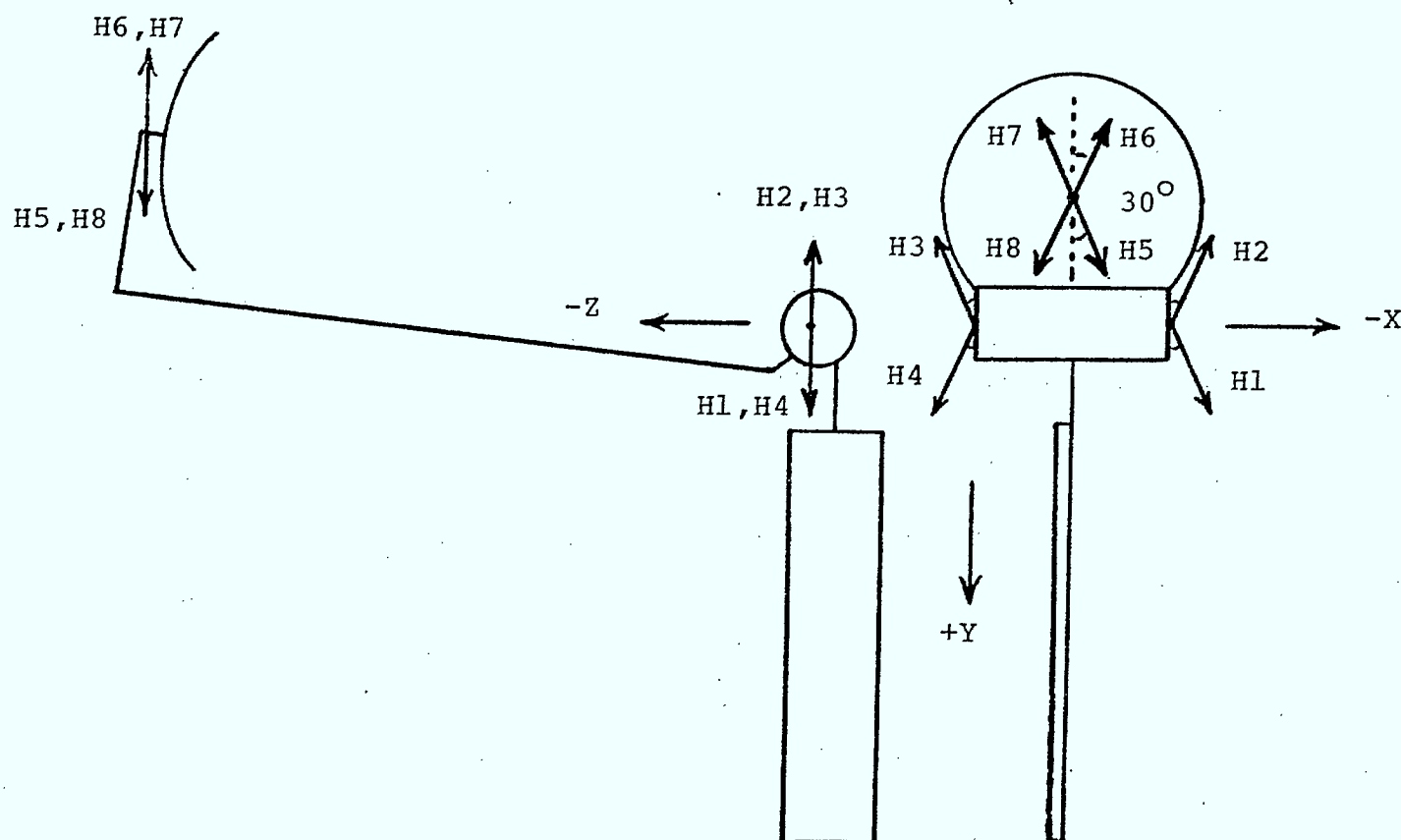
Figure 4-2: Alternative System Thruster Locations and Mass Expulsion Directions

Thruster Identification	Thruster Coordinates (m)			Reference Direction	Rotation to Thrust Direction	
	X	Y	Z		Axis	Angle (deg.)
H1	-4	0	0	-Y	Z	+30
H2	-4	0	0	+Y	Z	-30
H3	+4	0	0	+Y	Z	+30
H4	+4	0	0	-Y	Z	-30
H5	0	-15	-32.6	-Y	Z	+30
H6	0	-15	-32.6	+Y	Z	-30
H7	0	-15	-32.6	+Y	Z	+30
H8	0	-15	-32.6	-Y	Z	-30

CONTROL MODE	DUTY CYCLE								TORQUE COMPONENTS (N.m)		
THRUSTER	H1	H2	H3	H4	H5	H6	H7	H8	TX	TY	TZ
NSSK		1	1			0.398	0.398		0.052	0	0
EWSK			1	1			0.398	0.398	0	0.03	-109.2
+ ROLL						1	1		888.6	0	0
- ROLL					1			1	-888.6	0	0
+ PITCH	1							0.398	-0.026	204.2	5.77
- PITCH				1	0.398				-0.026	-204.2	-5.77
+ YAW	1				0.398				-0.026	-0.015	130.8
- YAW				1				0.398	-0.026	0.015	-130.8

TABLE 4-8: ALTERNATIVE SYSTEM MAGNETIC TORQUER CHARACTERISTICS **SPAR**

PARAMETER	UNITS	ARRAY-MOUNTED	REFLECTOR-MOUNTED
Area	m^2	100	800
No. of turns	-		
Ampere-turns	A		
Resistance	Ω		
Reference Direction	-	+X/+Z	+Z
Rotation to coil normal			
- Axis	-	-	X
- Angle	degrees	0	-5

..

10/6/mcs2/1

5.0 ENVIRONMENTAL DISTURBANCE FORCES AND TORQUES

Environmental disturbances, which may have been negligible in conventional satellites at geosynchronous altitude, could be a significant source of external forces and torques on M-SAT. This is due as much to the size of the spacecraft substructures as to its non-symmetric mass property distribution. The latter is demonstrated by the fact that the principal axes are skewed at close to 30° with respect to the control axes of the spacecraft.

The major sources of disturbance on M-SAT are gravitational, solar and the dynamic torques that result from ignoring the once-per-day rotation of the spacecraft in an inertial reference frame. Before we discuss each one of these disturbances, let us identify the reference frames shown in Figure 5-1. We ignore the declination of the satellite orbit from the ecliptic plane and define an earth-fixed inertial frame F_I with its Z-axis pointed toward Vernal Equinox; the Y-axis is the orbit normal. The position of the satellite is indicated by the orbital anomaly angle η in the inertial frame. The sun angle Ω rotates once per year in F_I.

There are three spacecraft-fixed reference frames. The main body frame (F_b) is centered at the vehicle centre-of-mass (CM) with the standard designation of roll (X_b), pitch (Y_b) and yaw (Z_b) axes. The reflector frame (F_r) is centered at the hub and is obtained from F_b via a rotation of 180° about the yaw axis. The array frame (F_a) rotates about an axis parallel to the pitch axis of F_b and has its Z-axis directed at the sun.

5.1 Gravitational Disturbance

In the case of M-SAT, the inertial matrix about the vehicle CM is defined in F_b by
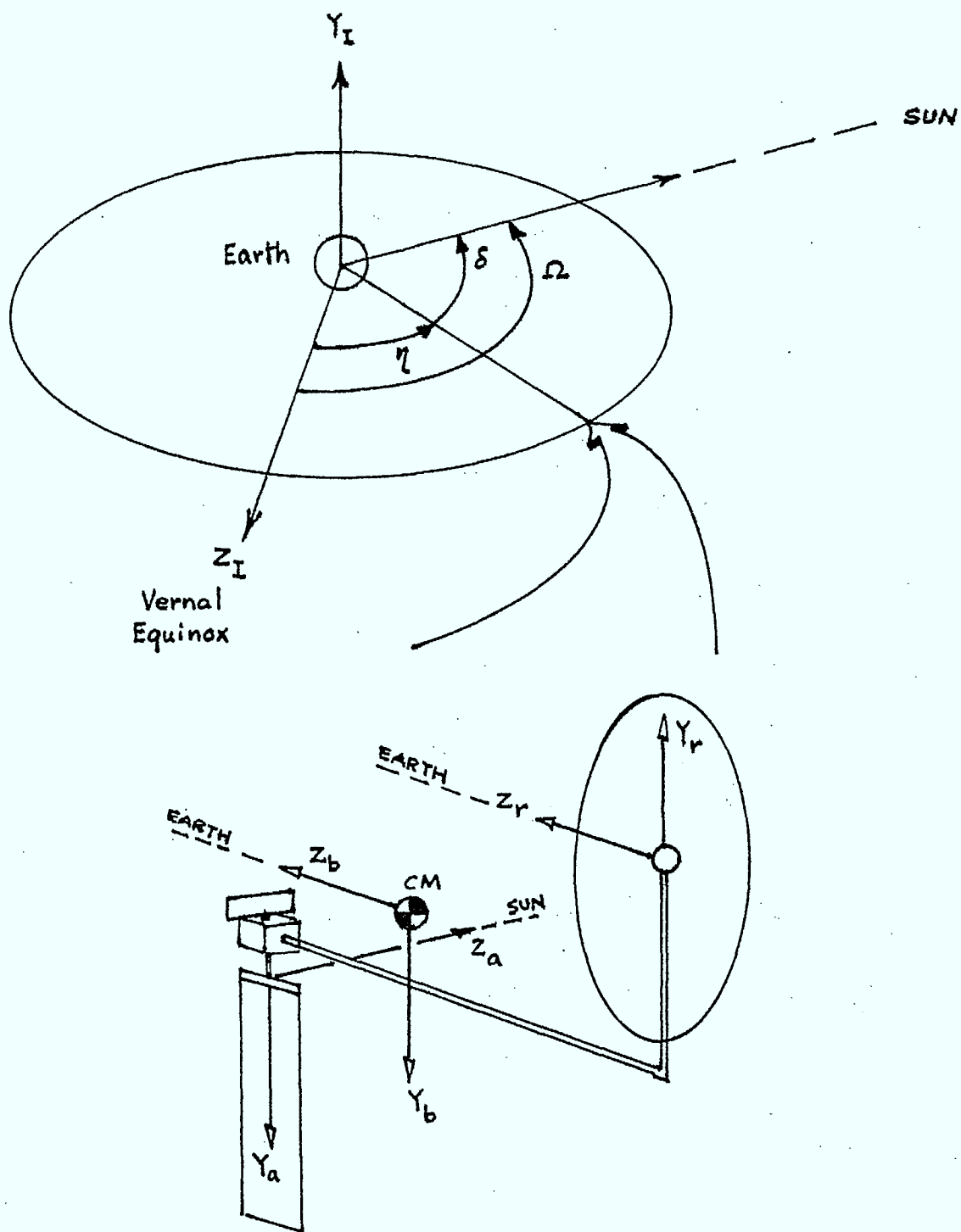
FIGURE 5-1   REFERENCE FRAMES

$$\mathcal{J} = \begin{bmatrix} I_x & 0 & 0 \\ 0 & I_y & -I_{yz} \\ 0 & -I_{yz} & I_z \end{bmatrix} \tag{5-1}$$

It can be shown that the gravitational forces acting on the spacecraft are expressed in F_b as

$$\left. \underset{\rightarrow}{F_g} \right|_{F_b} = \frac{3\omega_o^2}{R_O} \begin{bmatrix} 0 \\ -I_{yz} \\ 1/2 \ (I_z - I_x - I_y) \end{bmatrix} \tag{5-2a}$$

where ω_o and R_O denote the orbital rate and radius, respectively. The gravitational torque vector about the vehicle CM is given by

$$\left. \underset{\rightarrow}{L_g} \right|_{F_b} = 3\omega_o^2 \begin{bmatrix} I_{yz} \\ 0 \\ 0 \end{bmatrix} \tag{5-2b}$$

In the inertial frame F_I, these terms are given by

$$\left. \underset{\rightarrow}{F_g} \right|_{F_1} = \frac{3\omega_o^2}{R_O} \begin{bmatrix} -1/2 \ (I_z - I_x - I_y) \sin \eta \\ I_{yz} \\ -1/2 \ (I_z - I_x - I_y) \cos \eta \end{bmatrix} \tag{5-3a}$$

$$\left. \underset{\rightarrow}{L_g} \right|_{F_1} = 3\omega_o^2 \begin{bmatrix} I_{yz} \cos \eta \\ 0 \\ -I_{yz} \sin \eta \end{bmatrix} \tag{5-3b}$$

5.2   Once-per-Day Rotational Dynamic Disturbance

When the once-per-day rotation of the spacecraft is ignored from the equations of motion, the rotational dynamic appears as an equivalent disturbance torque described by the vector product

$$-\underrightarrow{w} \times \underrightarrow{H}$$

where $\underrightarrow{w}$ and $\underrightarrow{H}$ are the rotational rate vector and the angular momentum vector, respectively. In the body frame,

$$\underrightarrow{w}\Big|_{F_b} = \begin{bmatrix} 0 \\ -\omega_o \\ 0 \end{bmatrix}$$

and

$$\underrightarrow{H}\Big|_{F_b} = \mathcal{J}\,\underrightarrow{w} = -\omega_o \begin{bmatrix} 0 \\ Iy \\ -Iyz \end{bmatrix}$$

Hence, the rotational dynamic torque is given by

$$\underrightarrow{L_R}\Big|_{F_b} = -\underrightarrow{w} \times \underrightarrow{H} = \omega_o^2 \begin{bmatrix} Iyz \\ 0 \\ 0 \end{bmatrix} \qquad (5\text{-}4a)$$

In the inertial frame, we get

$$\underrightarrow{L_R}\Big|_{F_I} = \omega_o^2 \begin{bmatrix} Iyz \cos \eta \\ 0 \\ -I_{yz} \sin \eta \end{bmatrix} \qquad (5\text{-}4b)$$

Comparing (5-4) to (5-2b) and (5-3b), we observe that the rotational dynamic torques are of the same order of magnitude as the gravitational torques. For this reason, the two torques are sometimes combined into a single mass properties disturbance term.

5.3   Solar Disturbance

To calculate the total solar contribution, we assume all the reflective surfaces (namely, the reflector, the antenna feed array and the solar array) to be positioned orthogonal to the orbital (ecliptic) plane. Furthermore, insofar as solar exposure is concerned, the reflector can be approximated by a cylinder with an opaque frontal area equal to 5% of the reflector aperture.

Define the following parameters:

$\dfrac{Q}{C}$ = solar energy constant   .

ν_a, ν_f, ν_r = reflectivities of array, feed and reflector

A_a, A_f, A_r = surface areas of array, feed and reflector

Also, let the moment arms from the vehicle CM to the centres-of-pressure on the various surfaces be expressed in F_b as follows:

$$\text{reflector } \underline{\rho}_r \Big|_{F_b} = \begin{pmatrix} 0 \\ \rho_{ry} \\ \rho_{rz} \end{pmatrix}$$

feed
$$\underrightarrow{P_f}\Big|_{F_b} = \begin{bmatrix} 0 \\ P_{fy} \\ P_{fz} \end{bmatrix}$$

array
$$\underrightarrow{P_a}\Big|_{F_b} = \begin{bmatrix} 0 \\ P_{ay} \\ P_{az} \end{bmatrix}$$

Then the total solar force exerted on the spacecraft is given in F_b by

$$\underrightarrow{F_s}\Big|_{F_b} = -\frac{Q}{c} A_a (1+\nu_a) \begin{bmatrix} \sin\delta \\ 0 \\ -\cos\delta \end{bmatrix} + \frac{Q}{2c} A_f \begin{bmatrix} -(1-\nu_f)\sin 2\delta \\ 0 \\ (1+\nu_f)(1+\cos 2\delta) \end{bmatrix}$$

$$+ \frac{Q}{20c} A_r (1+\nu_r) \begin{bmatrix} -\sin\delta \\ 0 \\ \cos\delta \end{bmatrix} \tag{5-5a}$$

The solar torques about the vehicle CM are given by

$$\underrightarrow{L_s}\Big|_{F_b} = -\frac{Q}{c} A_a (1+\nu_a) \begin{bmatrix} -P_{ay}\cos\delta \\ P_{az}\sin\delta \\ -P_{ay}\sin\delta \end{bmatrix} + \frac{Q}{2c} A_f \begin{bmatrix} (1+\nu_f)P_{fy}(1+\cos 2\delta) \\ -(1-\nu_f)P_{fz}\sin 2\delta \\ (1-\nu_f)P_{fy}\sin 2\delta \end{bmatrix}$$

$$+ \frac{Q}{20c} A_r (1+\nu_r) \begin{bmatrix} P_{ry}\cos\delta \\ -P_{rz}\sin\delta \\ P_{ry}\sin\delta \end{bmatrix} \tag{5-5b}$$

In the inertial frame F_I, these terms are written as

$$\vec{F_s}\Big|_{F_I} = -\frac{Q}{c} A_a (1+\nu_a) \begin{bmatrix} \sin \Omega \\ 0 \\ \cos \Omega \end{bmatrix}$$

$$+ \frac{Q}{2c} A_f \begin{bmatrix} -(1-\nu_f)\sin 2\delta \cos \eta - (1+\nu_f)(1+\cos 2\delta)\sin \eta \\ 0 \\ (1-\nu_f)\sin 2\delta \sin \eta - (1+\nu_f)(1+\cos 2\delta)\cos \eta \end{bmatrix}$$

$$+ \frac{Q}{2_0 c} A_r (1+\nu_r) \begin{bmatrix} -\sin \Omega \\ 0 \\ -\cos \Omega \end{bmatrix} \tag{5-6a}$$

$$\vec{L_s}\Big|_{F_I} = -\frac{Q}{c} A_a (1+\nu_a) \begin{bmatrix} -\rho_{ay} \cos \Omega \\ -\rho_{az} \sin \delta \\ \rho_{ay} \sin \Omega \end{bmatrix}$$

$$+ \frac{Q}{2c} A_f \begin{bmatrix} (1+\nu_f)\rho_{fy}(1+\cos 2\delta)\cos \eta - (1-\nu_f)\rho_{fy}\sin 2\delta \sin \eta \\ (1-\nu_f)\rho_{fz}\sin 2\delta \\ -(1+\nu_f)\rho_{fy}(1+\cos 2\delta)\sin \eta - (1-\nu_f)\rho_{fy}\sin 2\delta \cos \eta \end{bmatrix}$$

$$+ \frac{Q}{20c} A_r (1 + \nu_r) \begin{bmatrix} \rho_{ry} \cos \Omega \\ \rho_{rz} \sin \delta \\ -\rho_{ry} \sin \Omega \end{bmatrix}$$

$$(5\text{-}6b)$$

5.4   Summary of Environmental Disturbances

We conclude by computing the total environmental disturbance forces and torques for the two M-SAT configurations presented earlier. Table 5-1 lists the relevant spacecraft parameters required for the calculations. The results are summarized in Tables 5-2 and 5-3.

PARAMETER	BASELINE SYSTEM	ALTERNATIVE SYSTEM
Inertial Products (Kg-m^2)		
I_x	1,286,397	541,353
I_y	910,579	390,920
I_z	449,828	171,775
I_{yz}	364,845	141,706
Moment Arms (m)		
ρ_{ry} / ρ_{rz}	-22.92 /-35.05	-14.18 /-21.72
ρ_{fy} / ρ_{fz}	0.48 / 5.95	0.77 / 9.16
ρ_{ay} / ρ_{az}	21.68 / 8.45	17.72 / 10.48
Surface Areas (m^2)		
A_r/ A_f/ A_a	1418.6/43.2/152	730.6/11.75/97.8
Reflectivities		
ν_r/ ν_f/ ν_a	0.25	0.25
Orbital Rate (rad/s) ω_o	7.2722×10^{-5}	
Orbital Radius (m) R_o	42.238×10^6	
Solar Energy Constant Q/c (N/m^2)	4.51×10^{-6}	

TABLE 5-1 SPACECRAFT PARAMETERS FOR DISTURBANCE CALCULATIONS

| | GRAVITATIONAL | | ROTATIONAL |
	FORCE (N)	TORQUE (N-m)	TORQUE (N-m)
Body Frame (F_b)			
X_b - Roll	0	5.788×10^{-3}	1.930×10^{-3}
Y_b - Pitch	-1.370×10^{-10}	0	0
Z_b - Yaw	-3.281×10^{-10}	0	0
Inertial Frame (F_I)			
X_I	$3.281 \times 10^{-10} S_\eta$	$5.788 \times 10^{-3} C_\eta$	$1.930 \times 10^{-3} C_\eta$
Y_I - Orbit Normal	1.370×10^{-10}	0	0
Z_I - Vernal Equinox	$3.281 \times 10^{-10} C_\eta$	$-5.788 \times 10^{-3} S_\eta$	$-1.930 \times 10^{-3} S_\eta$

$S_\eta = \sin \eta$; $C_\eta = \cos \eta$

a) Gravitational and Rotational Dynamic Disturbances

TABLE 5-2   ENVIRONMENTAL DISTURBANCES IN BASELINE SYSTEM

	SOLAR	
	FORCE (N)	TORQUE (N-m)
Body Frame (F_b)		
X_b - Roll	$-1.257 \times 10^{-3}\, s_\delta - 7.313 \times 10^{-5}\, s_{2\delta}$	$9.412 \times 10^{-3}\, c_\delta + 5.850 \times 10^{-5}(1 + c_{2\delta})$
Y_b - Pitch	0	$6.775 \times 10^{-3}\, s_\delta - 4.351 \times 10^{-4}\, s_{2\delta}$
Z_b - Yaw	$1.257 \times 10^{-3}\, c_\delta + 1.219 \times 10^{-4}(1 + c_{2\delta})$	$9.412 \times 10^{-3}\, s_\delta + 3.510 \times 10^{-5}\, s_{2\delta}$
Inertial Frame (F_I)		
X_I	$-1.257 \times 10^{-3}\, s_\Omega - 7.313 \times 10^{-5}\, s_{2\delta}\, c_\eta - 1.219 \times 10^{-4}(1 + c_{2\delta})\, s_\eta$	$9.412 \times 10^{-3}\, c_\Omega + 5.850 \times 10^{-5}(1 + c_{2\delta})\, c_\eta - 3.510 \times 10^{-5}\, s_{2\delta}\, s_\eta$
Y_I - Orbit Normal	0	$-6.775 \times 10^{-3}\, s_\delta + 4.351 \times 10^{-4}\, s_{2\delta}$
Z_I - Vernal Equinox	$-1.257 \times 10^{-3}\, c_\Omega + 7.313 \times 10^{-5}\, s_{2\delta}\, s_\eta - 1.219 \times 10^{-4}(1 + c_{2\delta})\, c_\eta$	$-9.412 \times 10^{-3}\, s_\Omega - 5.850 \times 10^{-5}(1 + c_{2\delta})\, s_\eta - 3.510 \times 10^{-5}\, s_{2\delta}\, c_\eta$

$s_\eta = \sin \eta$; $c_\eta = \cos \eta$; $s_\delta = \sin \delta$; $c_\delta = \cos \delta$; $s_{2\delta} = \sin 2\delta$; $c_{2\delta} = \cos 2\delta$

$s_\Omega = \sin \Omega$; $c_\Omega = \cos \Omega$

b) Solar Radiation

TABLE 5-2   ENVIRONMENTAL DISTURBANCES IN BASELINE SYSTEM (CONTINUED)

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| | GRAVITATIONAL | | ROTATIONAL |
	FORCE (N)	TORQUE (N-m)	TORQUE (N-m)
Body Frame (F_b)			
X_b - Roll	0	2.248×10^{-3}	7.494×10^{-4}
Y_b - Pitch	-5.323×10^{-11}	0	0
Z_b - Yaw	-1.428×10^{-10}	0	0
Inertial Frame (F_I)			
X_I	$1.428 \times 10^{-10} \, S_\eta$	$2.248 \times 10^{-3} \, C_\eta$	$7.494 \times 10^{-4} \, C_\eta$
Y_I - Orbital Normal	5.323×10^{-11}	0	0
Z_I - Vernal Equinox	$1.428 \times 10^{-10} \, C_\eta$	$-2.248 \times 10^{-3} \, S_\eta$	$-7.494 \times 10^{-4} \, S_\eta$

$S_\eta = \sin \eta$; $C_\eta = \cos \eta$

a) Gravitational and Rotational Dynamic Disturbances

TABLE 5-3 ENVIRONMENTAL DISTURBANCES IN ALTERNATIVE SYSTEM

	SOLAR	
	FORCE (N)	TORQUE (N-m)
Body Frame (F_b)		
X_b – Roll	$-7.573 \times 10^{-4}\, s_\delta - 1.987 \times 10^{-5}\, s_{2\delta}$	$6.850 \times 10^{-3}\, c_\delta + 2.550 \times 10^{-5}\, (1 + c_{2\delta})$
Y_b – Pitch	0	$-1.305 \times 10^{-3}\, s_\delta - 1.820 \times 10^{-4}\, s_{2\delta}$
Z_b – Yaw	$7.573 \times 10^{-4}\, c_\delta + 3.312 \times 10^{-5}\, (1 + c_{2\delta})$	$6.850 \times 10^{-3}\, s_\delta + 1.530 \times 10^{-5}\, s_{2\delta}$
Inertial Frame (F_I)		
X_I	$-7.573 \times 10^{-4}\, s_\Omega - 1.987 \times 10^{-5}\, s_{2\delta}\, c_\eta - 3.312 \times 10^{-5}\, (1 + c_{2\delta})s_\eta$	$6.850 \times 10^{-3}\, c_\Omega + 2.550 \times 10^{-5}\, (1 + c_{2\delta})c_\eta - 1.530 \times 10^{-5}\, s_{2\delta}\, s_\eta$
Y_I – Orbit Normal	0	$1.305 \times 10^{-3}\, s_\delta + 1.820 \times 10^{-4}\, s_{2\delta}$
Z_I – Vernal Equinox	$-7.573 \times 10^{-4}\, c_\Omega + 1.987 \times 10^{-5}\, s_{2\delta}\, s_\eta - 3.312 \times 10^{-5}\, (1 + c_{2\delta})c_\eta$	$-6.850 \times 10^{-3}\, s_\Omega - 2.550 \times 10^{-5}\, (1 + c_{2\delta})s_\eta - 1.530 \times 10^{-5}\, s_{2\delta}\, c_\eta$

$$s_\eta = \sin \eta \;,\; c_\eta = \cos \eta \;,\; s_\delta = \sin \delta \;,\; c_\delta = \cos \delta \;,\; s_{2\delta} = \sin 2\delta \;,\; c_{2\delta} = \cos 2\delta$$
$$s_\Omega = \sin \Omega \;,\; c_\Omega = \cos \Omega$$

b) Solar Radiation

TABLE 5-3   ENVIRONMENTAL DISTURBANCES IN ALTERNATIVE SYSTEM (CONTINUED)

6.0 CONTROL SYSTEM PERFORMANCE REQUIREMENTS

This section lists the major performance requirements for the attitude control system which will form part of the design goals for third generation spacecraft.

The beam pointing accuracy and stability of M-SAT are determined by the allowable gain drop-off of the beams at the outer edge of the coverage area. From the communications point of view, it is desirable to limit the gain drop to no more than 1 dBW. Thus, if the gain roll-off at the edge of the main beam is x dBW/deg., then the combined beam pointing accuracy and stability should be better than 1/x deg.

Based on this line of reasoning, a pointing error budget was drawn up in [3]. The conclusion reached there was that the attitude control accuracy could not be met given the current state of the art in sensor technology. A more realistic requirement would be to point the main beams to within half a beam width. Even in this case, RF sensing and high quality strap-down inertial sensors augmented by star sensors may be required.

A further concern in M-SAT is beam defocussing due to the structural flexibility of the reflector/boom/feeds combination. Defocussing results in increase in the beam sidelobes which in turn deteriorates the quality of the RF signals. To correct this situation, the control system needs information on the deflections of the structures as well as focal detection capability. This leads to the use of distributed sensors and actuators and possibly on-board processing of the communications signals for attitude control purposes.

Yet another requirement of the attitude control system for the third generation spacecraft is the continual maintenance of the pointing accuracies during periods of special operation such as stationkeep and momentum dump. In this regard, an earlier study [4] concluded that variable-gain control was feasible for a rigid symmetric spacecraft. The case here for some sort of gain-adjust or adaptive control would appear to be even stronger due to the non-symmetry and uncertain disturbance levels of M-SAT.

7.0 REFERENCES

[1] "M-SAT North American Operational - Report and Baseline Performance Document", SSD, Spar Aerospace Ltd., May 28, 1981.

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[3] "Final Report - Study on Control of Large Spacecraft", SPAR-R.1095, RMSD, Spar Aerospace Ltd., May 1981.

[4] "Integrated Attitude Sensing and Control System Follow-on Study: January-March, 1979 Report", SPAR-R.1001, RMSD, Spar Aerospace Ltd.,April 1979.