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APPROVED BY NASA

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BLOCK II

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PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM MANUAL

INDEXING DATA

DATE

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PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER

BY

DELCO ELECTRONICS
DIVISION OF GENERAL MOTORS
SANTA BARBARA, CALIFORNIA

NASA CONTRACT NAS 9-497

10 MAR 1966

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

Record of Revisions

Rev. Let.	Date	TDRR Number	Pages Revised	Approval	
				CCB	NASA
A	7-21-66	30233	Title/ii, iii/iv, vii, viii, xv, xvi, xix/xx, xxiii, xxiv, xxvi, 1-15/1-16 2-52, 2-67, 3-1, 3-2, 3-2A/3-2B 3-7 thru 3-12, 3-12A/3-12B, 3-13 3-21, 3-22A/3-22B, 3-27, 3-28, 3-28A/3-28B, 3-33 thru 3-35, 3-38 thru 3-40, 3-41/3-42, 6-8 7-3 thru 7-8, 7-18, 7-19, 7-22, 7-22A/7-22B, 7-27/7-28, 7-29/7-30, 7-33/7-34, 7-35/7-36, 7-39/7-40	EA <i>JM</i>	
B	9-8-66	31007	Title, iii/iv, vii, viii, xiii, xiv, xvii/xviii, xix/xx, xxiv, 3-2, 3-2A/3-2B, 3-30, 3-38, 6-5 thru 6-6A/6-6B, 6-9, 8-1 thru 8-4A/8-4B, 8-5 thru 8-13/8-14	EA <i>DD</i>	
C	11-17-66	31972	Title/ii, iii/iv, vii, viii, xi, xvi, xxivA/xxivB, xxvi, 1-3, 3-1, 3-2, 3-2A/3-2B, 3-16, 3-16A/3-16B, 3-24, 4-50, 4-52, 4-57, 4-58, 4-58A/4-58B, 4-61 thru 4-64, 4-66A/4-66B, 4-67, 6-11 thru 6-17/6-18, 7-29	EA <i>DD</i>	
D	12-1-66	32185	Title/ii thru iv, vii thru viii, xi, xiii, xvii/xviii, 3-2A/3-2B,	EA <i>DD</i>	

Record of Revisions

Rev. Let.	Date	TDRR Number	Pages Revised	Approval	
				CCB	NASA
D			7-2 thru 7-6, 7-25/7-26, 7-31/ 7-32, 7-37/7-38, 7-43/7-44, 7-45/7-46, 8-3/8-4, 8-9, A-12		
E	1-19-67	32726	Title/ii, iv, vii, viii, viiiA/ viiiB, xvii/xviii, xxivA/xxivB, 3-2A/3-2B, 3-34, 6-5, 6-7, 7-25/7-26, 7-43/7-44, 7-45/7-46, 7-47/7-48, 7-49/7-50, 8-2, 8-3/8-4, 8-6, 8-8 thru 8-10	EA	<i>[Signature]</i>
F	3-21-67	33373	Title/ii, iv, vii thru viiiA/viiiB, x, xi, xiv, xvi, xxiv, xxivA/ xxivB, 3-1 thru 3-2B, 3-3, 3-6, 3-10, 3-16 thru 3-16B, 3-21, 3-24, 3-25, 3-27, 4-23, 5-3, 6-8A/6-8B, 6-9, 7-25/7-26, 7-31/7-32, 7-37/7-38, 7-43/ 7-44, 8-1 thru 8-3/8-4, 8-10, A-2	EA	<i>[Signature]</i>
G	4-20-67	33651	Title/ii, iv, vii, viii, viiiA/viiiB, xv, xxv thru xxxii, 7-7, 7-25/7-26, 7-29/7-30, 7-31/7-32, 7-33/7-34, 7-35/7-36, 7-37/7-38, 7-43/ 7-44, 7-47/7-48, 7-49/7-50, 8-1, 8-2, 8-4A/8-4B, 8-7 thru 8-11/8-12, 8-13/8-14	EA	<i>[Signature]</i>

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

Record of Revisions

Rev. Let.	Date	TDRR Number	Pages Revised	Approval	
				CCB	NASA
H	4-27-67	33689	Title/ii, ivA/ivB, vii thru viiiA/viiiB, x, xvi, xix, xx, xxivA thru xxivE/xxivF, 3-1, 3-2, 3-2A thru 3-2AG/3-2AH, 3-3, 3-4, 3-6, 3-19, 3-20, 6-6, 6-6A/6-6B, 6-7, A-6	EA <i>CHIT for DED</i>	
J	5-11-67	33780	Title/ii, ivA/ivB, vii thru viiiA/viiiB, xxivE/xxivF, xxx, 2-41/2-42, 2-43, 3-2E thru 3-2H, 3-2K, 3-2S, 3-2V thru 3-2X, 3-2AB thru 3-2AE	EA <i>[Signature]</i>	
K	6-1-67	33872	Title/ii, ivA/ivB, vii thru viiiB, xii, xvi, xviA/xviB, 4-15 thru 4-17, 4-19, 4-21, 4-24 thru 4-26, 4-28 thru 4-29, 4-32 thru 4-34A/4-34B, 4-34C/4-34D, 4-34E/4-34F, 4-34G thru 4-34I/ 4-34J, 4-34K/4-34L, 4-34M thru 4-34O/4-34P, 4-34Q/4-34R, 4-34S thru 4-34U/4-34V, 4-34W/ 4-34X, 4-34Y/4-34Z, 4-34AA/ 4-34AB, 4-34AC thru 4-34AE/ 4-34AF, 4-34AG thru 4-34AI/ 4-34AJ, 4-34AK thru 4-34AM/ 4-34AN, 4-34AO thru 4-34AS/ 4-34AT, 4-34AU thru 4-34AW/ 4-34AX, 4-34AY thru 4-34BA/ 4-34BB, 4-35 thru 4-37, 4-41 thru 4-42B	EA <i>[Signature]</i>	

Record of Revisions

Rev. Let.	Date	TDRR Number	Pages Revised	Approval	
				CCB	NASA
L	6-15-67	33972	Title/ii, ivA, ivB, vii thru viiiB, xi, xii, xvi, xvii/xviii, xix, xx, xxivB, 3-2A, 3-2F, 3-2G, 3-2I thru 3-2J-4, 3-2V thru 3-2AB, 3-2AD, 3-2AE, 3-6, 3-6A/3-6B, 3-22, 3-27, 3-30 thru 3-38, 3-41 thru 3-47/3-48, 4-69 thru 4-71/4-72, 4-73/4-74 thru 4-82, 8-2	EA	<i>[Signature]</i>
M	8-17-67	34357	Title/ii, ivB, vii thru viiiB, xiii thru xiv, xx, xxivF, xxxi, 3-2A, 3-2F, 3-2G, 3-2J-4, 3-2K, 3-2P, 3-2P-1 thru 3-2P-3/3-2P-4, 3-2Q thru 3-2S, 3-2W, 3-2AB, 3-2AC, 3-2AF, 3-2AG/3-2AH, 3-8, 3-28, 3-44, 6-5 thru 6-6A/6-6B, 6-7 thru 6-8A/6-8B, 7-9, 7-10, 7-12, 7-16 thru 7-18, 7-20 thru 7-22A/7-22B, 7-24, 7-35/7-36, 7-49/7-50, 8-2 thru 8-2G/8-2H, 8-6A/8-6B, 8-7, 8-10, 8-10A/8-10B, A-1, I-1 thru I-15/I-16	EA	<i>[Signature]</i>

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

Record of Revisions

Rev. Let.	Date	TDRR Number	Pages Revised	Approval	
				CCB	NASA
N	9-21-67	34662	Title/ii, ivC/ivD, vii thru viiiB, xix, xxiv, xxivA, xxivD thru xxivG/xxivH, xxix, 2-47/ 2-48, 2-61/2-62, 2-63, 3-2A, 3-2D thru 3-2I, 3-2J-2 thru 3-2J-4, 3-2K, 3-2P-3/3-2P-4, 3-2R thru 3-2U, 3-2X thru 3-2AA, 3-2AD, 3-2AE, 3-2AG/ 3-2AH, 3-6A/3-6B, 3-8, 3-27, 3-34, 3-37, 6-5, 6-7, 7-12, 7-24, 7-27/7-28, 7-29/7-30, 8-6	EA	<i>[Signature]</i>
P	10-26-67	34958	Title/ii, ivC/ivD, vii thru viiiB, xix, xxivG, xxivH, 3-2A thru 3-2J-4, 3-2K thru 3-2L-2, 3-2M thru 3-2O/3-2P, 3-2P-1 thru 3-2P-5/3-2P-6, 3-2Q thru 3-2AG/3-2AH, 3-6, 3-6A/3-6B, 3-14, 3-15, 3-28, 7-9	EA	<i>[Signature]</i>
R	12-21-67	35270	Title/ii, ivC/ivD, vii thru viiiB, xxivG, xxivH, 3-2A, 3-2D, 3-2E, 3-2G thru 3-2I, 3-2J-2, 3-2J-3, 3-2K, 3-2L-2, 3-2M, 3-2P-2 thru 3-2P-4, 3-2R, 3-2S, 3-2U, 3-2AB, 3-2AC, 3-2AE thru 3-2AG/ 3-2AH	EA	<i>[Signature]</i>

Record of Revisions

Rev. Let.	Date	TDRR Number	Pages Revised	Approval	
				CCB	NASA
S	1-19-68	35466	Title/ii, ivC, ivD, vii thru viiiB, xix, xxiv thru xxivB, xxivF thru xxivH, xxvi, xxix/xxx, xxxi, xxxii, 2-47/2-48, 2-50, 2-53/2-54, 2-59, 2-61/2-62, 3-2A, 3-2L-3, 3-2L-4, 3-2M thru 3-2N-3/3-2N-4, 3-2O thru 3-2P-1, 3-7 thru 3-8D, 3-10, 3-10A/3-10B, 3-16A, 6-7, 6-8, 7-3, 7-4, 8-4A, 8-4B, 8-5, 8-6, 8-10A/8-10B, A-9	EA <i>4/5</i>	
T	2-8-68	35583	Title/ii, ivD, vii thru viiiB, xiii, xiv, xvii, xviii, xx, 7-1, 7-2, 7-24A/7-24B, 7-27/7-28, 7-33/7-34, 7-39/7-40, 7-41/7-42, 7-45/7-46, 7-47/7-48, 7-49/7-50, 7-51/7-52, 7-53/7-54, 7-55/7-56, 8-1, 8-2G/8-2H, 8-3/8-4, 8-6A/8-6B, 8-7, 8-8 and 8-9 deleted, 8-10, 8-10A/8-10B, 8-11/8-12 deleted	EA <i>CMK</i>	
U	3-7-68	35851	Title/ii, ivD, vii thru viiiB, xvi, xxivH, xxix/xxx, 3-2A, 3-2F, 3-2G, 3-2R, 3-2S, 3-7, 3-8A thru 3-8L, 3-9, 3-10, 3-11, 3-12A/3-12B, 3-24, 8-2C, 8-4A, 8-4B, 8-5, 8-6, A-3, A-13	EA <i>[Signature]</i>	

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

Record of Revisions

Rev. Let.	Date	TDRR Number	Pages Revised	Approval	
				CCB	NASA
V	5-9-68	36188	Title/ii, ivE/ivF, vii thru viiiB, xx, xxix/xxx, xxxi, 3-2B, 3-2C, 3-2J-3, 3-2J-4, 3-2K, 3-2L-2, 3-2L-3, 3-2P-2, 3-2P-3, 3-2AB, 3-2AC, 3-6A/ 3-6B, 3-15, 6-5 thru 6-6B, 6-7 thru 6-8B, 6-9, 6-10, 7-24A, 7-24B, 7-25/7-26, 7-29/7-30, 7-31/7-32, 7-35/ 7-36, 7-37/7-38, 7-51/7-52, 8-3/8-4, 8-6A/8-6B, 8-10	EA <i>EA</i>	
W	6-14-68	36345	Title/ii, ivE/ivF, vii thru viiiB, 7-27/7-28	SB <i>SB</i>	
Y	7-26-68	36596	Title/ii, ivE/ivF, vii thru viiiB, xi, xiii, xxivI/xxivJ, xxix/xxx, 3-2A thru 3-2C, 3-2J-3, 3-2J-4, 3-2K thru 3-2L-1, 3-2L-4, 3-2M thru 3-2N-1, 3-2O thru 3-2P-3, 3-2X thru 3-2AA, 3-10, 3-10A/ 3-10B, 6-8, 7-1 thru 7-2Q/7-2R, 7-24, 7-29/7-30, 7-35/7-36, 7-51/7-52, 8-10A/8-10B	EA <i>EA</i>	

Record of Revisions

Rev. Let.	Date	TDRR Number	Pages Revised	Approval	
				CCB	NASA
Z	11-6-68	36996	Title/ii, ivE thru ivF, vii thru viiiB, xxivI/xxivJ, 3-2A, 3-2F thru 3-2I, 3-2J-2, 3-2R thru 3-2S, 3-25, 7-24A thru 7-24B, 7-27/7-28, 7-29/7-30, 7-45/7-46, 8-2, 8-5 thru 8-6, 8-6A/8-6B, 8-7, 8-10 thru 8-10B	EA	<i>[Signature]</i>
AA	2-24-69	37366	Title/ii, ivF, vii thru viiiB, xxivI/xxivJ, xxxii, 1-15/1-16, 3-2A, 3-2B, 302I, 3-2K, 3-2L, 3-2N, 3-2N-1, 3-2O thru 3-2P-2, 3-2AA, 6-6, 6-8B, 7-4 thru 7-6, 7-25/7-26, 7-43/7-44	EA	<i>[Signature]</i>
AB	8-14-69	37754	Title/ii, ivF, vii, viii, xxivI/xxivJ, 3-2A, 3-2C, 3-2E, 3-2G thru 3-2I, 3-2J-3, 3-2J-4, 3-2K, 3-2L-2 thru 3-2L-4, 3-2M, 3-2N-3/3-2N-4, 3-2U, 3-2X, 3-2Z	EA	<i>[Signature]</i>
AC	6-22-70	38168	Title/ii, ivF, vii, viii, viiiA, xix, xxivI, xxivJ, 3-2A, 3-2F, 3-2G, 3-2J-4, 3-2K, 3-2L-4, 3-2M, 3-2N-3 thru 3-2N-6, 3-2O, 3-2P-2, 3-2P-3, 3-2AD, 3-2AE	EA	<i>[Signature]</i>

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

Record of Revisions

Rev. Let.	Date	TDRR Number	Pages Revised	Approval	
				CCB	NASA
AD	2-8-71	38312	Title/ii, ivG/ivH, vii thru viiiB, xxivJ, 3-2A, 3-2X thru 3-2AA	EA <i>[Signature]</i>	
AE	8-5-71	38454	Title/ii, ivG/ivH, vii, viii, xxivJ, 3-2A, 3-2D, 3-2E, 3-2R, 3-2S	EA <i>[Signature]</i>	
AF	7-6-72	38663	Title/ii, ivG/ivH, vii, viii, xxivI, xxivJ, 3-2A, 3-2L-4, 3-2M, 3-2N-2, 3-2N-3, 3-2AD, 3-2AE, 3-31, 3-32	JS <i>[Signature]</i>	
AG	11-24-72	38742	Title/ii, ivG/ivH, vii thru viii B, xxivK/xxivL, xxxi, 3-2A, 3-2D, 3-2E, 3-2X, 3-2Y, 3-2Z, 3-2AA, 3-2AB, 3-2AC, 3-6 3-6A/3-6B, 6-8B, 7-5, 7-41/7-42, 8-6A/8-6B, 8-7	EA <i>[Signature]</i>	

Table 1.1

Ref. No.	Description	Type	Approval	
			By	Date
1
2
3
4
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7
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BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

PREPARED BY

TITLE

DATE

Constance M. Todd
AC Electronics

Field Service Technical Writer

11 March 66

EXAMINED BY

TITLE

DATE

Don O'Dowd
AC Electronics

Publications Supervisor

11 March 1966

APPROVED BY

TITLE

DATE

G. Jettmar
AC Electronics

FIELD OPERATIONS

15 MARCH 1966

APPROVED BY

TITLE

DATE

L.S. Wilk by Abats
MIT/IL

Director STG

11 April 1966

APPROVED BY

TITLE

DATE

W. Kuyper
MIT/IL

CCB

11 April 66

APPROVED BY

TITLE

DATE

A. C. Metzger
NASA/MSC

chairman CCB

13 April 66

NO-1021843

FORM

FIELD INVESTIGATION REPORT AND CONTROL SYSTEM

DATE

TITLE

APPROVED BY

10/15/54
Investigation of ...
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BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

LIST OF EFFECTIVE PAGES
TOTAL NUMBER OF PAGES IN THIS PUBLICATION
IS 604 CONSISTING OF THE FOLLOWING:

Page No.	Rev.	Page No.	Rev.
Title	AG	xxxi	AG
ii Blank	Original	xxxii	AA
iii	D	1-1 thru 1-2	Original
iv	G	1-3	C
ivA	L	1-4 thru 1-14	Original
ivB	M	1-15	AA
ivC	S	1-16 Blank	Original
ivD	U	1-17 thru 1-21	Original
ivE	Z	1-22 Blank	Original
ivF	AC	2-1 thru 2-25	Original
ivG	AG	2-26 Blank	Original
ivH Blank	AD	2-27	Original
v	Original	2-28 Blank	Original
vi Blank	Original	2-29 thru 2-31	Original
vii thru viii	AG	2-32 Blank	Original
viiiA thru viiiB	AG	2-33	Original
ix	Original	2-34 Blank	Original
x	H	2-35 thru 2-39	Original
xi	Y	2-40 Blank	Original
xii	L	2-41	J
xiii	Y	2-42 Blank	Original
xiv	T	2-43	J
xv	G	2-44 thru 2-46	Original
xvi	U	2-47	S
xviA Added	K	2-48 Blank	Original
xviB Blank	K	2-49	Original
xvii thru xviii	T	2-50	S
xix	AC	2-51	Original
xx	V	2-52	A
xxi	Original	2-53	S
xxii Blank	Original	2-54 Blank	Original
xxiii	A	2-55	Original
xxiv thru xxivB	S	2-56 Blank	Original
xxivC Added	H	2-57	Original
xxivD thru xxivE	N	2-58 Blank	Original
xxivF thru xxivG	S	2-59	S
xxivH	U	2-60	Original
xxivI	AF	2-61	S
xxivJ	AF	2-62 Blank	Original
xxivK Added	AG	2-63	N
xxivL Blank	AG	2-64 thru 2-65	Original
xxv	G	2-66 Blank	Original
xxvi	S	2-67	A
xxvii thru xxviii Added	G	2-68 thru 2-83	Original
xxix	Y	2-84 Blank	Original
xxx Blank	S	3-1 thru 3-2	H
		3-2A	AG

LIST OF EFFECTIVE PAGES (CONT)

Page No.	Rev.	Page No.	Rev.
3-2B	AA	3-9	U
3-2C	AB	3-10 thru 3-10A	Y
3-2D	AG	3-10B Blank	S
3-2E	AG	3-11	U
3-2F	AC	3-12	A
3-2G	AC	3-12A	U
3-2H thru 3-2I	AB	3-12B Blank	A
3-2J thru 3-2J-1	P	3-13	A
3-2J-2	Z	3-14	P
3-2J-3	AB	3-15	V
3-2J-4	AC	3-16	F
3-2K	AC	3-16A	S
3-2L	AA	3-16B	F
3-2L-1	Y	3-17 thru 3-18	Original
3-2L-2 thru 3-2L-3	AB	3-19 thru 3-20	H
3-2L-4	AF	3-21	F
3-2M	AF	3-22	L
3-2N thru 3-2N-1	AA	3-22A Added	A
3-2N-2 thru 3-2N-3	AF	3-22B Blank	A
3-2N-4	AC	3-23	Original
3-2N-5 thru 3-2N-6 Added	AC	3-24	U
3-2O	AC	3-25	Z
3-2P thru 3-2P-1	AA	3-26	Original
3-2P-2 thru 3-2P-3	AC	3-27	N
3-2P-4	R	3-28	P
3-2P-5 Added	P	3-28A Added	A
3-2P-6 Blank	P	3-28B Blank	A
3-2Q	P	3-29	Original
3-2R thru 3-2S	AE	3-30	L
3-2T	P	3-31 thru 3-32	AF
3-2U	AB	3-33	L
3-2V thru 3-2W	P	3-34	N
3-2X thru 3-2AA	AG	3-35 thru 3-36	L
3-2AB thru 3-2AC	AG	3-37	N
3-2AD	AF	3-38	L
3-2AE	AE	3-39 thru 3-40 Added	A
3-2AF thru 3-2AG	R	3-41 thru 3-42	L
3-2AH Blank	H	3-43 Added	L
3-3 thru 3-4	H	3-44	M
3-5	Original	3-45 thru 3-47 Added	L
3-6	AG	3-48 Blank	L
3-6A	AG	4-1 thru 4-14	Original
3-6B Blank	L	4-15 thru 4-17	K
3-7	U	4-18	Original
3-8	S	4-19	K
3-8A thru 3-8D	U	4-20	Original
3-8E thru 3-8L Added	U	4-21	K
		4-22	Original
		4-23	F

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

LIST OF EFFECTIVE PAGES (CONT)

Page No.	Rev.	Page No.	Rev.
4-24 thru 4-26	K	4-58A Added	C
4-27	Original	4-58B Blank	C
4-28 thru 4-29	K	4-59 thru 4-60	Original
4-30 thru 4-31	Original	4-61 thru 4-64	C
4-32 thru 4-34	K	4-65 thru 4-66	Original
4-34A Added	K	4-66A Added	C
4-34B Blank	K	4-66B Blank	C
4-34C Added	K	4-67	C
4-34D Blank	K	4-68	Original
4-34E Added	K	4-69 thru 4-70	L
4-34F Blank	K	4-71 Added	L
4-34G thru 4-34I Added	K	4-72 Blank	L
4-34J Blank	K	4-73 Added	L
4-34K Added	K	4-74 Blank	L
4-34L Blank	K	4-75 thru 4-82 Added	L
4-34M thru 4-34O Added	K	5-1 thru 5-2	Original
4-34P Blank	K	5-3	F
4-34Q Added	K	5-4 thru 5-18	Original
4-34R Blank	K	6-1 thru 6-4	Original
4-34S thru 4-34U Added	K	6-5	V
4-34V Blank	K	6-6	AA
4-34W Added	K	6-6A thru 6-6B	V
4-34X Blank	K	6-7	V
4-34Y Added	K	6-8	Y
4-34Z Blank	K	6-8A	V
4-34AA Added	K	6-8B	AG
4-34AB Blank	K	6-9 thru 6-10	V
4-34AC thru 4-34AE Added	K	6-11 thru 6-14	C
4-34AF Blank	K	6-15 thru 6-17 Added	C
4-34AG thru 4-34AI Added	K	6-18 Blank	C
4-34AJ Blank	K	7-1 thru 7-2	Y
4-34AK thru 4-34AM Added	K	7-2A thru 7-2Q Added	Y
4-34AN Blank	K	7-2R Blank	Y
4-34AO thru 4-34AS Added	K	7-3	S
4-34AT Blank	K	7-4	AA
4-34AU thru 4-34AW Added	K	7-5	AG
4-34AX Blank	K	7-6	AA
4-34AY thru 4-34BA Added	K	7-7	G
4-34BB Blank	K	7-8	A
4-35 thru 4-37	K	7-9	P
4-38 thru 4-40	Original	7-10	M
4-41 thru 4-42	K	7-11	Original
4-42A thru 4-42B Added	K	7-12	N
4-43 thru 4-49	Original	7-13 thru 7-15	Original
4-50	C	7-16 thru 7-18	M
4-51	Original	7-19	A
4-52	C	7-20 thru 7-22A	M
4-53 thru 4-56	Original		
4-57 thru 4-58	C		

LIST OF EFFECTIVE PAGES (CONT)

Page No.	Rev.	Page No.	Rev.
7-22B Blank	A	8-1	T
7-23	Original	8-2	Z
7-24	Y	8-2A thru 8-2B Added	M
7-24A thru 7-24B	Z	8-2C	U
7-25	AA	8-2D thru 8-2F Added	M
7-26 Blank	Original	8-2G	T
7-27	Z	8-2H Blank	M
7-28 Blank	Original	8-3	V
7-29	Z	8-4 Blank	B
7-30 Blank	Original	8-4A thru 8-4B	U
7-31	V	8-5 thru 8-6	Z
7-32 Blank	Original	8-6A/8-6B	AG
7-33	T	8-7	AG
7-34 Blank	Original	8-8 thru 8-9 Deleted	T
7-35	Y	8-10 thru 8-10B	Z
7-36 Blank	Original	8-11 thru 8-12 Deleted	T
7-37	V	8-13	G
7-38 Blank	Original	8-14 Blank	B
7-39	T	A-1	M
7-40 Blank	Original	A-2	F
7-41	AG	A-3	U
7-42 Blank	Original	A-4 thru A-5	Original
7-43	AA	A-6	H
7-44 Blank	Original	A-7 thru A-8	Original
7-45	Z	A-9	S
7-46 Blank	D	A-10 thru A-11	Original
7-47	T	A-12	D
7-48 Blank	E	A-13	U
7-49	T	A-14 thru A-15	Original
7-50 Blank	E	A-16 Blank	Original
7-51	Y	B-1	Original
7-52 Blank	T	B-2 Blank	Original
7-53 Added	T	I-1 thru I-15 Added	M
7-54 Blank	T	I-16 Blank	M
7-55 Added	T		
7-56 Blank	T		

CONTENTS

Chapter		Page
1	SYSTEM TIE-IN	1-1
1-1	Scope	1-1
1-2	Apollo Mission and Program	1-1
	1-2.1 Earth Orbit	1-1
	1-2.2 Lunar Orbit	1-4
	1-2.3 Lunar Landing	1-4
1-3	Apollo Spacecraft	1-4
	1-3.1 Command Module	1-4
	1-3.2 Service Module	1-6
1-4	Spacecraft Systems	1-6
	1-4.1 Primary Guidance, Navigation, and Control System	1-6
	1-4.2 Stabilization and Control System	1-7
	1-4.3 Service Propulsion System	1-8
	1-4.4 Reaction Control Systems	1-8
	1-4.5 Electrical Power System	1-9
	1-4.6 Environmental Control System	1-9
	1-4.7 Communications and Instrumentation System	1-9
1-5	Primary Guidance, Navigation and Control System Interface	1-10
	1-5.1 Systems	1-10
	1-5.2 Controls and Displays	1-14
2	SYSTEM AND SUBSYSTEM FUNCTIONAL ANALYSIS	2-1
2-1	Scope	2-1
2-2	Primary Guidance, Navigation, and Control System	2-1
2-3	Block II PGNCS Axes	2-2
	2-3.1 Spacecraft Axes	2-2
	2-3.2 Inertial Axes	2-5
	2-3.3 Optical Axes	2-5

CONTENTS (CONT)

Chapter		Page
2-4	Inertial Subsystem	2-7
	2-4.1 Stabilization Loop	2-7
	2-4.2 Fine Align Electronics	2-11
	2-4.3 Accelerometer Loop	2-16
	2-4.4 IMU Temperature Control System	2-24
	2-4.5 Coupling Data Unit	2-31/2-32
	2-4.6 ISS Modes of Operation	2-36
2-5	Optical Subsystem	2-45
	2-5.1 Scanning Telescope	2-46
	2-5.2 Sextant	2-46
	2-5.3 SCT Servo Loops	2-49
	2-5.4 SXT Servo Loops	2-49
	2-5.5 Modes of Operation	2-51
2-6	Computer Subsystem	2-64
	2-6.1 Programs	2-69
	2-6.2 Machine Instructions	2-70
	2-6.3 Timer	2-73
	2-6.4 Sequence Generator	2-74
	2-6.5 Central Processor	2-75
	2-6.6 Priority Control	2-76
	2-6.7 Input-Output	2-78
	2-6.8 Memory	2-79
	2-6.9 Power Supplies	2-81
	2-6.10 Display and Keyboard	2-82
3	PHYSICAL DESCRIPTION	3-1
	3-1 Scope	3-1
	3-1A Block II Compatability	3-2A
	3-1B Astrosextant Passive Thermal Protective System (APTPS) Assembly	3-3
	3-2 Coupling Data Unit	3-3
	3-3 Command Module Computer Group	3-7
	3-3.1 Command Module Computer	3-7
	3-3.2 Display and Keyboard (DSKY)	3-10

CONTENTS (CONT)

Chapter		Page
3-4	Display and Control Group	3-10
	3-4.1 Indicator Control Panel	3-10
	3-4.2 Optics Shroud	3-10A/3-10B
3-4A	Dust Covers	3-16
3-5	Flexible Hose Assembly	3-16
3-5A	Eyepiece Storage Unit	3-16A
3-6	Inertial Measuring Unit and PIP Electronics Assembly	3-16B
	3-6.1 Inertial Measuring Unit	3-16B
	3-6.2 PIP Electronics Assembly	3-21
3-7	Navigation Base	3-23
3-8	Navigation Base, Optical Unit Assembly, and Bellows Installation Kit	3-24
3-9	Optical Unit Assembly	3-24
	3-9.1 Optical Base	3-24
	3-9.2 Sextant	3-24
	3-9.3 Scanning Telescope	3-24
3-9A	Optical Unit and Bellows Assembly. (PGNCS Part Number 2015000-021 and Above.)	3-24
3-10	PGNCS Interconnect Harness Group	3-27
3-11	Power and Servo Assembly	3-27
3-12	Signal Conditioner Assembly	3-42
4	COMPONENT THEORY OF OPERATION	4-1
4-1	Scope	4-1
4-2	Apollo II Inertial Reference Integrating Gyro	4-1
	4-2.1 Gyro Wheel Assembly	4-3
	4-2.2 Float Assembly	4-3
	4-2.3 Case	4-4
	4-2.4 Normalizing Network	4-4
	4-2.5 Apollo II IRIG Ducosyns	4-4
4-3	16 Pulsed Integrating Pendulum	4-10
	4-3.1 Float Assembly	4-13
	4-3.2 Housing Assembly	4-13
	4-3.3 Outer Case Assembly	4-13
	4-3.4 Normalizing Network	4-13
	4-3.5 PIP Ducosyns	4-13

CONTENTS (CONT)

Chapter		Page
4-4	Coupling Data Unit	4-15
4-4.1	Coarse System Module	4-15
4-4.2	Quadrant Selector Module	4-24
4-4.3	Main Summing Amplifier and Quadrature Rejection Module	4-29
4-4.4	Digital Mode Module	4-33
4-4.5	Read Counter Module	4-34S
4-4.6	Error Angle Counter and Logic Module	4-34AH
4-4.7	Interrogate Module	4-35
4-4.8	Digital to Analog Converter	4-37
4-4.9	Mode Module	4-41
4-5	Power Supplies	4-43
4-5.1	Pulse Torque Power Supply	4-44
4-5.2	-28 VDC Power Supply	4-46
4-5.3	ISS 800 CPS Power Supply	4-46
4-5.4	OSS 800 CPS Power Supply	4-48
4-5.5	3,200 CPS Power Supply	4-48
4-5.6	4 VDC Power Supply	4-49
4-6	Optical Unit Assembly	4-50
4-6.1	Scanning Telescope	4-50
4-6.2	Sextant	4-61
4-7	Command Module Computer	4-67
4-8	Signal Conditioner Assembly	4-70
4-8.1	Signal Conditioner Modules	4-70
4-8.2	Signal Conditioning Circuits	4-80
4-8.3	Reference Voltage Circuits	4-82
	PRELAUNCH AND IN-FLIGHT OPERATIONS	5-1
5-1	Scope	5-1
5-2	Preparation for Launch and Prelaunch IMU Alignment	5-1
5-3	Guidance Monitor	5-3
5-4	Orbital Navigation	5-6
5-4.1	Landmark Navigational Measurement	5-7
5-4.2	Star Horizon Navigational Measurement	5-7

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

CONTENTS (CONT)

Chapter		Page
5-5	In-Flight IMU Alignment	5-9
5-6	Midcourse Navigation	5-11
5-7	Midcourse Correction and Thrust Vector Control	5-14
5-8	Attitude Control	5-14
5-9	Entry	5-16
6	CHECKOUT AND MAINTENANCE EQUIPMENT	6-1
6-1	Scope	6-1
7	CHECKOUT	7-1
7-1	Scope	7-1
7-2	Primary Guidance, Navigation, and Control System	7-1
	7-2.1 Preparation	7-1
	7-2.2 Checkout	7-1
	7-2.3 Test Descriptions	7-1
7-3	Inertial Subsystem	7-2P
	7-3.1 Preparation	7-2P
	7-3.2 Checkout	7-2P
7-4	Optical Subsystem	7-2P
	7-4.1 Preparation	7-2P
	7-4.2 Checkout	7-2P
7-5	Computer Subsystem	7-2Q/7-2R
	7-5.1 Preparation	7-2Q/7-2R
	7-5.2 Checkout	7-2Q/7-2R
7-6	Signal Conditioner Assembly	7-2Q/7-2R
	7-6.1 Preparation	7-2Q/7-2R
	7-6.2 Checkout	7-2Q/7-2R
7-7	Pre-Installation Acceptance	7-2Q/7-2R
8	MAINTENANCE	8-1
8-1	Scope	8-1
8-2	Maintenance Concept	8-1

CONTENTS (CONT)

Chapter		Page
8-3	Malfunction Isolation	8-1
	8-3.1 Electrical Adapter Cable Assembly Set	8-2
	8-3.2 Arrangement of ND-1021040, Supplement B	8-2A
	8-3.3 Test Point Signal Characteristics	8-2B
	8-3.4 CSS Malfunction Isolation	8-2B
8-4	Removal and Replacement	8-2G/8-2H
8-5	Repair Verification	8-2G/8-2H
8-6	Deleted	
8-7	Pre-Power Assurance	8-10
8-8	Malfunction Verification	8-10
8-9	Malfunction Analysis	8-10
8-10	Maintenance Schedule	8-10
	8-10.1 Scheduled Maintenance	8-10
	8-10.2 Preventive Maintenance	8-10
8-11	Auxiliary Airborne Equipment	8-10
Appendix A	LIST OF TECHNICAL TERMS AND ABBREVIATIONS	A-1
Appendix B	RELATED DOCUMENTATION	B-1/B-2
Index	I-1

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

LIST OF ENGINEERING CHANGE PROPOSALS (cont)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
706	Vibration proof mounting in ESU	0106053	8106084	Y
724	OUA stop override	0102701	8102782	Y
725	Reidentification of lamp retainer panels as GSE	0102702	8102785	Y
526	Eyepiece vacuum shim			Z
702	GSE sextant and telescope cover modification			Z
704	Retrofit of fixed memory jumper modules	0104323	8104249	Z
719	Computer alarm module modification, V-fail detection	0104132	8104248	Z
735	DSKY IL and EL safety glass fix	0104135	8104251	Z
746	New anti-rattle leaf springs for pushbutton switches on Indicator Control panel			AA
752	Sunfilters for SCT eyepieces			AA
753	Incorporate steel threaded inserts into SXT Mirror Housings			AA
768	DSKY EL thermal/vacuum screen modification	0104138	8103954	AB
790	Addition of ESU finger tab and replacement of SXT eyepiece retainer bracket	0102704	8102790	AC

LIST OF ENGINEERING CHANGE PROPOSALS (cont)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
792	GNIC wiring change to allow computer aided optics tracking	-	8102791 8102792	AC
795	Addition of teflon locking rings to the SCT eyepiece assembly and the SXT eyepiece assembly	-	8106094 8106095	AC
796	Retrofit CM PSA's to prevent a reverse power problem	-	8102794 8102795	AC
815	Replace AGC connector assembly with a restart monitor	0102705	8102793	AC
1013	Refurbish OUA's by cleaning and relubricating the bearings and gears, adding new motor-tachs., and using new mounting techniques for SCT light rods.	-	8106097	AD
1017	Replace blower motor in IMU to increase reliability.	-	8102796	AE
1030	Replace ECDU modules containing 1010274 transformers to increase reliability.	-	-	AE
1038	Addition of heater to prevent eyepiece moisture during high humidity conditions	-	-	AF
1040	Modification of 800 hz 5% amplifier in PSA to eliminate amplifier oscillation during system turn-on	-	-	AF

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
1054	Modify G Harness to insure serviceability of cable, PN 1010707-1	0102706	8102798	AG
1055	Add RC filter to mode module and change transformer in D/A converter to prevent CDU digital counter bit changes caused by electrical transients	-	8102850	AG
1056	Revise OSS test procedure to accomplish Acceptance Testing at OSS level instead of using Precision Test Fixture	-	-	AG
1060	Install OUA SXT and SCT base harness assemblies of new design to eliminate possibility of intermittent circuits	0102707	8102799	AG

TO THE HONORABLE THE COMMISSIONERS OF THE GENERAL LAND OFFICE
FROM THE ATTORNEY GENERAL

RE: APPLICATION FOR A PATENT IN CONNECTION WITH THE
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YOUR LETTER OF THE 11TH INSTANT HAS BEEN RECEIVED

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ANDREW W. WOOD

ILLUSTRATIONS

Figure		Page
1-1	Primary Guidance, Navigation, and Control System	xxxii
1-2	Three Phases of Apollo	1-2
1-3	Apollo Spacecraft	1-3
1-4	PGNCS Functional Interface, Block Diagram	1-11
1-5	Main D and C Panel	1-15/1-16
2-1	PGNCS, Functional Block Diagram	2-3
2-2	Command Module and PGNCS Axes	2-4
2-3	Orientation of Optical Axes to Nav Base Axes	2-6
2-4	Inertial Subsystem, Block Diagram	2-8
2-5	Stabilization Loop, Block Diagram	2-9
2-6	Fine Align Electronics - Computer Inputs	2-12
2-7	Fine Align Electronics - Gyro Selection	2-13
2-8	Binary Current Switch	2-14
2-9	DC Differential Amplifier and Precision Voltage Reference	2-15
2-10	Accelerometer Loop	2-17
2-11	AC Differential Amplifier and Interrogator Module	2-19
2-12	Accelerometer Timing	2-20
2-13	PIPA Calibration Module	2-23
2-14	IMU Temperature Control System for PGNCS Part Number 2015000-011	2-25/2-26
2-15	IMU Temperature Control System for PGNCS Part Number 2015000-021	2-27/2-28
2-16	ISS-CDU Moding	2-33/2-34
2-17	IMU Cage Mode	2-41/2-42
2-18	OSS, Functional Block Diagram	2-47/2-48
2-19	Zero Optics and Manual Direct Modes	2-53/2-54
2-20	Image Presentations, Manual Direct Mode	2-57/2-58
2-21	SXT Image Presentations, Manual Resolved Mode	2-60
2-22	Manual Resolved Mode	2-61/2-62
2-23	Computer Subsystem Block Diagram	2-65/2-66
2-24	Program Organization	2-69
2-25	Timer Block Diagram	2-73
2-26	Sequence Generator Block Diagram	2-74
2-27	Central Processor Block Diagram	2-75
2-28	Priority Control Block Diagram	2-77
2-29	Input-Output Block Diagram	2-78
2-30	Memory Block Diagram	2-80
2-31	Power Supplies Block Diagram	2-81
2-32	Display and Keyboard Block Diagram	2-82

ILLUSTRATIONS (CONT)

Figure		Page
3-1	Primary Guidance, Navigation, and Control System	3-2
3-1A	Astrosextant Passive Thermal Protective System	3-4
3-2	Coupling Data Unit	3-4
3-3	CDU Module Locations	3-5
3-4	Command Module Computer, Part Numbers 2003200 and 2003993	3-7
3-5	Tray A Module Side	3-8
3-6	Tray B Module Side (CMC, Part Numbers 2003200 and 2003993)	3-9
3-7	Main Panel and Navigation Panel DSKY, Part Number 2003950	3-11
3-8	Indicator Control Panel	3-14
3-8A	Dust Covers	3-16
3-8B	Eyepiece Storage Unit	3-16A
3-9	Inertial Measuring Unit	3-16B
3-10	Stable Member	3-18
3-11	PIP Electronics Assembly	3-21
3-12	Navigation Base	3-23
3-13	Optical Unit Assembly, Front View	3-25
3-14	Optical Unit Assembly, Rear View	3-26
3-14A	PGNCS Interconnect Harness Group	3-28A/3-28B
3-15	Power and Servo Assembly	3-29
3-16	Operational Signal Conditioner Assembly	3-42
3-17	Flight Qualification Signal Conditioner Assembly	3-43
4-1	Apollo II IRIG, Simplified Cutaway View	4-2
4-2	Apollo II IRIG Normalizing Network	4-5
4-3	IRIG Signal Generator and Suspension Microsyn	4-6
4-4	IRIG Torque Generator and Suspension Microsyn	4-8
4-5	Ducosyn RLC Equivalent Circuit	4-10
4-6	Definition of 16 PIP Axes	4-11
4-7	Result of Acceleration Along Input Axes	4-12
4-8	PIP Torque Generator	4-14
4-9	Read Counter Relationship to Coarse and Fine Systems	4-16
4-10	Coarse System Module Block Diagram	4-17
4-11	Resolver Sine and Cosine Phase Relationship	4-18
4-12	Coarse Switch Circuit and Logic Equations	4-19
4-13	Coarse Switching Diagram	4-21
4-14	Quadrant Selector Module Block Diagram	4-25
4-15	Fine Switching Diagram	4-27
4-16	Main Summing Amplifier and Quadrature Rejection Module, Block Diagram	4-30

ILLUSTRATIONS (CONT)

Figure		Page
4-16A	Basic Storage and Counter Circuits, Logic Diagram	4-34A/4-34B
4-16B	Clock and Decoder Logic, Logic Diagram	4-34C/4-34D
4-16C	Countdown Circuit, Logic Diagram	4-34E/4-34F
4-16D	25.6 KPPS Generator	4-34G
4-16E	ISS and OSS I ₂ , I ₃ Generators, Logic Diagram	4-34I/4-34J
4-16F	ISS Moding Sync Logic, Logic Diagram	4-34K/4-34L
4-16G	OSS Moding Sync Logic, Logic Diagram	4-34O/4-34P
4-16H	ISS Ambiguity Logic, Logic Diagram	4-34Q/4-34R
4-16I	Read Counter Module, Block Diagram	4-34U/4-34V
4-16J	Read Counter Register, Logic Diagram	4-34W/4-34X
4-16K	Coarse and Fine Switch Logic, Logic Diagram	4-34AE/4-34AF
4-16L	Coarse and Fine Switch Logic, Switching Diagram	4-34AG
4-16M	Error Angle Counter and Logic Module, Block Diagram	4-34AI/4-34AJ
4-16N	Read Counter Logic Section, Logic Diagram	4-34AM/4-34AN
4-16O	Error Counter	4-34AS/4-34AT
4-16P	Error Counter Logic Circuits, Logic Diagram	4-34AW/4-34AX
4-17	Simplified 3 Bit Converter and Switch Configuration	4-38
4-17A	ISS Moding Buffers	4-42A
4-17B	Phase Buffers	4-42B
4-18	Pulse Torque Power Supply	4-45
4-19	-28 VDC Power Supply	4-46
4-20	800 CPS Power Supply	4-47
4-21	3,200 CPS Power Supply	4-49
4-22	4 VDC Power Supply, Block Diagram	4-50
4-23	SCT, Cutaway View	4-51
4-24	SCT Optics	4-53

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

ILLUSTRATIONS (CONT)

Figure		Page
4-25	Double Dove Prism	4-54
4-26	SCT Reticle	4-56
4-27	SCT Reticle Housing and Lamp Assembly	4-56
4-28	Pechan Prism	4-57
4-29	SCT Gearing Diagram	4-59
4-30	SCT Differential Gear Assembly	4-60
4-31	SCT Angle Counter	4-61
4-32	SXT, Cutaway View	4-62
4-33	SXT Optics	4-64
4-34	Beam Splitter Construction	4-65
4-35	SXT Reticle	4-66
4-36	SXT Reticle Housing and Lamp Assembly	4-66
4-37	SXT Optics Light Transmittance	4-68
4-38	SXT Gearing Diagram	4-69
4-39	Operational Signal Conditioner Assembly Block Diagram	4-71/4-72
4-40	Flight Qualification Signal Conditioner Assembly Block Diagram	4-73/4-74
5-1	Prelaunch Alignment Axes Orientation	5-2
5-2	Prelaunch IMU Alignment	5-4
5-3	Guidance Monitor	5-5
5-4	Orbital Navigation Sighting	5-6
5-5	Orbital Navigation	5-8
5-6	In-Flight IMU Alignment	5-10
5-7	Midcourse Navigation	5-12
5-8	Midcourse Position Determination	5-13
5-9	Midcourse Correction and Thrust Vector Control	5-15
5-10	Attitude Control	5-17
5-11	Entry	5-18
6-1	Universal Test Station Layout (KSC)	6-2
6-2	Universal Test Station Layout (MSC)	6-3
6-3	Universal Test Station Layout (NAA)	6-4
7-1	PGNCS Master Checkout Flowgram	7-25/7-26
7-2	PGNCS Checkout Preparation Flowgram	7-27/7-28
7-3	PGNCS Checkout Flowgram	7-29/7-30
7-4	ISS Master Checkout Flowgram	7-31/7-32
7-5	ISS Checkout Preparation Flowgram	7-33/7-34
7-6	ISS Checkout Flowgram	7-35/7-36
7-7	OSS Master Checkout Flowgram	7-37/7-38
7-8	OSS Checkout Preparation Flowgram	7-39/7-40
7-9	OSS Checkout Flowgram	7-41/7-42
7-10	CSS Master Checkout Flowgram	7-43/7-44
7-10A	CSS Checkout Preparation Flowgram	7-45/7-46
7-10B	CSS Checkout Flowgram	7-47/7-48
7-11	SCA Master Checkout Flowgram	7-49/7-50
7-12	IMU Pre-Installation Acceptance Flowgram	7-51/7-52
7-13	CDU Pre-Installation Acceptance Flowgram	7-53/7-54

ILLUSTRATIONS (CONT)

Figure		Page
7-14	PSA Pre-Installation Acceptance Flowgram	7-55/7-56
8-1	Maintenance Flowgram	8-3/8-4
8-2	Deleted	

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

TABLES

Number		Page
1-I	Spacecraft and Launch Vehicle Physical Characteristics	1-5
1-II	SCS Interface Signals	1-12
1-III	CIS Interface Signals	1-13
1-IV	S-4B Interface Signals	1-14
1-V	Displays and Controls	1-17
2-I	Instructions Classes	2-70
3-A1	Component and ECP Number Matrix	3-2A
3-I	APTPS Compatibility	3-2B
3-IA	CDU Compatibility	3-2D
3-IB	CMC Compatibility	3-2F
3-IC	DSKY Compatibility	3-2H
3-IC-1	CMC Installation Kit Compatibility	3-2J
3-IC-2	Jumper Module (Sundial 2021104) Compatibility	3-2J-2
3-IC-3	Rope Modules (Sundial) Compatibility	3-2J-3
3-ID	Indicator Control Panel Compatibility	3-2J-4
3-IE	Optics Shroud Compatibility	3-2L
3-IF	Dust Cover, SCT Compatibility	3-2L-2
3-IG	Dust Cover, SXT Compatibility	3-2L-3
3-IH	SCT Eyepiece Assembly (Adjustable Focus) Compatibility	3-2L-4
3-IJ	SCT Long Eye Relief Eyepiece Compatibility	3-2N
3-IK	SCT Prism Housing Assembly Compatibility	3-2N-2
3-IL	SXT Eyepiece Assembly Compatibility	3-2N-4
3-IM	SXT Long Eye Relief Eyepiece Compatibility	3-2N-6
3-IN	SXT Mirror Housing and Eyepiece Compatibility	3-2P
3-IP	Eyepiece Storage Unit Compatibility	3-2P-2
3-IQ	Flexible Hose Assembly (2 required) Compatibility	3-2P-4
3-IR	Interconnect Harness J	3-2Q
3-IS	IMU and PEA Compatibility	3-2R
3-IT	Nav Base Compatibility	3-2T
3-IU	Nav Base, Optical Unit and Bellows Installation Kit Compatibility	3-2V
3-IV	Optical Unit and Bellows Assembly Compatibility	3-2X
3-IW	OUA Compatibility	3-2Z
3-IX	PGNCS Interconnect Harness Group Compatibility	3-2AB
3-IY	PSA Compatibility	3-2AD
3-IZ	Signal Conditioner Assembly Compatibility	3-2AF
3-II	Function of CDU Modules	3-6
3-IIA	Computer Modules	3-8A
3-IIB	DSKY Modules	3-10A/3-10B
3-III	DSKY Operating Controls and Indicators	3-12
3-IV	Location and Functions of IMU Electronics	3-19
3-V	Location and Functions of PIP Electronics Assembly Modules	3-22
3-VA	PEA List Points	3-22A/3-22B
3-VI	PGNCS Interconnect Harness Group Connections	3-28
3-VII	Location and Functions of PSA Modules	3-30
3-VIII	PSA Test Points for Jack 45J1	3-39

TABLES (cont)

Number		Page
3-IX	PSA Test Points for Jack 45J3	3-40
3-X	Location and Functions of Operational SCA Modules	3-44
3-XI	Location and Functions of Flight Qualification SCA Modules	3-45
4-I	Circuits in SCA Modules	4-75
6-I	Checkout and Maintenance Test Equipment	6-5
6-II	Checkout and Maintenance Tools	6-10
6-III	List of Operating Procedure JDC's for GSE	6-11
7-I	Equipment Required for Checkout	7-3
7-II	PGNCS Interconnect Harness Group Connections (PGNCS Checkout)	7-7
7-III	G and N Mounting Fixture Harness Connections (PGNCS Checkout)	7-8
7-IV	GSE Interconnect Cable Connections (PGNCS Checkout)	7-8
7-V	GSE Interconnect Cable Connections (ISS Checkout)	7-14
7-VI	GSE Interconnect Cable Connections (OSS Checkout)	7-19
7-VII	GSE Interconnect Cable Connections (CSS Checkout)	7-23
7-VIII	PIA Test Requirements	7-24A
8-I	PGNCS, ISS, and OSS Loop Diagrams and Schematics	8-4A/8-4B
8-IA	MCD and Loop Diagram Selection	8-2C
8-II	CSS Logic Diagrams and Schematics	8-4A/8-4B
8-III	Removal and Replacement JDC's	8-6
8-IV	Retest Requirements	8-6A/8-6B
8-IVA	Retest Requirements	8-7
8-V	Deleted	
8-VI	List of PPA JDC's	8-13/8-14

LIST OF RELATED MANUALS

ND-1021038	Packing, Shipping and Handling Manual
ND-1021039	Auxilliary Ground Support Equipment Manual
ND-1021040	Bench Maintenance Ground Support Equipment Manual
ND-1021042	LEM Primary Guidance, Navigation and Control System Manual

EXHIBIT - ON

1981

EXHIBIT - ON

1981

EXHIBIT - ON

1981

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

LIST OF ENGINEERING CHANGE PROPOSALS

Engineering change proposals (ECP's) which affect this manual and from which pertinent data has been incorporated are listed below. Unless otherwise specified, the ECP number listed is an AC Electronics ECP number.

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Incorporated In Manual Revision
172	GSE Statement of Work Change	-	Basic
204	Block II Temperature Control Change	-	Basic
210	Block II Photometer and Tracker Modifi- cation	-	Basic
221	180 Degree Z IRIG Rotation	-	Basic
226	Aluminum to Magnesium Conversion of Computer and DSKY	-	A
252	Improved EL Alarm Light	-	A
254	Computer Multilayer Board Layout (MLB)	-	A
257	Redesign of Rope and Erasable Drivers	-	A
287	Command Module PSA Header Change	-	A
291	DSKY Alarm Panel Change	-	A

LIST OF ENGINEERING CHANGE PROPOSALS (CONT)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
324	Sense Amplifier Threshold Voltage Change	-	-	A
351	Alarm Module Temperature Stabilization of Warning Integrator and Improvement of Oscillator Fail Alarm	-	-	A
352	Sextant Anti-Creep Circuit	-	-	A
361	Exhibit D Statement of Work Changes	-	-	A
86	GSE Changes to AC Elec- tronics Statement of Work	-	-	B
248	Pulse Torque Power Supply Redesign	0102639	8102664	B
301	G and N Thermal Instrumen- tation for Vehicles 2TV-1 and LTA-8	-	-	B
362	Mounting Vibration Trans- ducers	-	-	B
373	Fabrication of Stainless Steel Bellows, Block II	0102642	8102667	B
388	Outgassing Protection, Block II and LEM	0102644 0102648 0102649 0102650 0102667	8102669 8102674 8102675 8102676 8102697	B

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

LIST OF ENGINEERING CHANGE PROPOSALS (CONT)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
398	ECP of Record to Change Block II ECDU Header Material for Reduced Weight and Improved Bonding of Cover	-	-	B
230	Incorporate Eyepiece Storage Unit	-	-	C
290	Incorporate Quick Disconnects for OUA Eyepieces	0102646	8102671	C
318	Corrosion Protection of Exposed Beryllium on OUA and AOT	-	-	C
346	Incorporate SXT and SCT LER Eyepieces	-	-	C
389	ECP of Record for Improved Protective Covers for Cables (PN 2014137-011 and PN 2014199-011)	-	-	E
461	Gimbal Servo Amplifier Module Diode Replacement	0102665 0102666	8102699 8102696 8102707	E
428	Ablative Shield Block II	0102673 0102674 0102675	8102704 8102705 8102706	F
457	Covers for Block II Switch Actuating Mechanisms	0102664	8102694 8102695	F
489	Replacement of Transformers in the Block II and LEM CDU	-	-	F

LIST OF ENGINEERING CHANGE PROPOSALS (CONT)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
149	Improved Receptacle Connector (Resolver Trim Module)	-	-	H
176	In-Process Vibration and Thermal Cycle of AGC Modules	-	-	H
210	Remove Tracker/Photometer	-	-	H
258	Redesign Power Supply Module - Block II	-	-	H
259	Redesign of Erasable Memory - Block II	-	-	H
287	Change Header Material from Cast to Wrought Magnesium	-	-	H
291	Alarm Lights	-	-	H
322	Retrofit of AGC 601	-	-	H
355	IRIG change	-	-	H
359	Replacement of IMU Mounting Bolts	0102643	8102668	H
367	Addition of Light Diffusing Paint to DSKY	-	-	H
368	Improved Power Supply Module Relays	-	-	H

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

LIST OF ENGINEERING CHANGE PROPOSALS (CONT)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Incorporated In Manual Revision
379	Rework Harness D Allow- ing Accessibility to PSA Hold Down Bolts	-	H
402	Clear Circuit Driver Modification	-	H
403	Strobe Adjustment	-	H
419	IDM Relay Replacement in DSKY	-	H
440	"Clear Rope" Driver Circuit Modification	-	H
443	Replacement of Short Screws	-	H
447	Incorporation of Plastic Pads Under Tray A&B Covers	-	H
452	Wiring Change to Accommodate Auxiliary Memory Unit	-	H
453	Beryllium Wedge Fix to Eliminate SXT Head Shift	-	H
460	Addition of Jumper Wires in Tray A	-	H
470	Random Workmanship Vibration Blk II AGC and DSKY	-	H
474	Manufacture Test Connector Jumpers for Blk I-100, Blk II and LEM Computers to Ground Certain Gate Inputs	-	H

LIST OF ENGINEERING CHANGE PROPOSALS (CONT)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
475	Chamfer of Telescope Adjust Seal Seat	-	-	H
476	Painting of Alarm Indicator Face	-	-	H
478	Paint Exposed Surfaces on Mid Tray Spacer	-	-	H
479	DSKY Teflon Coated Push-button Shaft	-	-	H
485	Redesign Power Supply to Remove 28 VDC Regulator	-	-	H
486	Cut Pins on AGC Power Supply to Remove 28 VDC Regulator	-	-	H
493	Y-Line Feedback of Base Resistor	-	-	H
494	DSKY Wiring Improvement	-	-	H
499	Addition of ECDU Modules Damper Plate	-	-	H
500	Replacement of PIP Pre-amplifier Capacitors and IMU Conformal Coating	0102678 0102680	8102709 8102713	H
501	Implementation of Flight Processing Spec	-	-	H
505	Implementation of Flight Processing Spec and New Diode	-	-	H
507	Modification of Solvere Motor-Tach in OUA	-	-	H

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

LIST OF ENGINEERING CHANGE PROPOSALS (CONT)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
515	Modify Coarse System Modules	-	-	H
518	Standby Change on Com- puter	0104121	8104233	H
529	Humidity Qualification Fix on PSA	0102682 0102683	8102724 8102725	H
67	Replacement of metallic bellows for Block II with rubber spacecraft vacuum seals	-	-	J
148	ECDU transformer change	-	-	J
149	Improved receptacle connector (resolver trim module)	-	-	J
191	CDU electronics module change	-	-	J
217	Delete signal conditioner power supply assembly	-	-	J
307	Middle axis assembly clamp changes	-	-	J
308	Stable member heat transfer change	-	-	J
309	PIP temp deviation re- duction and temp alarm test	-	-	J
310	IMU cross coupling change	-	-	J

LIST OF ENGINEERING CHANGE PROPOSALS (cont)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
358	ECP of record for redesign of handling fixture	-	-	M
509	ECP of record for AGC/GSE compatibility	0104107	-	M
536	ECP of record computer diagnostic capability extension	-	-	M
538	Replacement of reticle variable brightness resistor	-	-	M
551	Replace G & N harness lacing tape	-	8102731	M
585	PIP preamplifier capacitor replacement	-	8102732 8102734	M
605	IMU blanket removal	0102687	8102733 8102734	M
506	Modification of AGC handling fixture	-	-	N
569	GNIC panel modification	-	8102736	N
581	Interconnect cable set			N
587	IRIG end cap assembly	-	-	N
590	Removal of anti-creep protection	-	8102739 8102740	N
597	CSM-101	-	8102736 8102742	N

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

LIST OF ENGINEERING CHANGE PROPOSALS (cont)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
603	Capacitor replacement in MSA and quadrature reject module	-	8102741 thru 8102747	N
609	Elimination of ECDU DAC saturation	-	-	N
620	Flammability fix	0102679	8102752	N
621	Flammability fix for PSA	0102689	8102755	N
623	Flammability fix for SCA	-	8102731	N
630	Flammability fix for Nav Base	0102688	8102750	N
564	Implementation of flat pack specifications ND 1002359A & 1002358B	-	-	P
572	PSA gasket replacement	0102693	8102758	P
612	Retrofit G & N System 202 (CSM 098)	0102696	8102765	P
616	CSM 103 G & N 208	-	-	P
627	Modification of Block II harness group for flammability protection	-	-	P

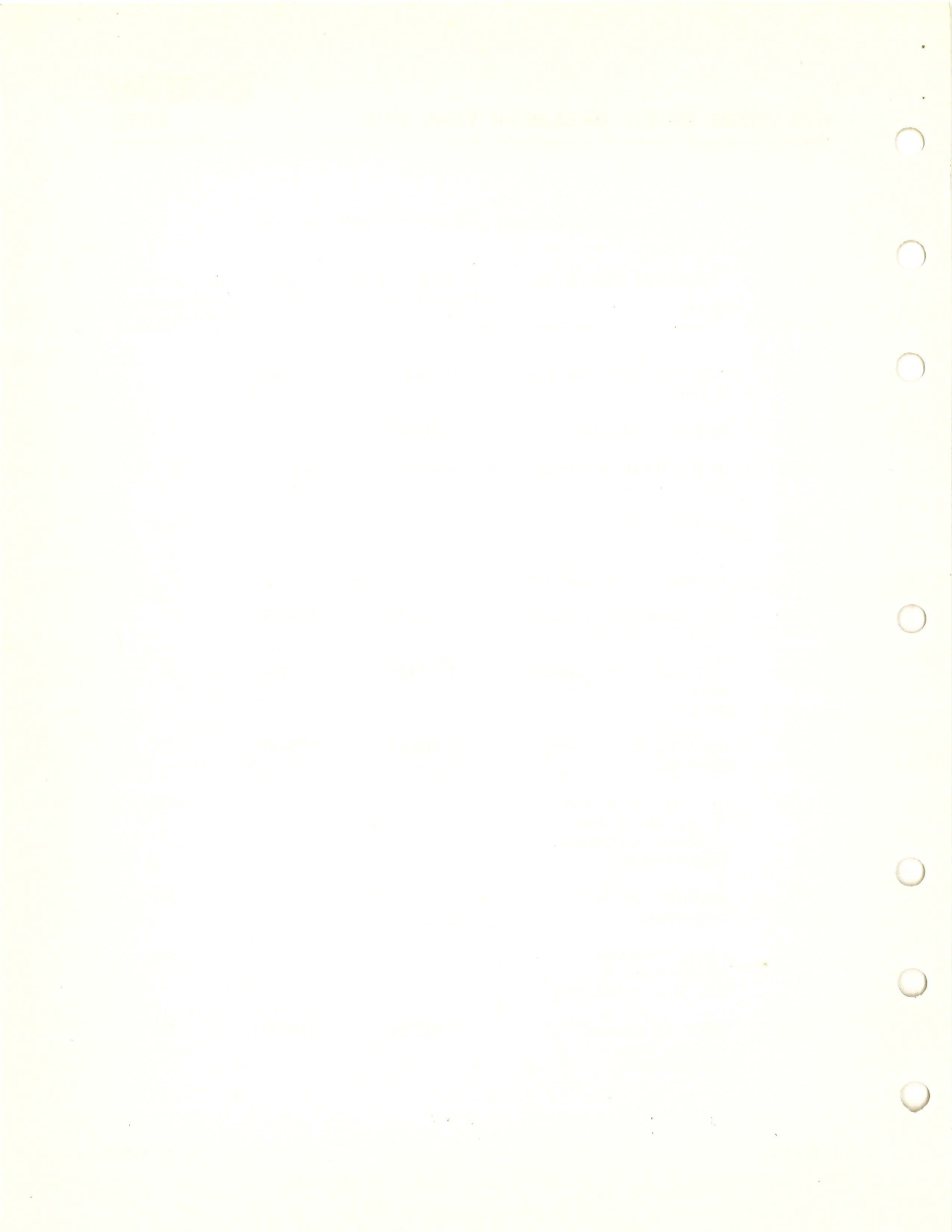
LIST OF ENGINEERING CHANGE PROPOSALS (cont)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
628	ESU redesign as result of flammability study	-	-	P
629	Optics shroud redesign	0102694	8102759	P
631	Replacement of RTV-102 with RTV-109	-	-	P
653	Modification of IMU wiring to reduce IRIG pre-amp oscillation	-	-	P
606	G & N 207-211 system change	-	-	R
641	Non-metallic materials modification for DSKY	0104126	8104241	R
667	CM harness "A" re- design for VHF ranging	-	8102769	R
673	DSKY pushbutton cap housing assembly leaf spring	-	-	R
678	End cap mounting hard- ware torque change to Apollo II IRIG	-	8102768	R
595	Potentiometers from 2K to 5K in PTC and LTC	-	-	S
632	CM OUA eyepiece assem- blies flammability modi- fications	-	-	S
604	AGC erasable memory unit	-	-	U
688	Modification of IMU to reduce sporadic oscilla- tion of IRIG preamp	-	8102773 8102774	U

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

LIST OF ENGINEERING CHANGE PROPOSALS (cont)

ECP No.	Functional Description	Retrofit Instruction Bulletin (RIB) No.	Kit No.	Incorporated In Manual Revision
706	Vibration proof mounting in ESU	0106053	8106084	Y
724	OUA stop override	0102701	8102782	Y
725	Reidentification of lamp retainer panels as GSE	0102702	8102785	Y
526	Eyepiece vacuum shim			Z
702	GSE sextant and tele- scope cover modification			Z
704	Retrofit of fixed memory jumper modules	0104323	8104249	Z
719	Computer alarm module modification, V-fail detection	0104132	8104248	Z
735	DSKY IL and EL safety glass fix	0104135	8104251	Z
746	New anti-rattle leaf springs for pushbutton switches on Indicator Control panel			AA
752	Sunfilters for SCT eyepieces			AA
753	Incorporate steel threaded inserts into SXT Mirror Housings			AA
768	DSKY EL thermal/ vacuum screen modification	0104138	8103954	AB



WARNING AND CAUTION PAGE

The following warnings and cautions apply to the system for which this manual was written. These warnings and cautions must be observed to avoid personal injury and/or equipment damage.

CHEMICAL HAZARDS**WARNING**

Beryllium is a highly toxic metal. Processes such as deburring, machining, or grinding of beryllium, if expressly authorized, as well as other procedures that produce airborne beryllium particles, should be performed with local exhaust ventilation which is connected with an acceptable exhaust-filtration system. The inhalation of airborne beryllium particles can cause serious respiratory disability. All injuries where the skin is damaged, including puncture wounds and skin rashes, should be reported immediately so that proper treatment measures can be taken. Air in these areas should be monitored to insure that the amount of airborne beryllium remains below the established threshold limit value.

CAUTION: Wear nylon gloves or finger cots while handling beryllium parts unless parts are to be immediately cleaned after handling. Moisture and oils from the skin have a highly corrosive effect on beryllium.

WARNING AND CAUTION PAGE (Continued)

WARNING

When using equipment containing mercury (i.e. battery power pack; auxiliary battery pack; differential voltmeter, Fluke 803B; or mercury pool used with alignment mirror assembly) avoid spilling mercury. Mercury is a toxic metal and may contaminate equipment, clothing and body. After use of such equipment, visually inspect area and equipment to verify absence of mercury. If mercury contamination exists, see JDC 18086.

WARNING

Do not leave mercury in open containers. Keep in tightly closed unbreakable container when not in use, since mercury vapor is toxic.

CAUTION: Avoid spilling mercury from open mercury pool. Liquid or vaporous mercury can cause severe corrosion and/or cracking of unprotected aluminum and other non-ferrous metals. If mercury contamination of equipment exists, report to AC Electronics site manager and site quality control representative.

WARNING AND CAUTION PAGE (Continued)

WARNING

Wear goggles, rubber gloves, and rubber aprons when working with acid solutions. Do not spatter solution. If acid contacts skin or eyes, immediately flush affected area with mild boric acid solution or water and report to nearest medical facility.

WARNING

Do not spatter battery cell electrolyte. If electrolyte contacts skin or eyes, immediately wash affected area with mild boric acid solution or water and report to nearest medical facility.

CAUTION: Always add acid to water; do not pour water into acid. Equipment surface can be damaged from uncontrolled chemical reaction. However, distilled water, if required, may be slowly added to replenish the electrolyte of a battery.

CAUTION: Keep battery cells upright, unless otherwise noted, to prevent loss of electrolyte.

WARNING AND CAUTION PAGE (Continued)

CAUTION: Add only distilled water to nickel cadmium battery cells. Never add acid or use instruments that have been used to fill lead acid batteries. Acid will damage the plates of the nickel cadmium cells.

CAUTION: Prevent acid from contacting painted surfaces. Immediately wipe off any acid on a painted surface, since prolonged contact will cause damage.

ELECTRICAL HAZARDS

WARNING

Use extreme care when reaching inside the G and N coolant and power console. Lethal voltages exist at power line filters and at the power and signal input panel.

WARNING

Always use moisture-proofing protection (covers, rubber seals, gaskets, and sealing compound) for PSA connectors and pins not secured to mating connectors. Otherwise, connector and pin corrosion may cause short circuits.

CAUTION: Maintain IMU heater power at all times, except during use of IMU shipping container, to prevent damage to IMU.

CAUTION: Handle G and N harnesses with care to avoid damage to connectors, conductor wires, and/or potting.

CAUTION: Turn off equipment when any power connections are being made to prevent pin damage due to arcing.

WARNING AND CAUTION PAGE (Continued)

CAUTION: Exercise extreme care when removing and installing electrical connectors. Visually check all pins for straightness. Where applicable, use special gages and reticles to check pin alignment during connector removal or installation. Install or remove connectors with jack screws by turning each screw one turn at a time to maintain parallelism between plug and connector during entire operation. Otherwise, bent pins can cause short circuits and additional equipment damage. Reference JDC 18080 for straightening bent pins.

CAUTION: Do not use jumper module part number 2003076-011 in slots B5 and B6 of "C" computers. Use of the module can damage the "C" computers.

MECHANICAL HAZARDS

WARNING

Do not extend any equipment cabinet drawer slides until drawers are to be installed. The extended slides can cause serious injury to personnel.

WARNING

Do not extend more than one cabinet drawer at a time. Extension of more than one drawer at a time can cause console to topple, endangering personnel and equipment.

CAUTION: Do not exercise SXT and SCT manual drives located on SCT panel. Excessive and improper use of manual drives can damage positioning capabilities of optics.

CAUTION: Do not manually position optics gear train output shafts (such as SXT index mirror). Overloading damages gear teeth and bearings.

OPTICAL HAZARDS

CAUTION: Always keep lens covers on optical equipment when it is not in use to protect optically ground and coated surfaces from damage.

1957-58

NEW YORK STATE DEPARTMENT OF TAXATION

INVESTMENT IN REAL ESTATE

SECTION 179(b)(1)(B)

SECTION 179(b)(1)(C)

SECTION 179(b)(1)(D)

SECTION 179(b)(1)(E)

SECTION 179(b)(1)(F)

SECTION 179(b)(1)(G)

SECTION 179(b)(1)(H)

SECTION 179(b)(1)(I)

SECTION 179(b)(1)(J)

SECTION 179(b)(1)(K)

SECTION 179(b)(1)(L)

SECTION 179(b)(1)(M)

SECTION 179(b)(1)(N)

SECTION 179(b)(1)(O)

SECTION 179(b)(1)(P)

SECTION 179(b)(1)(Q)

SECTION 179(b)(1)(R)

SECTION 179(b)(1)(S)

SECTION 179(b)(1)(T)

SECTION 179(b)(1)(U)

SECTION 179(b)(1)(V)

SECTION 179(b)(1)(W)

SECTION 179(b)(1)(X)

SECTION 179(b)(1)(Y)

SECTION 179(b)(1)(Z)

SECTION 179(b)(1)(AA)

SECTION 179(b)(1)(AB)

SECTION 179(b)(1)(AC)

SECTION 179(b)(1)(AD)

SECTION 179(b)(1)(AE)

SECTION 179(b)(1)(AF)

SECTION 179(b)(1)(AG)

SECTION 179(b)(1)(AH)

SECTION 179(b)(1)(AI)

SECTION 179(b)(1)(AJ)

INTRODUCTION

This manual provides information necessary for checkout, maintenance, and repair of the command module Block II primary guidance, navigation, and control system (PGNCS) (figure 1-1). The manual provides an introduction and complete familiarization with the PGNCS. Included in the manual are system tie-in, functional analysis, component description, detailed theory of operation, and description of flight operations.

Job Description Cards (JDC's) containing detailed step-by-step procedures are contained in separate supplementary volumes. This manual lists support equipment required for the JDC procedures and provides tabular and diagram references to the JDC's to insure that the preparation for checkout, checkout analysis, and repair and replacement procedures are performed in sequence and conform to all program requirements.

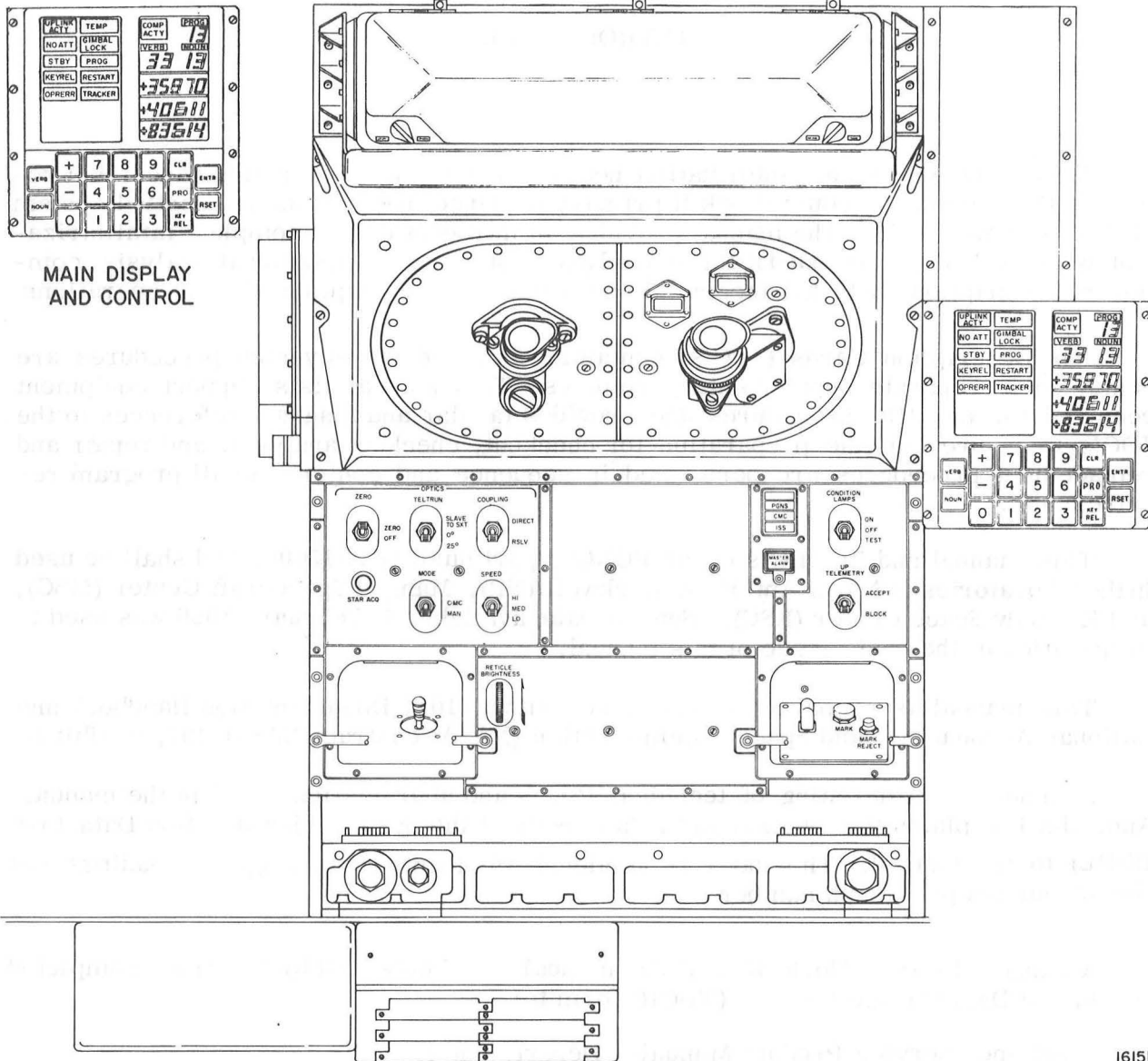
This manual and its JDC's cover PGNCS, part number 2015000, and shall be used in the laboratories at North American Rockwell (NR), Manned Spacecraft Center (MSC), and Kennedy Space Center (KSC). Source data available 4 February 1966 was used in preparation of the basic issue of this manual.

This manual is prepared in accordance with E-1087 Documentation Handbook and National Aeronautics and Space Administration (NASA) contract NAS 9-497, exhibit D.

Appendix A is a listing of technical terms and abbreviations used in the manual. Appendix B explains the function and relationship of the System Identification Data List (SIDL) to the manual. The index is an alphabetical listing of paragraph headings with the associated paragraph number.

Changes to the Block II PGNCS manual are requested by sending a completed Technical Data Change Request (TDCR) form to:

Customer Service Product Manuals, Department 47
Delco Electronics Division
Santa Barbara Operations, GMC
6767 Hollister Avenue
Goleta, California 93017



16151D

NAVIGATION STATION
(LOWER DISPLAY AND CONTROL)

Figure 1-1. Primary Guidance, Navigation, and Control System

Chapter 1

SYSTEM TIE-IN

1-1 SCOPE

This chapter presents the concepts of the Apollo mission. The chapter also describes the functional interface between the Block II primary guidance, navigation, and control system (PGNCS) and the other spacecraft systems.

1-2 APOLLO MISSION AND PROGRAM

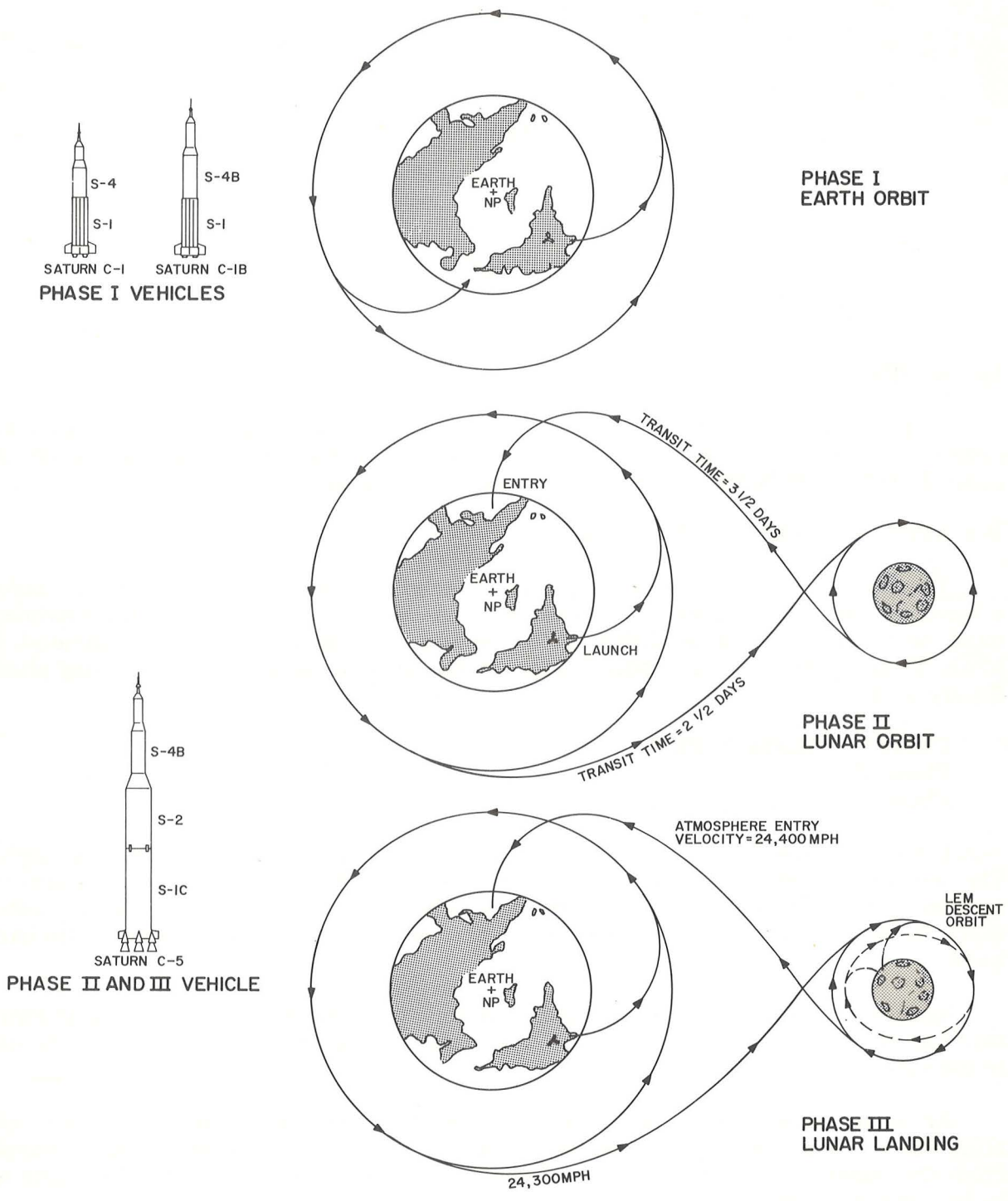
The purpose of the Apollo program is to land two astronauts on the moon. Before a lunar landing is attempted, the spacecraft hardware must be proven, the astronauts must be trained, and a capability for extended space flight must be demonstrated. To attain these objectives the Apollo program has been divided into the following phases (figure 1-2):

- Phase I Earth Orbit.
- Phase II Lunar Orbit.
- Phase III Lunar Landing.

1-2.1 EARTH ORBIT. Phase I consists of unmanned and manned earth orbital flights. The unmanned flights are designed for performance and interface testing of spacecraft systems under flight conditions. Manned earth orbital flights are designed for the astronauts to develop navigation, rendezvous, and entry techniques using the Apollo hardware.

Rendezvous practice in earth orbit will develop lunar excursion module (LEM) (a two man vehicle for lunar landing) separation and docking techniques which will be used in the lunar landing phase.

Entry practice in earth orbit requires acceleration of the spacecraft into a parabolic entry trajectory to simulate the flight path and speed of the spacecraft returning from the lunar orbit or lunar landing phases. Earth landing for all Apollo flights will be by parachute descent of the command module (figure 1-3).



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Figure 1-2. Three Phases of Apollo

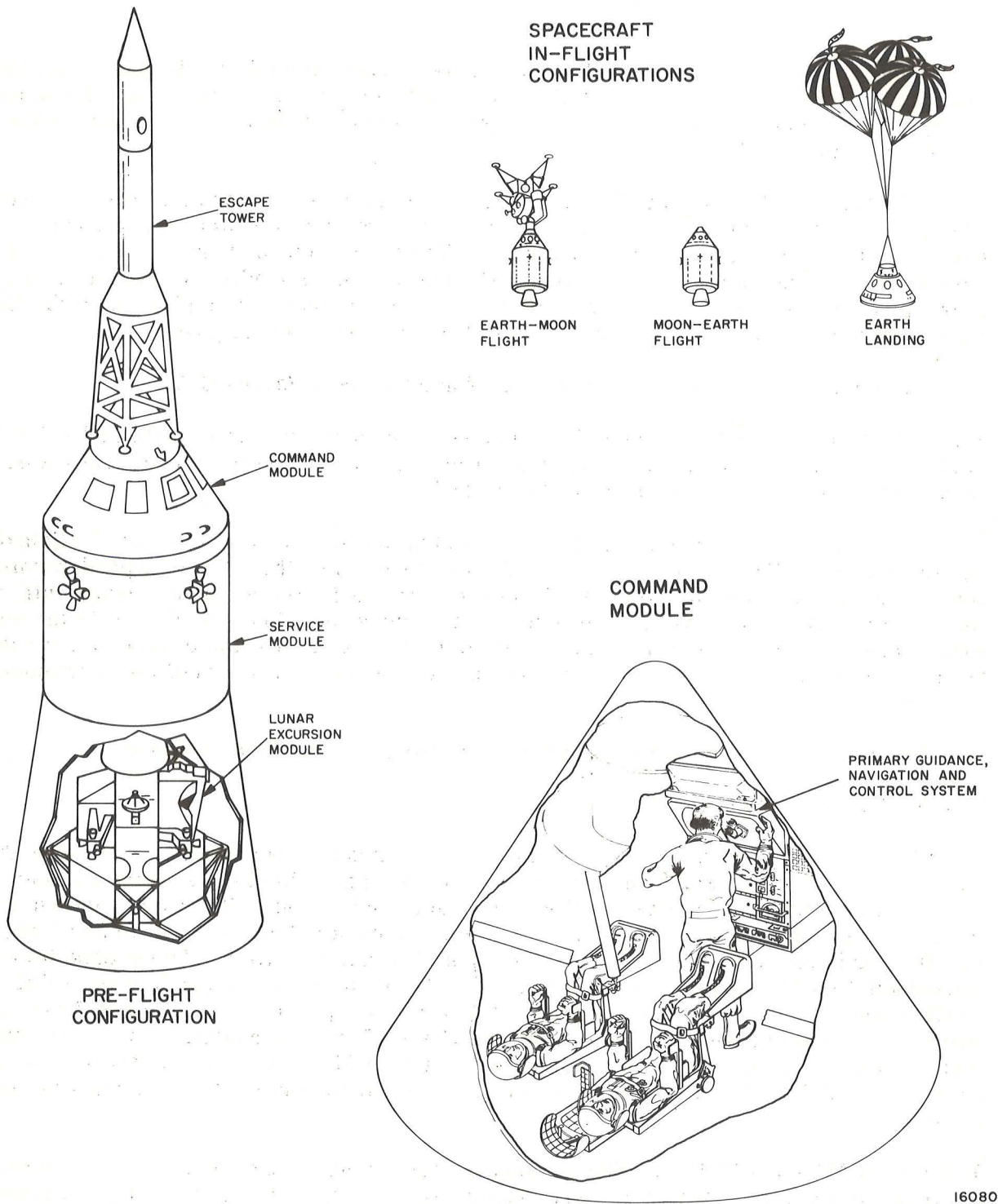


Figure 1-3. Apollo Spacecraft

Phase I will use two launch vehicles. The two stage Saturn C-1 will place the spacecraft and a three man crew into earth orbit. The larger two stage Saturn C-1B will orbit the spacecraft and LEM simultaneously. Physical characteristics of the Apollo vehicles are given in table 1-I.

1-2.2 LUNAR ORBIT. Phase II is a manned flight around the moon and return to earth. The flight requires about six and one half days and employs techniques of navigation and entry practiced during the earth orbit phase. The astronauts will make scientific observations during the lunar voyage, and will view the lunar landscape from a relatively close position. These observations, combined with astronomical and lunar probe findings, will determine a suitable landing site for the lunar landing phase.

The launch vehicle for the lunar orbit phase will be a Saturn C-5.

1-2.3 LUNAR LANDING. Phase III consists of landing two astronauts and scientific equipment on the moon, performing scientific experiments on the moon, and returning lunar samples and the three astronauts to earth.

The flight requires about eight days. Landing on the moon will be accomplished by separating the LEM from the spacecraft and deorbiting the LEM to a predetermined lunar landing site. Two astronauts will descend in the LEM and one astronaut will remain in the spacecraft orbiting the moon. Rendezvous techniques perfected in the earth orbit phase will be applied when the LEM returns to lunar orbit and the two astronauts transfer to the command module. The spacecraft then will be injected into a transearth trajectory.

The Saturn C-5 will be the launch vehicle for the lunar landing phase.

1-3 APOLLO SPACECRAFT

The spacecraft (figure 1-3) consists of a command module and a service module. The LEM is attached to the top of the S-4B stage behind the service module, and an escape tower is mounted atop the command module. Solid fuel motors in the escape tower inject the command module into an escape trajectory if an abort occurs on the pad or any time during the first 3 minutes of flight. The escape tower is jettisoned shortly after a satisfactory launching and only the spacecraft, LEM, and S-4B stage are inserted into earth orbit. After the spacecraft is injected from earth orbit into a lunar trajectory, the LEM and the S-4B are separated from the command and service module. The command module and LEM dock and the S-4B is separated from the LEM. The LEM is carried to lunar orbit where it is detached for lunar landing.

1-3.1 COMMAND MODULE. The command module contains crew accommodations, all controls and displays which the astronauts use to control the spacecraft, and systems for spaceflight and earth entry.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

Table 1-I. Spacecraft and Launch Vehicle Physical Characteristics

Vehicle	Stage	Nominal Thrust (pounds)	Propellants and Oxidizers	Height (feet)	Diameter (feet)
Spacecraft	Command module	N/A	Aerozine 50 Nitrogen te- troxide	12	13
	Service module	21,900	Aerozine 50 Nitrogen te- troxide	23	12.8
	LEM	10,000 on landing 4,000 on takeoff	Aerozine 50 Nitrogen te- troxide	17	10
Saturn C-1	First (S-1)	1,500,000	Kerosene Liquid oxygen	81.6	21.6
	Second (S-4)	90,000	Liquid hydrogen Liquid oxygen	40	18
Saturn C-1B	First (S-1)	1,500,000	Kerosene Liquid oxygen	81.8	21.6
	Second (S-4B)	200,000	Liquid hydrogen Liquid oxygen	60	22
Saturn C-5	First (S-1C)	7,500,000	Kerosene Liquid oxygen	138	33
	Second (S-2)	1,000,000	Liquid hydrogen Liquid oxygen	82	33
	Third (S-4B)	200,000	Liquid hydrogen Liquid oxygen	60	22

The command module has a heat shield for entry into the earth's atmosphere, parachutes for final descent, and a reaction control system for attitude control after separation from the service module. The command module has several windows for general observation, rendezvous, horizon reference and earth landing reference.

The PGNCS is located in the command module lower equipment bay area, frequently referred to as the navigation station. Additional displays and controls associated with the operation of the PGNCS are located on the main display and control (D and C) panel above the astronaut couches.

1-3.2 SERVICE MODULE. The service module houses a propulsion system, reaction control system, radar, fuel cells, and other gear and supplies that do not need to be pressurized. Until separation occurs just prior to entry, the service module provides the spacecraft with thrust, attitude control, and electrical power.

1-4 SPACECRAFT SYSTEMS

Functionally, there are seven spacecraft systems associated with the command and service module. Four of these systems are used for control of the spacecraft. The PGNCS or the stabilization and control system (SCS) receive manual inputs from the crew and electrical inputs from inertial sensors to generate commands that result in rotation and translation maneuvers. The reaction control system (RCS) or service propulsion system (SPS) provide all the external forces and mechanical couples to maneuver the spacecraft under the control of the PGNCS or the SCS. The crew obtains information from the on board computer (part of the PGNCS), Manned Space Flight Network (MSFN), or displays that indicate the necessity to initiate one or more of the basic spacecraft motions. The remaining three spacecraft systems are indirectly related to spacecraft control in that they provide power through the electrical power system (EPS), environment through the environmental control system (ECS), and communications through the communications and instrumentation system (CIS).

The PGNCS can also be considered a functional system of the S-4B stage since the astronaut can select the PGNCS as the guidance source for S-4B thrusting. The PGNCS uses manual inputs from the crew, electrical inputs from inertial sensors, and S-4B interface signals to generate steering and thrust commands to the S-4B instrumentation unit (IU).

1-4.1 PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM. The PGNCS performs three basic functions: inertial guidance, optical navigation, and spacecraft control. The PGNCS makes astronaut manual inputs, reaction control system operation, service propulsion system operation and spacecraft operation compatible.

The inertial measuring unit (IMU) is the primary element used for inertial guidance. It consists of three gyros and three accelerometers mounted on the innermost member of a three degree-of-freedom gimbal structure. Angular orientation of this inner platform is obtained from resolvers mounted on the gimbals. The IMU senses velocity and attitude changes and provides these changes to a computer.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

For optical navigation, a scanning telescope and sextant are used to take sightings on celestial bodies and landmarks. The sightings are used to determine spacecraft position and velocity, and to establish proper alignment of the stable member. Communications with ground tracking stations also provide information for spacecraft navigation.

A digital computer, command module computer (CMC) in the PGNCS is the primary spacecraft data processing element. The CMC, which contains a catalog of celestial objects, is programmed to calculate thrust and steering commands using information obtained from optical sightings and from the IMU. The computer issues on/off thrust discretetes to the SPS, RCS, and S-4B IU. Data on gravitational fields and celestial movements are used in the CMC to calculate future position and velocity of the spacecraft.

The PGNCS provides the following spacecraft capabilities:

(1) Rate stabilization of command module during high altitude abort, after separation from the booster, stabilization of the command module after escape tower jettison, and during entry.

(2) Attitude and stabilization control about three axes in response to either manual or automatic rotation commands.

(3) Translational control along three axes in response to manual commands for ullage maneuvers, transposition, and docking.

(4) Translational control along three axes in response to manual commands as a backup mode to the LEM-controlled rendezvous and docking operation.

(5) Translational control along three axes in response to manual commands for small changes in velocity.

(6) Thrust vector control for the SPS.

(7) SPS engine thrust on/off functions.

(8) Guidance control of S-4B engine.

(9) Displays for visual indication of control parameters.

1-4.2 STABILIZATION AND CONTROL SYSTEM. The SCS, as a backup system to the PGNCS, can sense and control spacecraft attitude and velocity changes during flight.

The SCS is located in the command module and consists of a reaction jet and propulsion solenoids driver assembly, electronics control assembly, electronics display assembly, gyro display coupler, thrust vector position servo amplifier, two gyro assemblies, and various displays and controls. The SCS can hold the spacecraft to a

local vertical attitude during earth or lunar orbit by using orbital rates, or to a specific attitude by using reference gyros mounted in the spacecraft. Information about spacecraft attitude and rate of attitude change is displayed to the astronaut. Steering and thrust signals can be sent to operate the reaction control and service propulsion systems.

1-4.3 SERVICE PROPULSION SYSTEM. The SPS provides thrust to control spacecraft velocity. It is normally controlled and operated by the PGNCS sending SPS signals through the SCS, but can be operated manually through the SCS in emergencies.

The SPS is utilized for mission abort, midcourse velocity corrections, and orbital injections. The SPS is located in the rear of the service module and consists of helium, fuel, and oxidizer tanks; gimbaled thrust chamber; propellant control valves; and propellant distribution system.

1-4.4 REACTION CONTROL SYSTEMS. Two reaction control systems are used for attitude control and stabilization of the spacecraft. One RCS is required for the service module and the other for the command module. Both systems can be operated by the PGNCS or the SCS. Each RCS consists of helium, fuel and oxidizer tanks; propellant control valves; fixed thrust chambers; and distribution subsystem.

Spacecraft attitude, rate of rotation, and small translational velocity changes are normally controlled by an automatic RCS loop, but also may be controlled by a direct RCS loop. Each RCS fuel valve and oxidizer valve contains one solenoid winding controlled by the direct RCS loop.

The automatic RCS loop is activated by signals from the PGNCS, SCS, or by signals from hand controls. The RCS jets fire to rotate the spacecraft and, when the rotational command is removed, jets are automatically fired in the opposite direction to stop the spacecraft rotation.

The direct RCS loop bypasses the automatic circuits to provide direct astronaut control of the RCS jets. After the hand control is released, the spacecraft continues to rotate until the astronaut commands an opposite rotation.

1-4.4.1 Service Module Reaction Control System. The service module RCS provides thrust to control spacecraft attitude during all phases of flight except launch and entry. The service module RCS consists of four hypergolic-propellant-pressurized reaction jet subsystems. The subsystems are mounted in the forward section of the service module and are located 90 degrees apart. Eight of the 16 nozzles are used for roll control, four for pitch, and four for yaw. Each nozzle produces a nominal thrust of 100 pounds in space.

1-4.4.2 Command Module Reaction Control System. The command module RCS is used only during entry after separation of the command and service modules. The RCS of the command module consists of two independent subsystems, which are operated simultaneously and provide three axis control of the command module. If one subsystem fails, the remaining reaction jet subsystem provides the control necessary for safe entry.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

1-4.5 ELECTRICAL POWER SYSTEM. The primary power required to operate the PGNCS is supplied by a hydrox fuel cell in the service module and is used during all mission phases except entry and recovery. Three zinc-silver oxide storage batteries in the command module furnish power for entry and recovery.

The hydrox fuel cell contains three independent modules operating in parallel and capable of producing a 27 to 31 vdc output. In any emergency, any two fuel cell modules could meet all power requirements. Three solid state inverters change the fuel cell output into 115 volt, 400 cps power. The fuel cell dc output and the inverter ac output are both used to provide power for normal PGNCS operation. The 115 volt, 400 cps power is also used to produce 5 volt, 400 cps power for PGNCS operation.

Each fuel cell module contains 31 fuel cell units. The fuel cells utilize hydrogen and oxygen as reactants and potassium hydroxide as an electrolyte. The reactants are stored in tanks furnished as a part of the ECS. The water formed by operating the fuel cells is collected and delivered to the ECS, for potable water.

Two of the storage batteries in the command module can be used to supply supplementary power during peak loads on the fuel cell. The third battery is isolated from the EPS and is used only for post-landing loads. A solid state battery charger is provided to recharge any of the three batteries after normal or emergency discharge.

1-4.6 ENVIRONMENTAL CONTROL SYSTEM. An ECS is used to sustain life in space by providing breathable atmosphere, acceptable temperatures, radiation protection, humidity and pressure control, water, food, and waste disposal. In addition, a water-glycol coolant fluid is circulated about temperature sensitive components of the PGNCS system to aid in maintaining constant operating temperatures. The fluid temperature at the input to the IMU is 45 degrees Fahrenheit and is circulated at a rate of 33 pounds per hour. The ECS includes oxygen tanks and space radiators in the service module, and a distribution system, connections for spacesuit oxygen lines, sensors, regulators, cabin blowers, and heat exchangers in the command module.

1-4.7 COMMUNICATIONS AND INSTRUMENTATION SYSTEM. The Apollo CIS provides voice, television, and telemetry communication with the earth; voice communication among crew members; tracking radar; and communication between the command module and the LEM. A closed loop television system permits viewing external sections of the spacecraft during flight.

Data is transmitted to earth or received from earth by Apollo telemetry. Telemetry data can be stored when direct communication with the earth is not possible because of the location of earth stations or because the spacecraft is behind the moon. Earth stations can determine spacecraft position and transmit this information through telemetry to the PGNCS computer to supplement information obtained from the PGNCS. Critical signals of the PGNCS and other spacecraft systems are conditioned and supplied to pulse code modulated (PCM) equipment for transmission to ground stations.

After earth landing, a recovery beacon in the command module transmits a VHF signal to direct aircraft and ships to the landing area.

1-5 PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM INTERFACE

PGNCS operation during the Apollo mission requires the interface of the PGNCS with other spacecraft systems and with the displays and controls on the main D and C panel.

1-5.1 SYSTEMS. Four spacecraft systems (SCS, SPS, RCS, and EPS) have direct interface with the PGNCS and two systems (ECS and CIS) have indirect interface with the PGNCS. The functional interface between the PGNCS and the other systems is shown in figure 1-4.

The direct interfaces of the PGNCS with the SCS include attitude signals, attitude error signal display, and control signals. The function of each signal depends on the mode of operation of the guidance and control systems. During primary guidance and control, the PGNCS provides attitude and attitude errors to the flight director attitude indicator (FDAI) for display. During backup guidance and control, the SCS body mounted attitude gyro's (BMAG's) can be used to provide attitude data to the CMC by means of the gyro display coupler (GDC). See table 1-II for SCS and PGNCS interface signals.

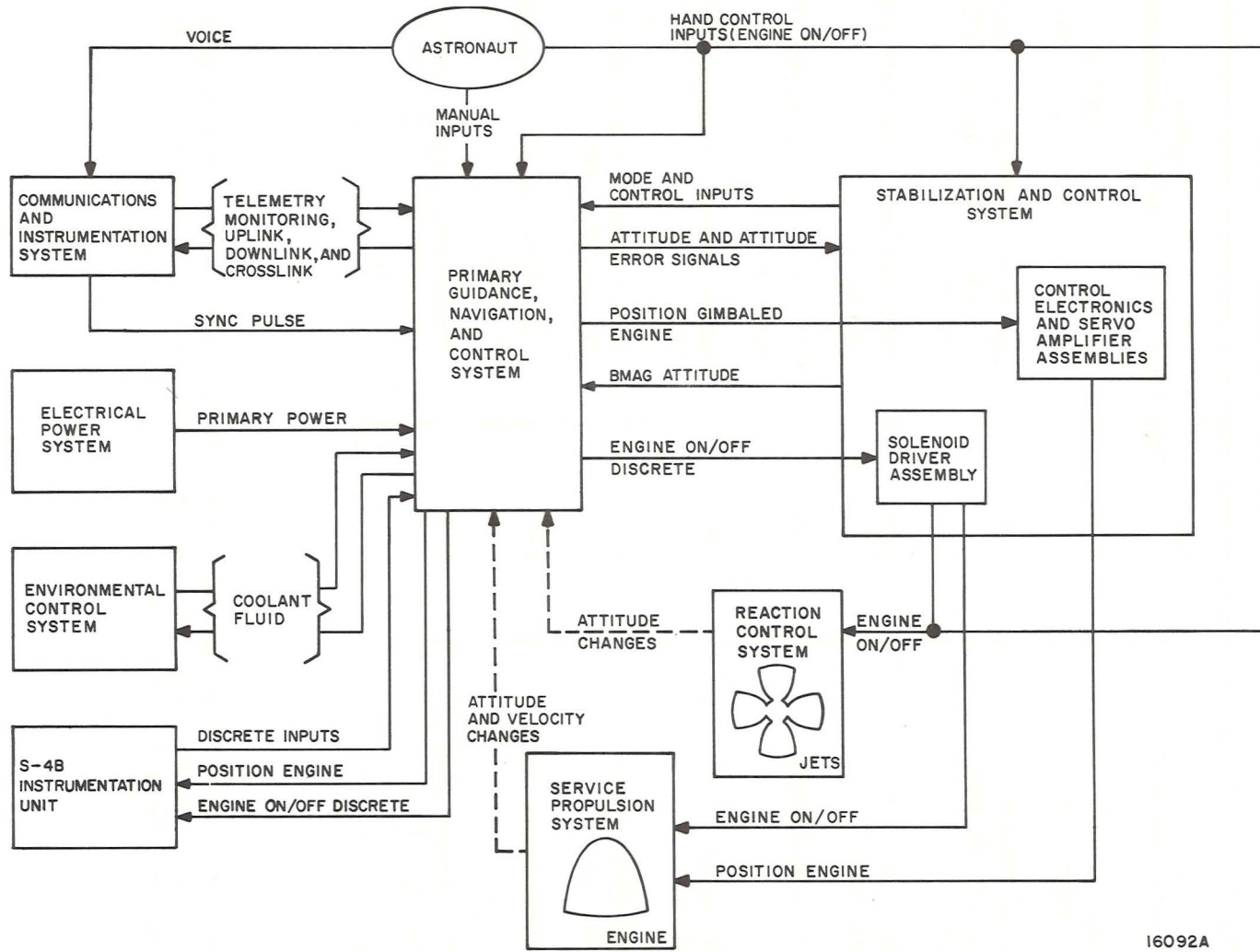
The CMC of the PGNCS generates SPS engine on/off commands and positions the SPS gimbaled engine in pitch and yaw through the generation of steering signals.

The CMC also has direct interface with the RCS of both the command and service modules. The output signals from the CMC, which can be generated by a digital autopilot program, or the output of the CMC resulting from the operation of one of three hand controls is used to turn the RCS jets on and off. The same computer outputs used for command module RCS control are used for service module RCS control. Switchover from the service module to command module operation is external to the PGNCS.

The EPS provides 28 vdc, 5 vac 400 cps, and 115 vac 400 cps power to the PGNCS.

The ECS circulates an ethylene glycol-water coolant about the temperature sensitive electronic equipment in the PGNCS.

The CIS telemetry can transmit spacecraft positional data to the PGNCS computer to supplement the information obtained from the optics and inertial subsystems. Critical signals of the PGNCS are conditioned and supplied to the PCM equipment for transmission to ground stations. The PGNCS can also update the LEM computer through the CIS. See table 1-III for CIS and PGNCS interface signals.



16092A

Figure 1-4. PGNCS Functional Interface, Block Diagram

Table 1-II. SCS Interface Signals

Signal Name	Source	Description
Manual translation commands ($\pm x$, $\pm y$, $\pm z$)	SCS	Signals from translation control which fire jets by CMC control.
Manual rotation commands (roll, pitch, yaw)	SCS	Signals from rotation control which fire RCS jets by CMC control.
BMAG attitude (roll, pitch, yaw)	SCS	BMAG attitude information to the CMC.
RCS jet selection	CMC	RCS jet on/off commands (16 signals).
SPS engine on/off command	CMC	Initiates and terminates SPS thrusting.
Total attitude (roll, pitch, yaw)	ISS	IMU 1X gimbal angle information (IG, MG, OG).
Attitude error (roll, pitch, yaw)	ISS	CDU error signals (roll, pitch, and yaw).
Thrust vector control error (pitch, yaw)	OSS	SPS steering signals from the CMC through the OSS CDU channel.

The PGNCS also has an interface with the S-4B guidance IU. The S-4B guidance provides moding signals to the CMC and the CMC issues thrust maneuver signals to the S-4B stage. See table 1-IV for S-4B and PGNCS interface signals.

Table 1-III. CIS Interface Signals

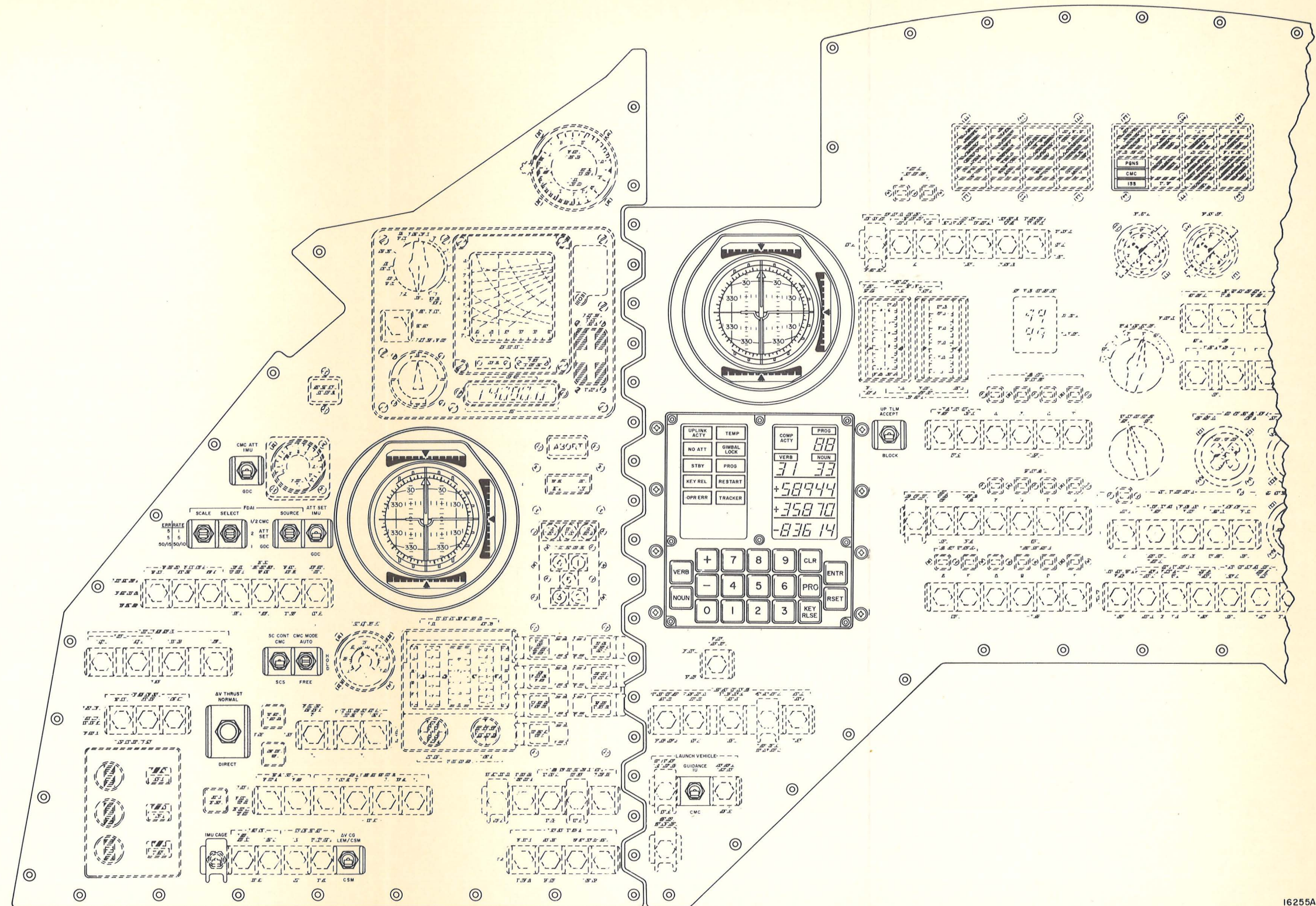
Signal Name	Source	Description
Downlink data	CMC	Computer words to be transmitted to MSFN.
Crosslink data	CMC	Computer words to be transmitted to LEM PGNCS.
Downlink end pulse	CIS	Synchronizes downlink word transfer from CMC to CIS.
Downlink start	CIS	Indicates to CMC that CIS is ready for downlink data.
Downlink bit sync	CIS	Synchronizes bit transfer from CMC to CIS.
Uplink 1	CIS	Transmitted to uplink counter in CMC. Adds one to uplink counter and shifts counter one increment.
Uplink 0	CIS	Transmitted to uplink counter in CMC. Shifts counter one increment.

Table 1-IV. S-4B Interface Signals

Signal Name	Source	Description
S-4B injection sequence start	CMC	Initiates S-4B thrust maneuver during PGNCS control of Saturn.
S-4B cutoff	CMC	Terminates S-4B thrust maneuver during PGNCS control of Saturn.
Attitude error (roll, pitch, yaw)	ISS	PGNCS steering error during PGNCS control of S-4B thrust maneuver.
Liftoff	IU	Indicates to CMC that liftoff has occurred.
Guidance reference release	IU	Indicates to CMC that IU stable platform is inertial.
S-4B separation/abort	IU	Indicates to CMC that S-4B separation/abort has been initiated.
Ullage thrust present	IU	Indicates to CMC that an ullage maneuver will occur.

1-5.2 CONTROLS AND DISPLAYS. Controls and displays on the main D and C panel, figure 1-5, either have a direct interface with the PGNCS, modify the operation of the PGNCS, or modify the display of data generated by the PGNCS. The rotation control and the translation control have an indirect interface with the PGNCS. Descriptions of the controls and displays are provided in table 1-V.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM



16255A

Figure 1-5. Main D and C Panel

Table 1-V. Displays and Controls

Displays or Controls	Function
IMU CAGE	Causes IMU to be coarse aligned to its zero position, independent of CMC operation.
UPTLM	Allows or blocks telemetry data from ground stations to be entered into CMC.
LAUNCH VEHICLE GUIDANCE	Selects either launch vehicle (IU) or CMC control of S-4B guidance. During command module control, CMC executes a STAB control program.
ΔV CG	<p><i>No 7 10/50</i></p> <p>Position of switch indicates whether LEM is attached to command module or not. This permits CMC to take into account change in center of gravity for spacecraft maneuvers.</p>
SC CONT	<p>Three position switch which is used to select controlling source for spacecraft attitude control. Switch allows either CMC or SCS to control spacecraft attitude and thrust vector. In a <u>third position</u>, CMC computed attitude errors are routed through the SCS to generate attitude steering signals. <i>CHANGED</i></p>
CMC MODE	<p>Three position switch which is used to select manner in which spacecraft attitude is controlled by CMC.</p> <p>AUTO position - computer program can automatically control spacecraft and rotation control can be utilized to change spacecraft attitude. <i>OVERRIDE</i></p> <p>HOLD position - computer set in an attitude hold configuration, it cannot perform automatic functions except to maintain attitude. Spacecraft attitude can be controlled by rotation control inputs to digital autopilot program. When rotation control is released, CMC will maintain selected attitude.</p>

Table 1-V. Displays and Controls

Displays or Controls	Function
CMC MODE (cont)	<p><i>TC inputs</i></p> <p>FREE position - computer cannot generate commands to maintain spacecraft attitude or to control spacecraft angular drift rate. Computer can issue RCS on/off commands in response to rotation control or minimum impulse control commands. Upon release of controls, spacecraft is free to drift to any attitude.</p>
ΔV THRUST	<p><i>CMC/THRUST</i></p> <p>Three position switch to remove, enable or directly apply power to SPS propellant valves.</p> <p>NORMAL position - indicates that procedures in engine start checklist have been completed and SPS is ready for firing. This switch position enables normal SPS ignition through CMC or THRUST ON switch. CMC will control RCS ullage firing cycle prior to SPS ignition. When this switch position is selected, a SPS ready discrete is received by CMC.</p> <p>OFF position - removes all ability to energize propellant valves.</p> <p>DIRECT ON position - applies power directly to solenoids bypassing CMC.</p>
CMC ATT	<p><i>CMC P/Rog Does Not Use</i></p> <p>CMC attitude is a two position switch used to select attitude reference for spacecraft.</p> <p>IMU position - spacecraft attitude is referenced to PGNCS stable member.</p> <p>GDC position - GDC provides single axis attitude data to CMC. CMC is not capable of firing SPS engines or RCS jets in GDC position if CMC ATT switch has been selected.</p>
FDAI SOURCE	<p>Selects source of signals displayed on FDAI.</p> <p>CMC position - IMU gimbal angles and PGNCS attitude error signals are selected for display.</p>

(Sheet 2 of 5)

Table 1-V. Displays and Controls

Displays or Controls	Function
<p>FDAI SOURCE (cont)</p>	<p>ATT SET position - difference between attitude set dials and total attitude reference selected by ATT SET (IMU or GDC) is displayed. <i>on ERK NEEDLES</i></p> <p><i>WRONG</i> GDC position - IMU angles are displayed on FDAI ball and BMAG error signals are displayed as attitude errors on error needles.</p>
<p>ATT SET</p>	<p>Used to select source of total attitude input signal to attitude set resolvers.</p> <p>IMU position - IMU gimbal angles 1X resolver signals are used as reference. FDAI error needles display difference between IMU gimbal angles and attitude set angles. Error signals can be zeroed by moving the spacecraft around IMU or changing settings of attitude set dials.</p> <p>GDC position - GDC resolvers provide attitude reference.</p>
<p>FDAI SELECT</p>	<p>Used to select either or both of FDAI's for display of data.</p> <p>1/2 position - permits both FDAI's to receive and display attitude and attitude errors. FDAI 1 would be driven by PGNCS inputs and FDAI 2 would be driven by SCS inputs.</p> <p>1 position - permits signal selected by SOURCE switch to be displayed on FDAI 1, other FDAI is inoperative.</p> <p>2 position - permits signal selected by SOURCE switch to be displayed on FDAI 2, other FDAI is inoperative.</p>
<p>FDAI SCALE</p>	<p>Used to select scales used on FDAI for display of attitude error and attitude rate. Attitude error signals provided by PGNCS are scaled in following manner. Pitch and yaw attitude errors represent plus and minus</p>

Table 1-V. Displays and Controls

Displays or Controls	Function
FDAI SCALE (cont)	<p>15 degrees full scale during guidance monitor and entry and plus and minus 5 degrees during all other modes. Roll attitude error needle represents plus and minus 50 degree full scale deflection in entry and guidance monitor and plus and minus 5 degrees under all other positions.</p> <p>1st and 2nd position - maintain plus and minus full scale deflection for pitch, yaw and roll attitude error.</p> <p>3rd position - selects plus and minus 50 degree full scale deflection for roll and plus and minus 15 degree full scale deflection for pitch and yaw attitude errors.</p>
FDAI	<p>Attitude Indications. FDAI displays total spacecraft attitude with respect to reference source (CMC, ATT SET, GDC) as selected by SOURCE switch. Spacecraft attitude is displayed on gimbale attitude ball. Inputs to FDAI from PGNCS are 800 cps signals representing sine and cosine of gimbal angles. These signals are taken from 1X gimbal resolvers.</p> <p>Attitude Error Displays. Spacecraft attitude errors are displayed by three attitude error needles on FDAI. If SOURCE switch is in CMC position, the attitude error signal is fed to FDAI from CDU. Attitude error signal is proportional to difference between a desired angle and an actual angle.</p> <p>If SOURCE switch is placed in ATT SET position and ATT SET switch is in IMU position, the difference between attitude set dials and IMU gimbal angles will be displayed on attitude error needles. These signals are generated by using 800 cps signals representing sine and cosine of gimbal angles as a reference for attitude set resolvers. When resolver dials read out gimbal angles, attitude error needles indicate zero. If attitude set dials are set at some angles other than</p>

(Sheet 4 of 5)

Table 1-V. Displays and Controls

Displays or Controls	Function
FDAI (cont)	<p>gimbal angles, attitude error needles of FDAI will display difference between attitude set angles and gimbal angles. Error signal can be driven to zero by movement of spacecraft or by resetting attitude set dials. Attitude Rate Displays. PGNCS does not generate attitude rate signals for display on FDAI.</p>
Rotation Controls (2)	<p>Three degree-of-freedom device for manual attitude control of spacecraft. When positioned at more than 1.5 degrees from normal position, supplies switching inputs to CMC input channels. Six lines, one each for plus and minus pitch, roll and yaw make up interface between rotation controls and CMC. Rotation controls are designed so that attitude control signals can be inserted in either direction into any one axis, and any two axes, or all three axes simultaneously. Rotation controls can also be used directly with SCS and RCS.</p>
Translation Control	<p>'T' handle configuration which is capable of being moved up-down, right-left, and in-out simultaneously. Translation control has one line for each direction of motion for each of X, Y, and Z spacecraft axes. Motion of the translation control in two or three directions simultaneously will cause composite spacecraft maneuvers. The translation control, when rotated in a counterclockwise direction will initiate a mission early termination during high altitude and translunar injection phases of the mission. An abort during launch will occur if launch escape tower is attached. A clockwise rotation of the translation control will disable the PGNCS (digital autopilot control of spacecraft).</p>

Chapter 2

SYSTEM AND SUBSYSTEM FUNCTIONAL ANALYSIS

2-1 SCOPE

This chapter provides functional descriptions of the PGNCS, the three subsystems and the signal conditioner. The chapter describes how the three subsystems and the displays and controls work together to perform the PGNCS operations.

2-2 PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

The PGNCS is functionally divided into inertial, optical, and computer subsystems. These subsystems, with control by the astronaut, can perform the following functions for providing inertial guidance, optical navigation, and spacecraft control:

- (1) Periodically establish an inertial reference for measurements and computations.
- (2) Align the inertial reference by precise optical measurements.
- (3) Calculate the position and velocity of the spacecraft by optical or inertial techniques.
- (4) Generate attitude control and thrust commands to maintain the spacecraft on a satisfactory trajectory.
- (5) Display pertinent data about the status of guidance equipment.

For inertial guidance, the PGNCS employs an IMU containing accelerometers mounted on a stabilized, gimbal-mounted platform. The inertial guidance system senses acceleration and attitude changes instantaneously and provides the resulting data to a computer which generates attitude control and thrust control signals to the SPS and RCS. The IMU, portions of the coupling data unit (CDU), pulsed integrating pendulum (PIP) electronics assembly (PEA) and portions of the power and servo assembly (PSA) form the inertial subsystem (ISS) of the PGNCS.

For optical navigation, the PGNCS employs an optical unit assembly (OUA) consisting of a scanning telescope (SCT) and a sextant (SXT). The OUA is used to sight celestial bodies and lunar and earth landmarks. The sightings are used by the computer to determine spacecraft position and velocity and to establish proper alignment of the stable member. The OUA, portions of the CDU, portions of the PSA, and indicator control panel form the optical subsystem (OSS) of the PGNCS.

The CMC, a digital computer, serves as the central data processing element of the PGNCS. It contains a catalog of celestial bodies and is programmed to calculate steering, thrust, and alignment commands using the information obtained from the optical sightings and the IMU. The computer uses data on gravitational fields and celestial perturbations to calculate future position and velocity of the spacecraft. The CMC and two display and keyboard's (DSKY's) form the computer subsystem (CSS) of the PGNCS.

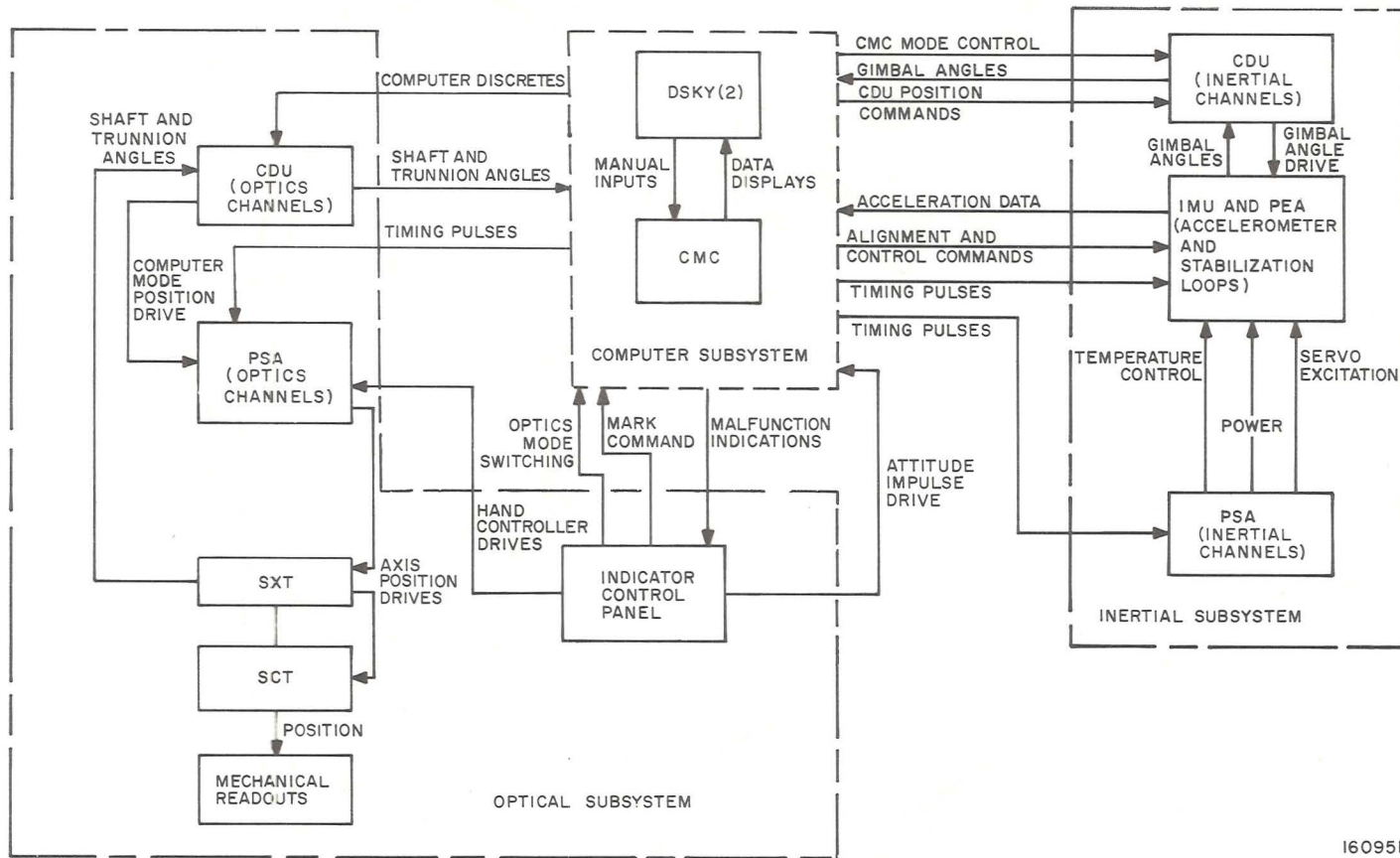
The spacecraft control portion of the PGNCS uses information from the inertial guidance portion to keep the axes that are fixed to the spacecraft structure aligned to the stabilization reference axes. The control portion also keeps the spacecraft velocity vector approximately aligned with the stabilization reference velocity vector. The control portion manages spacecraft rotational and translational orientation.

Figure 2-1 illustrates the signal flow and interface between the PGNCS subsystems.

2-3 BLOCK II PGNCS AXES

Command module and PGNCS axes are illustrated in figure 2-2. Each set of axes illustrated and discussed are orthogonal with positive rotation defined by the advancing right hand rule.

2-3.1 SPACECRAFT AXES. The spacecraft axes have two functions: they provide a reference for all other sets of Apollo axes and they define the point about which attitude and thrust maneuvers are completed. These two functions are accomplished with definition of the spacecraft symmetry axes and the spacecraft body axes. The spacecraft symmetry axes (X, Y, Z) have a fixed relationship to the spacecraft. The X symmetry axis passes through the geometric center of the command module circular base and through the apex of the command module. The Y and Z symmetry axes are perpendicular to one another and lie in the plane of the command module base. The Z axis points along the downrange centerline of the spacecraft. The spacecraft body axes (X_{SC} , Y_{SC} , Z_{SC}) are parallel to the symmetry axes and are located at the spacecraft center of gravity. Rotation about the X_{SC} , Y_{SC} , and Z_{SC} axes (attitude changes or corrections) is defined as roll, pitch, and yaw respectively.



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Figure 2-1. PGNCS, Functional Block Diagram

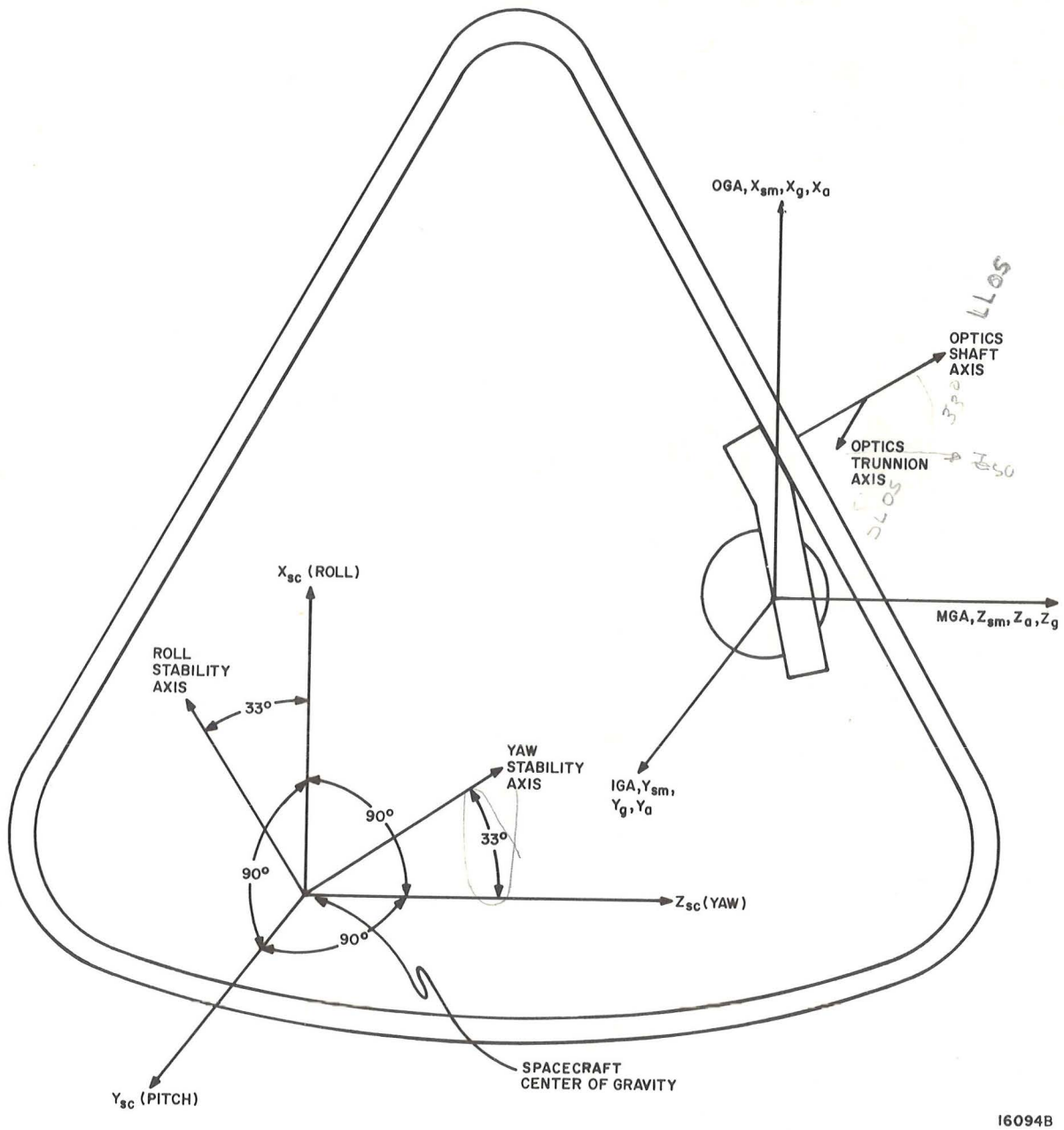


Figure 2-2. Command Module and PGNCS Axes

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

The spacecraft center of gravity is a movable point. Its location depends on the amount of fuel consumed, the astronauts' locations in the command module, and the stages of the Apollo spacecraft attached to the command module. The center of gravity must be determined for accurate attitude and thrust maneuvers.

2-3.2 INERTIAL AXES. Four sets of inertial axes provide references for measuring changes in velocity and attitude. The inertial axes are parallel to the spacecraft axes at zero degrees. The PGNCS senses changes in the alignment of the inertial axes in order to determine the relationship of the inertial axes to the spacecraft axes during flight. Comparison of the alignment of the inertial axes with the spacecraft axes is used in calculating spacecraft attitude changes.

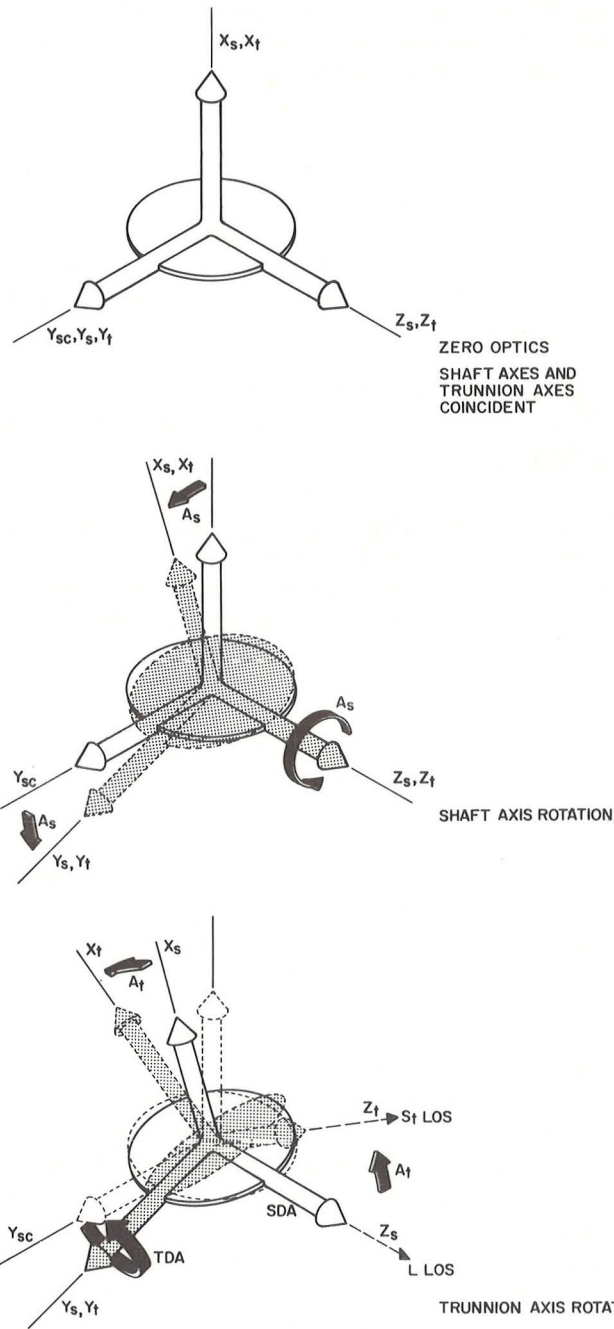
2-3.2.1 Stable Member Axes. The stable member axes (X_{sm} , Y_{sm} , Z_{sm}) provide a reference for aligning the inertial components and for defining the angular orientation of the inertial axes during flight.

2-3.2.2 Gyro Axes. The gyro axes (X_g , Y_g , Z_g) define the positive input axes of the gyros and are parallel to the stable member axes. If the attitude of the stable member is changed with respect to space, each gyro senses the change about its input axes and the circuitry associated with the gyros realigns the stable member to its original position.

2-3.2.3 Accelerometer Axes. The accelerometer axes (X_a , Y_a , Z_a) define the positive input axes of the accelerometers and are parallel to the stable member axes. Velocity changes are measured along the accelerometer input axes. The velocity data is used to determine spacecraft position and velocity.

2-3.2.4 Gimbal Axes. The gimbal axes, outer (OGA), inner (IGA), and middle (MGA) are the axes of the movable gimbals. The gimbal angles are zero when the gimbal axes and the stable member axes are parallel. The attitude of the spacecraft with respect to the stable member is measured by gimbal resolvers.

2-3.3 OPTICAL AXES. The optical axes (figure 2-3) provide a reference for measuring the attitude of the command module and for aligning the IMU. The shaft axis is perpendicular to the conical surface of the command module and is displaced 33 degrees from the Z_{sc} (yaw) axis. The trunnion axis is parallel to the Y_{sc} (pitch) axis when the shaft angle (A_s) is at zero degrees. The shaft drive axis (SDA) is coincident with the Z component of the shaft axis (Z_s) and the trunnion drive axis (TDA) is coincident with the Y component of the trunnion axis (Y_t). The Z_s axis has a fixed orientation and is used to direct the optics line of sight (LOS) toward landmark targets; therefore, the Z_s axis is also referred to as the landmark LOS (LLOS). The Z_t axis is used to direct the optics LOS toward star targets; therefore, the Z_t axis is also referred to as the star LOS (S_t LOS).



16093

Figure 2-3. Orientation of Optical Axes to Nav Base Axes

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

2-4 INERTIAL SUBSYSTEM

The ISS performs three major functions: measures changes in spacecraft attitude, assists in generating steering commands, and measures spacecraft velocity due to thrust. To accomplish these functions, the IMU provides an inertial reference consisting of a stable member gimballed in three degrees of freedom and stabilized by three integrating gyros. Note: During normal operation, the middle gimbal angle is not allowed to exceed ± 70 degrees. The IMU stable member is aligned to a predetermined reference each time the ISS is energized and during prolonged use. Prior to launch, the alignment is accomplished by vertical erection and azimuth alignment controlled by a computer gyro compassing program. During flight, alignment can be accomplished by sighting the optical instruments on celestial objects or it can result from information transmitted through MSFN.

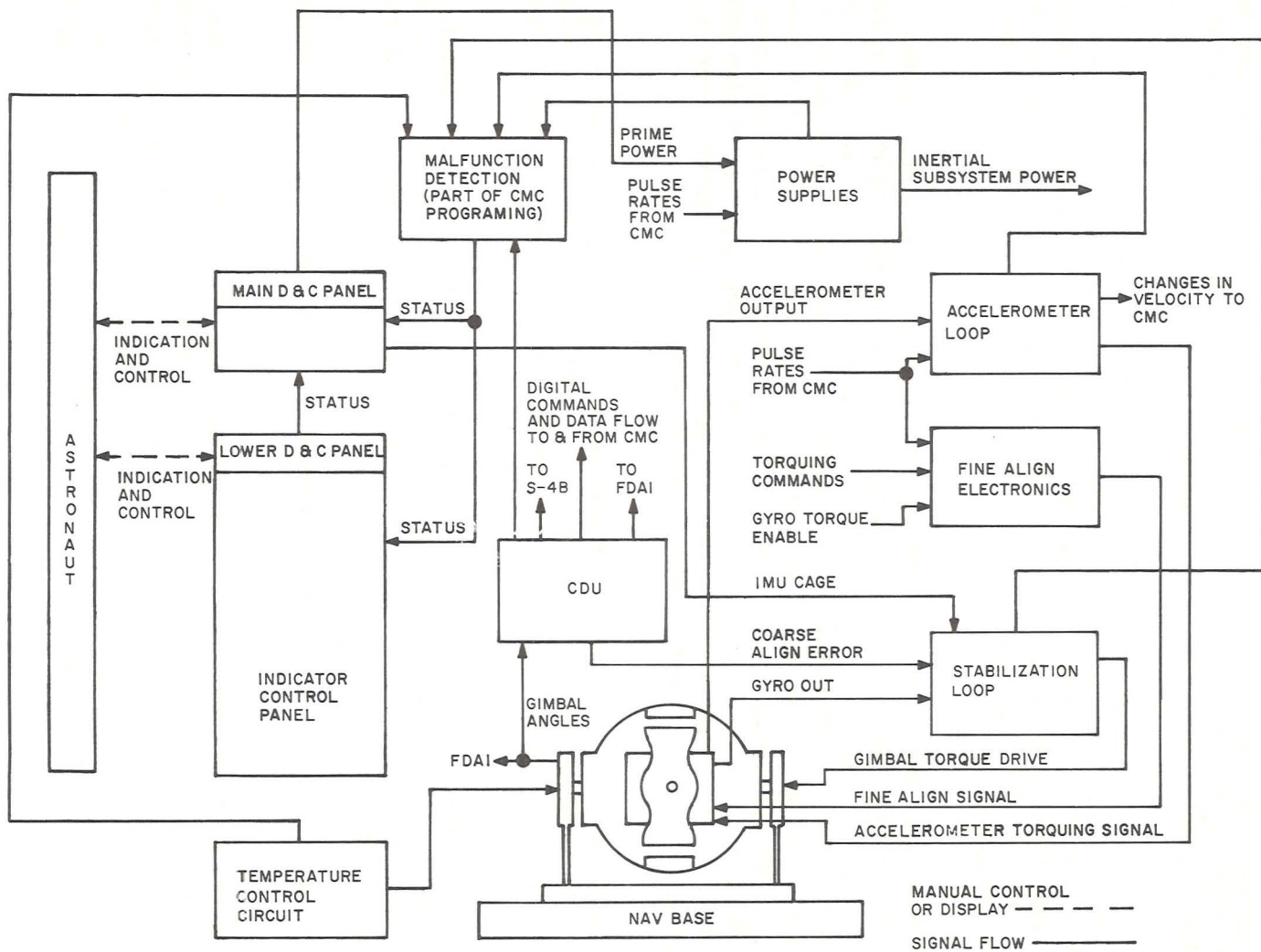
Once the inertial subsystem is energized and aligned, any rotational motion of the spacecraft will be about the stable member, which remains fixed in space. Resolvers, mounted on the gimbal axes, act as angular sensing devices and measure the attitude of the spacecraft with respect to the stable member. These angular measurements are displayed to the astronaut by the FDAI and angular changes are sent to the CMC through the CDU. Desired spacecraft attitude is calculated by the CMC and is compared with the actual IMU gimbal angles. Any difference between the actual gimbal angles and the desired gimbal angles, as calculated by the computer, results in attitude error signals which change spacecraft attitude through the SPS or RCS.

Three pendulous accelerometers mounted on the stable member sense changes in velocity along the axes of the IMU stable member and supply incremental velocity data to the computer. The CMC uses the incremental velocity data to calculate spacecraft total velocity.

The modes of operation of the inertial subsystem can be initiated automatically by the CMC or by the astronaut selecting computer programs through the computer keyboard. The status or mode of operation is displayed on the display and keyboard as determined by data from the computer.

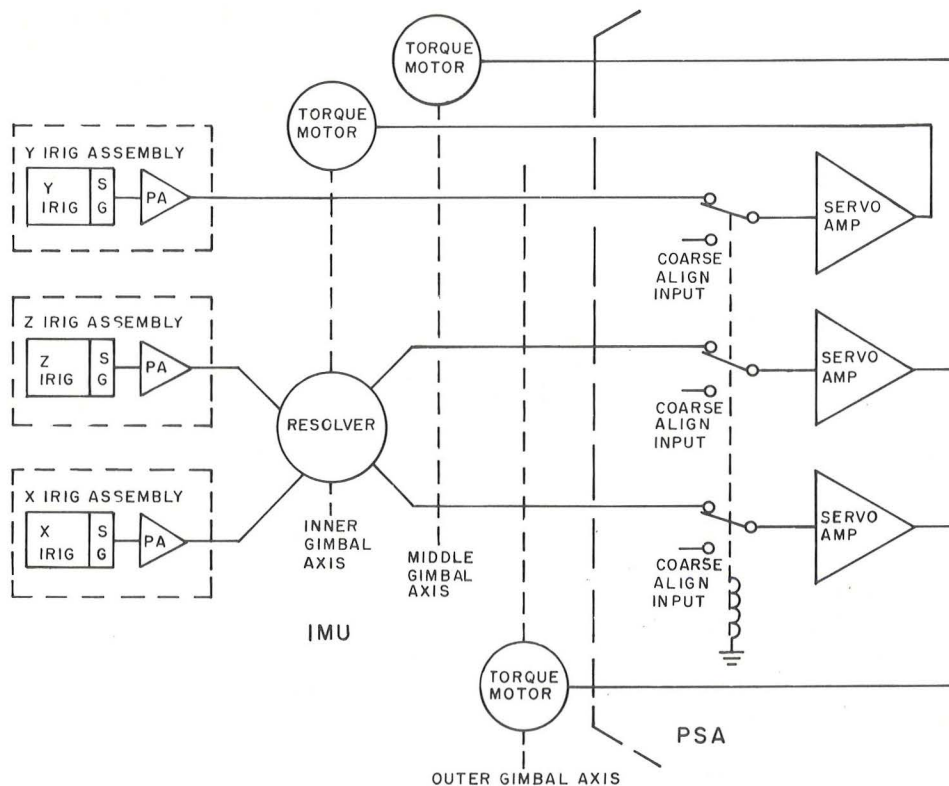
For explanation purposes, the ISS is divided into functional blocks as shown in figure 2-4.

2-4.1 STABILIZATION LOOP. The stabilization loop (figure 2-5) maintains the stable member in a specific spatial orientation so that three mutually perpendicular (16) PIP accelerometers can measure the proper components of command module acceleration with respect to the coordinate system established by the stable member orientation. An input to the stabilization loop is created by any change in spacecraft attitude with respect to the spatial orientation of the stable member. With near zero



16128 A

Figure 2-4. Inertial Subsystem, Block Diagram



15191

Figure 2-5. Stabilization Loop, Block Diagram

gimbal angles, the inertia of the stable member tends to maintain the stable member in a fixed spatial orientation. Because of gimbal friction and unbalances, motion of the spacecraft relative to the stable member produces a torque on the stable member which tends to change its orientation. This change is sensed by the stabilization gyros. When the gyros sense an input, they issue error signals which are amplified, resolved (if necessary) into appropriate components, and applied to the gimbal torque motors. The gimbal torque motors then drive the gimbals to maintain the stable member in fixed spatial orientation.

The stabilization loop consists of three pre-aligned Apollo II inertial reference integrating gyro (IRIG) assemblies, a gyro error resolver, three gimbal servo amplifiers, three gimbal torque motors, three gimbals, and circuitry associated with these components. The inner gimbal is the stable member upon which the three stabilization gyros are mounted. The gyros are mounted with their input axes oriented in an orthogonal configuration. Movement of any gimbal tends to result in a movement of the stable member and rotation about the input axes of one or more of the stabilization gyros.

The stabilization loop contains three parallel channels. Each channel starts with a stabilization gyro (X, Y, or Z) and terminates in a gimbal torque motor. The torque motor drives gimbals resulting in a movement of the stable member and a movement of the stabilization gyros. When a movement of the IMU support gimbal attempts to displace the stable member from its erected position, one or more of the stabilization gyros senses the movement and issues error signals. The phase and magnitude of the 3,200 cps gyro error signal represents the direction and amount of rotation experienced by the gyro about its input axis. The error signal is fed from the gyro signal generator ducosyn to the associated IRIG preamplifier, which is a part of the prealigned Apollo II IRIG assembly. Amplification of the error signal is required to achieve a high signal-to-noise ratio through the gimbal slip rings.

The amplified gyro error signals also represent motion of the stable member about its axis since the stable member axes (X_{SM} , Y_{SM} , Z_{SM}) and the gyro axes (X_g , Y_g , Z_g) are parallel to one another. If the middle and outer gimbal axes remain parallel with the stable member axes, then movement of the outer gimbal (a roll movement of the spacecraft) is sensed by only the X gyro and movement of the middle gimbal (yaw movement of the spacecraft) is sensed by only the Z gyro. Movement of the stable member about the inner gimbal axis (Y_{SM}), however, changes the relationship of the X and Z gyro input axes to the outer and middle gimbal axes. As a result, a movement of the middle or the outer gimbal is sensed by both X and Z gyros. The input required by the gimbal servo amplifiers to drive the gimbals and move the stable member back to its original position must be composed of components of both the X and Z gyros. The required gimbal error signals are developed by the gyro error resolver. The gyro error signals, $E(X_g)$ and $E(Z_g)$, are applied to the stator windings of the gyro error resolver. The rotor windings are connected to the inputs of the outer and middle gimbal servo amplifiers. Movement of the stable member about the inner gimbal axis (pitch movement of the spacecraft) changes the position of the resolver rotor relative to the resolver stator. This change corresponds electromagnetically to the change in the relationship of the stable member axes to the outer and middle gimbal axes. The outputs taken from the rotor are the required middle and outer gimbal error signals (E_{mg} and E_{og}). Since the inner gimbal torque motor axis and the Y axis of the stable member are the same axis, the Y gyro error signal, $E(Y_g)$, is equal to the inner gimbal error signal, (E_{ig}), and is fed directly to the inner gimbal servo amplifier.

The three identical gimbal servo amplifier modules are located in the PSA and each contains a phase sensitive demodulator, a filter, and a dc operational power amplifier. The phase sensitive demodulator converts either the 3,200 cps gimbal error or 800 cps coarse align error, zero or pi phase, signals into a representative positive or negative dc signal. The dc signal is filtered and applied to a dc operational amplifier with current feedback. The compensation network in the feedback circuit of the amplifier controls the response characteristics of the entire stabilization loop. The output of the dc amplifier has an operating range between +28 volts dc and -28 volts dc and drives the respective gimbal torque motor directly in either angular direction.

The gain required for each stabilization loop differs. This difference compensates for the differences in gimbal inertia. The proper gain is selected by the connections to the gimbal servo amplifier module. A single torque motor is mounted on each gimbal at the positive end of the gimbal axis. The torque motors drive the gimbals to complete the stabilization loop.

The orientation of the stable member can be changed in either the coarse align, fine align, or IMU cage modes. Signals to reposition the gimbals are injected into the gimbal servo amplifiers from the CDU during the coarse align and IMU cage modes and into the stabilization gyros from the fine align electronics during the fine align mode. During the IMU cage mode and the coarse align mode, the reference signal for the demodulator in the gimbal servo amplifier is externally switched from 3,200 cps to 800 cps.

2-4.2 FINE ALIGN ELECTRONICS. The fine align electronics (figure 2-6) provides torquing current to the stabilization gyros to change the orientation of the IMU gimbals during the fine align mode. The operation of the fine align electronics is controlled by the CMC.

The components of the fine align electronics are common to the three stabilization gyros. The fine align electronics provides torquing signals to the stabilization gyros one at a time on a time shared basis. The fine align electronics consists of a gyro calibration module, a binary current switch module, and a dc differential amplifier and precision voltage reference module; all are located in the PSA.

The fine align electronics is enabled and controlled by CMC inputs to the gyro calibration module. The CMC inputs are torque enable pulses, gyro select pulses, a torque set command, and a torque reset command. The fine align electronics is enabled by the torque enable pulses. The torque enable pulse is a train of pulses three microseconds in width and occurring at 102.4 kpps. The torque enable pulses are applied through a relay driver to energize the torque enable relay in the calibration module. When the torque enable relay is energized, system 28 vdc is applied to the precision voltage reference (PVR) and regulated 120 volts dc from the pulse torque power supply is applied to the dc differential amplifier and the binary current switch. The torque enable pulse train is received 20 milliseconds prior to any gyro set command.

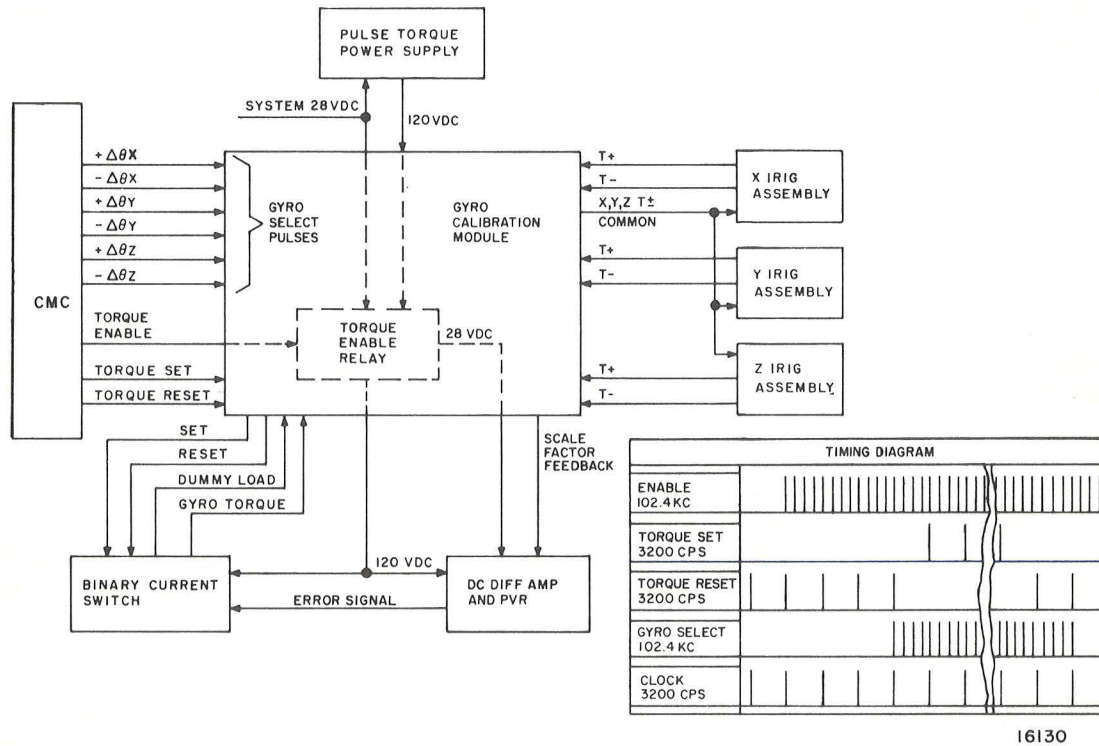
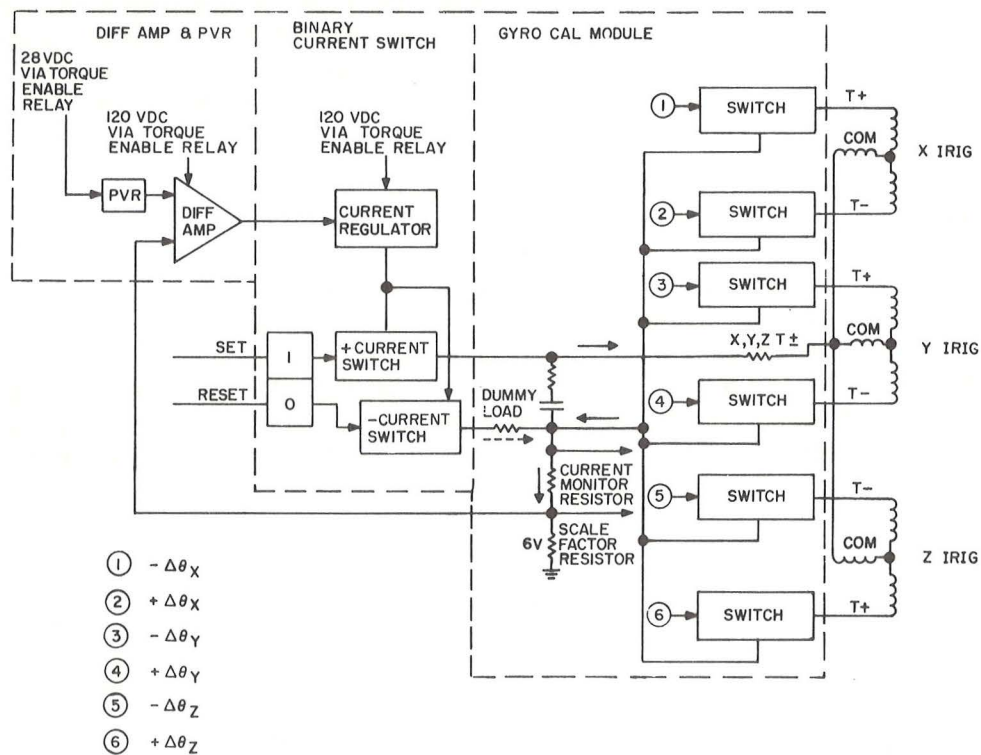


Figure 2-6. Fine Align Electronics - Computer Inputs

The gyro to be torqued and the direction it is to be torqued is selected by the CMC sending gyro select pulses to one of the six $+\Delta\theta$ or $-\Delta\theta$ inputs. (See figure 2-7.) The gyro select pulse is a train of pulses three microseconds in width and occurring at 102.4 kpps. The pulse train activates a transistor switch network which controls current through the T+ or T- coils of the torque generator ducosyn in the gyro selected. The gyro select pulse train is received 312.5 microseconds (one CMC clock time at 3,200 pps) prior to any torque set command.

The torque set and reset commands are 3,200 pps pulse trains containing pulses that are three microseconds in width. A 3,200 pps pulse train will be present on the torque set line when any gyro is to be torqued. A 3,200 pps pulse train is present on the torque reset line at all other times. This ensures that the binary current switch is in the reset condition prior to receipt of a torque enable command from the CMC. When the gyro has been torqued the proper amount, a torque reset command is issued which



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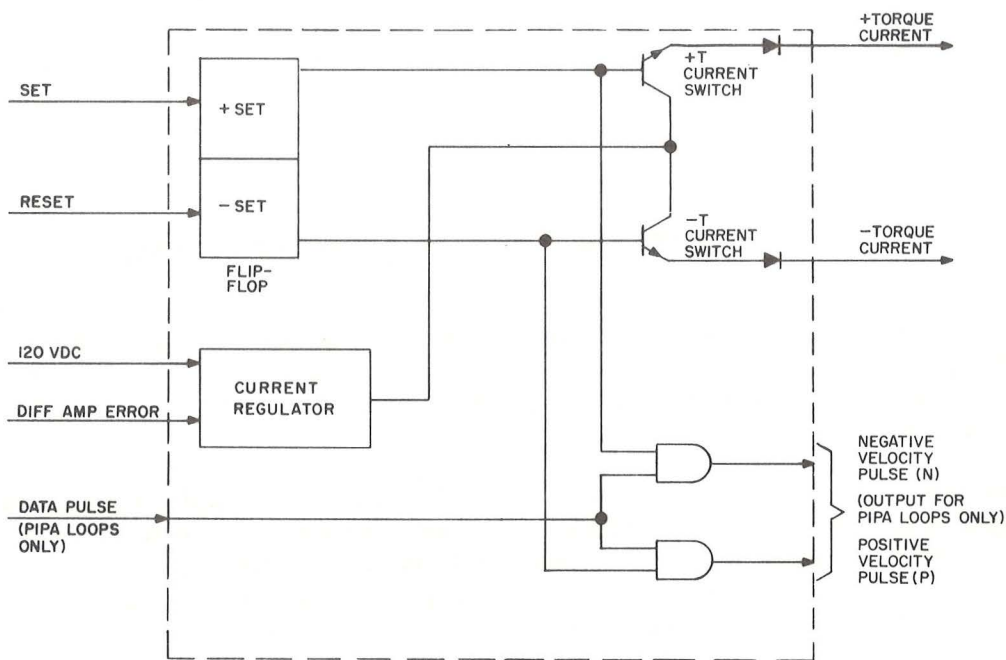
Figure 2-7. Fine Align Electronics - Gyro Selection

cuts off the torque current. The gyro select pulse train is removed 312.5 microseconds after the torque reset command is issued. The torque set and torque reset pulses feed through a 1:2 step-up transformer in the calibration module to the set and reset inputs of the binary current switch.

The torque current from the binary current switch is applied through a tuned resistive-capacitive compensation network in the calibration module to make the torque generator ducosyn windings appear as a pure resistive load to the binary current switch. The torque current to the gyros is through the T±(common) line. Current will flow only through the selected torque generator coils, the current monitor resistor, and the scale factor resistor. The voltage drop developed across the scale factor resistor is used as a feedback to the differential amplifier to regulate the torquing current. The voltage drop across the current monitor resistor is applied to PSA test points for external monitoring of gyro torque current.

When no gyro is being torqued, the binary current switch provides current flow through a dummy load resistor and through the current monitor and scale factor resistors. In this manner, the binary current switch maintains a continuous flow of torque current. The dummy load resistor simulates the impedance of the torque generator coil and a compensation network.

The torque set and torque reset pulses trigger a flip-flop (bi-stable multivibrator) in the binary current switch. See figure 2-8 for a block diagram of the binary current switch. If the flip-flop is in the +set condition, the +set condition will remain until a reset command resets the flip-flop. The outputs of the flip-flop control two transistor switches. If the flip-flop is in the +set condition, the +set output is present at the base of the +torque current switch, and causes the switch to turn on. The +torque current switch closes the path from the 120 volt supply through the current regulator to the proper T+ or T- winding of the selected gyro through the calibration module. If the flip-flop is in the -set condition, the -torque current switch will turn on to close the current path through the dummy load resistor.

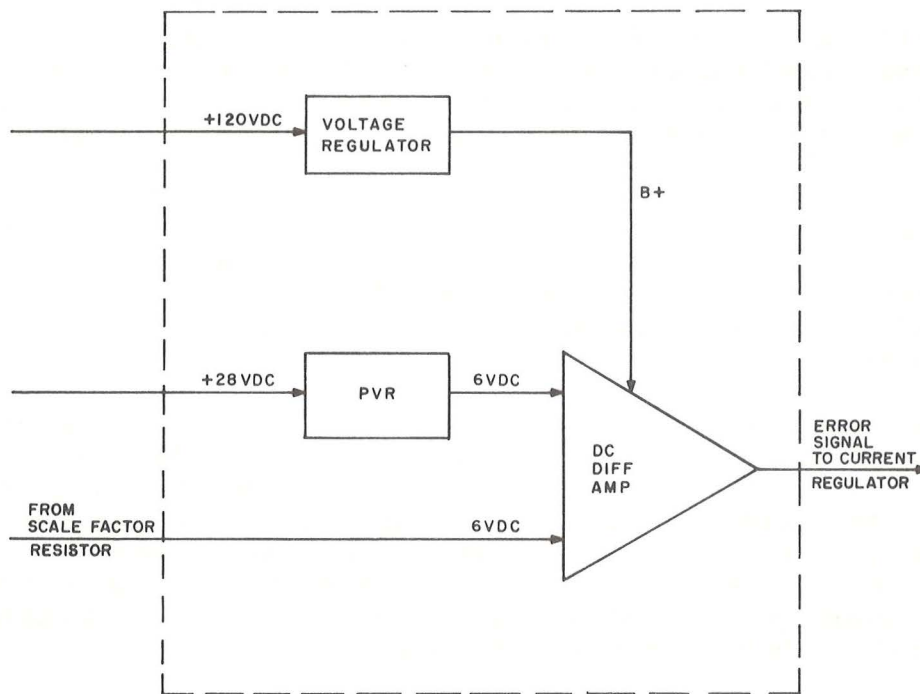


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Figure 2-8. Binary Current Switch

The binary current switch used in the fine align electronics is identical to the binary current switch used in the accelerometer loops. The portion of the binary current switch used only for the accelerometer loops is disabled in the fine align electronics application. In the accelerometer loop application, current to the accelerometer T+ torque generator coil is provided by the +torque current switch and current to the T- torque generator coil is provided by the -torque current switch. Therefore, the +torque and -torque designations of the switches have significance. In the fine align electronics application, the switch designations have no significance since current to both the T+ and T- coils of the gyro torque generators is provided by the +torque current switch while the -torque current switch provides only the dummy load current.

The dc differential amplifier and PVR module (figure 2-9) maintains the current through the windings of the torque generator ducosyn at 84 milliamperes. The PVR is supplied with 28 volts dc and, through the use of zener diode circuits, develops an accurate 6 volts dc for use as a reference voltage. The scale factor resistor in the



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Figure 2-9. DC Differential Amplifier and Precision Voltage Reference

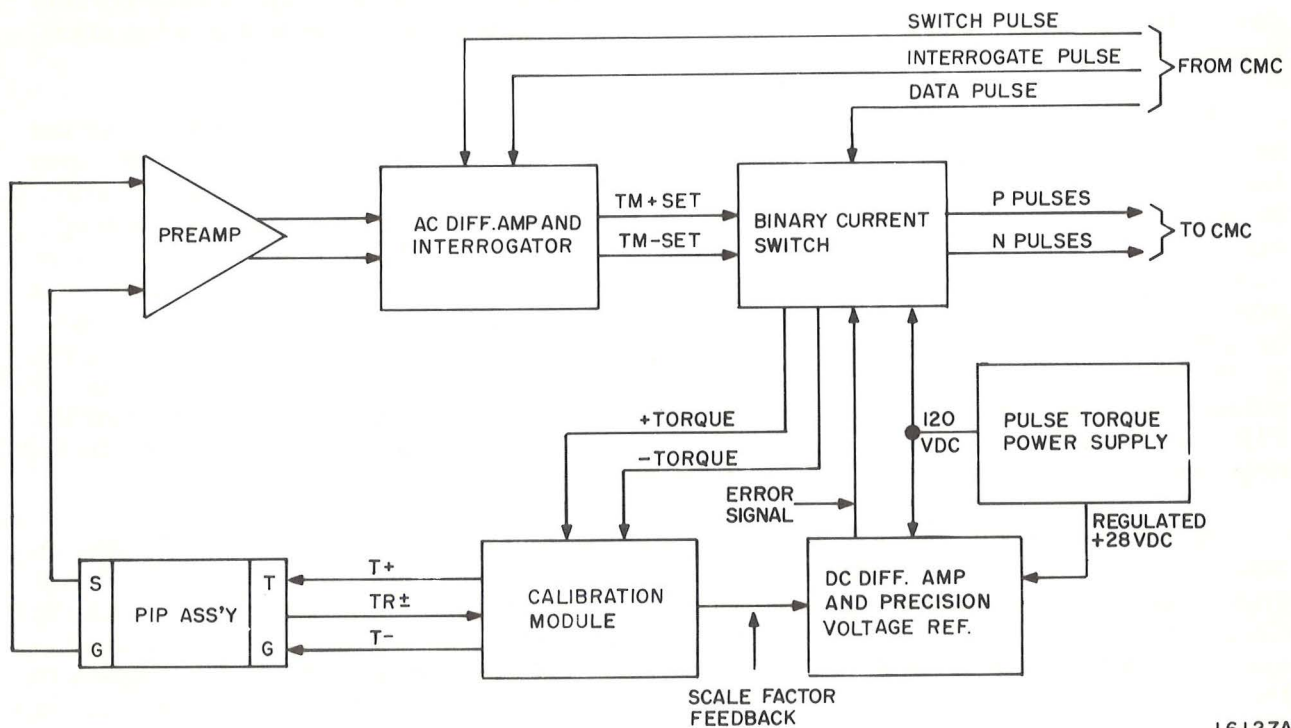
calibration module also has 6 volts developed across it when 84 milliamperes of current flows through it. A comparison is made by the dc differential amplifier of the PVR 6 volts and the scale factor resistor 6 volts. Any deviation from the nominal 84 milliamperes of torquing current will increase or decrease the voltage developed across the scale factor resistor and cause an output error signal from the dc differential amplifier. This error signal controls the current regulator in the binary current switch. The current regulator, which is in series with the torque generator coils of the selected gyro and the 120 volt dc source, will maintain the torquing current at 84 milliamperes.

The current flow through the windings of the torque generator ducosyn causes the gyro float to rotate about the gyro's output axis. A $+\Delta\theta$ gyro select command from the CMC will allow torque current to flow through a T- torque generator coil which results in a positive rotation of the gyro float about the output axis. A $-\Delta\theta$ gyro select command produces a negative float rotation. Float rotation results in an error output from the signal generator ducosyn. The error signal is applied to the stabilization loop to reposition the gimbals and the stable member. The change in gimbal angles is transmitted by the CDU read counters to the CMC.

2-4.3 ACCELEROMETER LOOP. The three accelerometer loops measure the acceleration of the stable member along three mutually perpendicular axes. This data is integrated by the CMC to determine velocity which is used by the CMC to determine the spacecraft velocity vector. Figure 2-10 is a functional diagram of an accelerometer loop.

The three accelerometer loops contain three prealigned 16 PIP assemblies, three PIP preamplifiers, three ac differential amplifier and interrogator modules, three binary current switches, three calibration modules, three dc differential amplifier and precision voltage reference modules, a pulse torque isolation transformer, and associated electronics.

The three mutually perpendicular PIP's are acceleration sensitive devices which, in closed loops with their associated accelerometer loop electronics, become integrating accelerometers. The PIP is basically a pendulum-type device consisting of a cylinder with a pendulous mass unbalance (pendulous float) pivoted with respect to a case. The pivot axis defines the PIP output axis. A signal generator at the positive end of the output axis provides electrical outputs indicative of the rotational float position. A torque generator, at the other end of the float acts as a transducer to convert electrical signals into torque about the float shaft.



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Figure 2-10. Accelerometer Loop

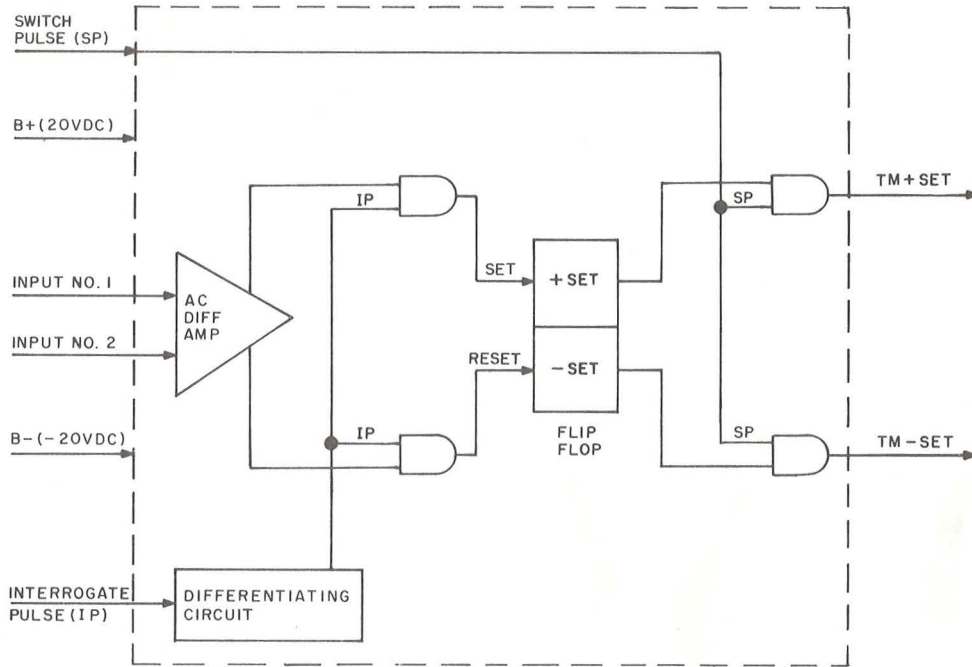
A PIP accelerometer loop can be mechanized to operate in a ternary (three state) or binary (two state) mode. The Block II loops are designed to operate in the binary mode with the PIP pendulous float in continuous oscillatory rotation. The oscillation is produced by alternate positive and negative torquing from the torque generator. The torque generator has two windings, one to produce torque in a positive direction, the other to produce torque in a negative direction. Only one winding is energized at any one time. The windings are energized alternately by the setting of a flip-flop in the binary current switch (figure 2-7). When the accelerometer loop is first energized, the interrogator sets the flip-flop to route torquing current to the winding which will rotate the float to null. As the float passes through null, the phase of the output signal generator changes so that the interrogator issues pulses to reset the flip-flop in the binary current switch and thus route torquing current to the other torque winding. The float is then torqued in the opposite direction until the signal generator output again changes phase as the float passes through null which reinstates the cycle.

The output of the signal generator, after being amplified by the PIP preamplifier, is interrogated 3,200 times a second by the interrogate pulse. The binary current switch flip-flop can be reset only when the interrogate pulse is present and the signal generator output is of the proper phase.

The PIP pendulous float oscillates about its null point at a frequency near the natural resonant frequency of the loop. The resonant frequency is dependent upon float damping, signal and torque generator sensitivity, and other loop characteristics. In Block II accelerometer loops, this resonant frequency is approximately 500 cps. Since the accelerometer loop is designed with a torque winding selection rate of 3,200 pulses per second, the loop will oscillate at a frequency equal to $3,200 \div x$ where x is any even number. Solving the equation $f = 3,200 \div x$, the frequency closest to 500 is $533\frac{1}{3}$. In this case, the value of x is six. Thus one complete pendulum cycle will occur during six torque selection pulses. Dividing the time for the six pulses into the time required for positive and negative rotations, it is seen that the PIP functions in a 3-3 mode (positive rotation for three torque selection pulses, negative rotation for three torque selection pulses).

In its 3-3 mode and sensing no acceleration, the PIP pendulous float rotates an equal angular distance on both sides of an electrical and mechanical null. The 2 volt rms, 3,200 cps, one phase signal generator excitation voltage is synchronized with the CMC clock. The signal generator has a center tapped secondary winding which provides a double ended output, one side having a zero phase reference with respect to the 3,200 cps excitation and the other side a pi phase reference. The center tap is connected to ground. The output signal is representative of the magnitude and direction of the rotation of the pendulous float about the output axis. The error signal is routed to the preamplifier mounted on the stable member. The phase of the output signal from the preamplifier is shifted -45° from the reference excitation. The output signal is then in phase with the interrogate pulse. The phase shifted zero or pi phase signals from the preamplifier are applied as separate inputs to the ac differential amplifier. The ac differential amplifier outputs are routed to the interrogator.

The ac differential amplifier and the interrogator are packaged in the same module located in the PEA. The interrogator analyzes the ac differential amplifier outputs to determine the direction of the 16 PIP float movement and to generate appropriate torquing commands. See figure 2-11 for a block diagram of the ac differential amplifier and interrogator module. The two amplified signals from the ac differential amplifier go to two summing networks and threshold amplifiers (represented in figure 2-11 by AND gates). Interrogate pulses (IP) are continuously being received by the interrogator from the CMC. An interrogate pulse is a two microsecond pulse occurring at 3,200 pps and timed to occur 135 degrees after the positive going zero crossing of the reference excitation. (See figure 2-12.) With this phasing, the interrogate pulse occurs at the 90 degree peaks of the phase shifted zero or pi phase input signals from the PIP preamplifiers. The interrogate pulse occurs at a positive 90 degree peak of the zero phase signal if the float angle is positive and at a positive 90 degree peak of the pi phase signal if the float angle is negative. The zero and pi phase signals and the interrogate pulses

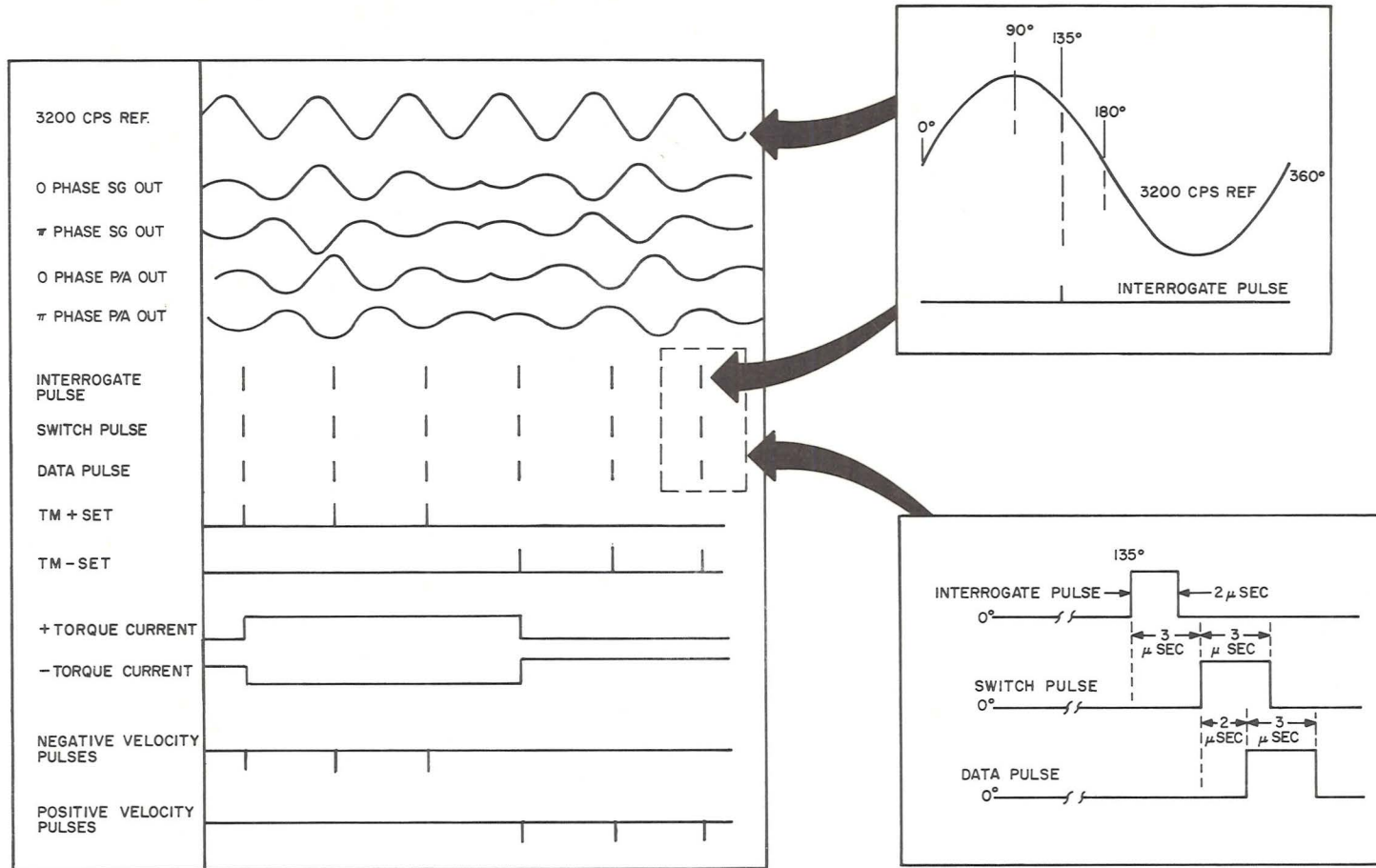


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Figure 2-11. AC Differential Amplifier and Interrogator Module

are ANDed by the summing network and threshold amplifier. The gated outputs of the threshold amplifier are applied to a flip-flop as set or reset pulses. If the flip-flop is in the +set condition, a succession of set pulses will maintain the +set condition. The +set condition will remain until the float angle passes through null. At this time a reset pulse will be produced to cause the flip-flop to go to the -set condition.

The outputs of the flip-flop are applied to two AND gates which are also driven by switch pulses received from the CMC. The switch pulses are a train of clock driven 3,200 pps pulses three microseconds in width that are timed to occur three microseconds after the leading edge of the interrogate pulse. The flip-flop enables only one output gate at any switch pulse time. The outputs of the AND gates are called the TM+ set pulse and the TM- set pulse.



15175C

Figure 2-12. Accelerometer Timing

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

The binary current switch utilizes the TM+ set and TM- set outputs of the interrogator to generate 16 PIP torquing current and velocity change pulses. See figure 2-8 for a block diagram of the binary current switch. The TM+ set and the TM- set pulses furnish the input to a flip-flop. If the flip-flop is in the +set condition, a succession of TM+ set pulses will maintain the +set condition. The +set condition will persist until the float passes through null. The phase change will cause the flip-flop of the ac differential amplifier and interrogator module to reset to the -set condition. At this time, a TM- set pulse will be developed and will cause the binary current switch flip-flop to go to the -set condition. The outputs of the flip-flop control two transistor current switches. If the flip-flop is in the +set condition, the +set output will be at the base of the +torque current switch and will turn it on. The +torque current switch closes the path from the current regulated 120 volt dc supply through the PIPA calibration module to the 16 PIP T+ torque generator coils. If the flip-flop is in the -set condition, the -torque current switch will be turned on, closing the path through the T- torque generator coils.

An acceleration along the PIP input axis causes the pendulous mass to produce a torque which tends to rotate the float about the output axis in proportion to the magnitude of the acceleration. The acceleration produced torque aids and opposes the torque generator forces causing changes in the time required for the float to be torqued back through null. A change in velocity (ΔV) is the product of acceleration and incremental time (Δt); the torque is proportional to an incremental change in velocity (ΔV).

$$T_{ACCEL} = K_1 a \Delta t = K_1 \Delta V$$

With the float in motion due to loop torquing, additional torque is required to overcome the acceleration torque and to keep the pendulous float in its oscillatory motion. Torque current at any one time is a constant. Additional torque is obtained by supplying torque current through the torque windings longer. To determine the amount of acceleration sensed by the PIP, it is necessary only to measure the time that torque current is applied to each torque winding. The amount of acceleration is derived from the following identity:

$$ACCEL_{IND} = K_2 \Sigma [(T+) - (T-)] \Delta t$$

From the above identity, it is seen that torquing time (Δt) is proportional to the change in velocity (ΔV).

$$K_1 \Delta V = K_2 \Sigma [(T+) - (T-)] \Delta t$$

$$\Delta V = \frac{K_2}{K_1} \Sigma [(T+) - (T-)] \Delta t$$

The time (Δt), representative of the ΔV , is sent to the CMC in the form of P and N pulses (figure 2-9).

In addition to selecting the proper torque generator winding, the outputs of the binary current switch flip-flop also go to two AND gates where they are ANDed with the 3,200 cps data pulses from the CMC. The data pulse is three microseconds in width and is timed to occur two microseconds after the leading edge of the switch pulse. (See figure 2-11). Since the data pulse and switch pulse are both 3,200 cps, the CMC receives either a P pulse or an N pulse once every $1/3,200$ second. When the PIP is sensing no acceleration, the pendulum is oscillating at a frequency of $533-1/3$ cps and the CMC is receiving three P pulses and three N pulses once every cycle or once every $\frac{1}{533-1/3}$ seconds. The CMC contains a forward-backward counter which receives the $533-1/3$ velocity pulses and detects any actual gain in velocity.

The counter counts forward on the three P pulses and then backward on the three N pulses. The counter continues this operation and generates no ΔV pulses. With an acceleration input to the PIP, however, the loop no longer operates at the 3-3 ratio and the counter exceeds its capacity and reads out the plus or minus ΔV pulses which are then stored and used by the CMC. The pulses above the 3-3 ratio are representative of the additional torque supplied by the torque generator to compensate for the acceleration felt by the spacecraft. Each pulse indicates a known value of ΔV due to the loop scale factor.

The PIPA calibration module (figure 2-13) compensates for the inductive load of the 16 PIP torque generator ducosyns and regulates the balance of the plus and minus torques. The calibration module consists of two load compensation networks for the torque generator coils of the 16 PIP. The load compensation networks tune the torque generator coils to make them appear as a pure resistive load to the binary current switch. A variable balance potentiometer regulates the amount of torque developed by the torque generator coils. Adjustment of this potentiometer precisely regulates and balances the amount of torque developed by the T+ and T- torque generator coils. This balancing insures that for a given torquing current an equal amount of torque will be developed in either direction.

The calibration module also includes a current monitor resistor and an adjustable scale factor resistor network in series with the torque generator coils. A nominal six volts is developed across the scale factor resistor network due to the torquing current and is applied as an input to the dc differential amplifier and precision voltage reference module. The voltage drop across the current monitor resistor is used for external monitoring purposes.

The dc differential amplifier and PVR are identical to those used in the fine align electronics. (See figure 2-9.) The dc differential amplifier and PVR module maintain the current through the ducosyn torque generator coils at 100 milliamperes. The PVR is supplied with regulated 28 volts dc and, through the use of precision circuits, develops an accurate 6 volts for use as a reference voltage. The scale factor resistor in the calibration module also develops 6 volts when 100 milliamperes of current flows through it. A comparison is made by the dc differential amplifier of the PVR 6 volts and the scale factor 6 volts. Any deviation of the binary current switch torquing current

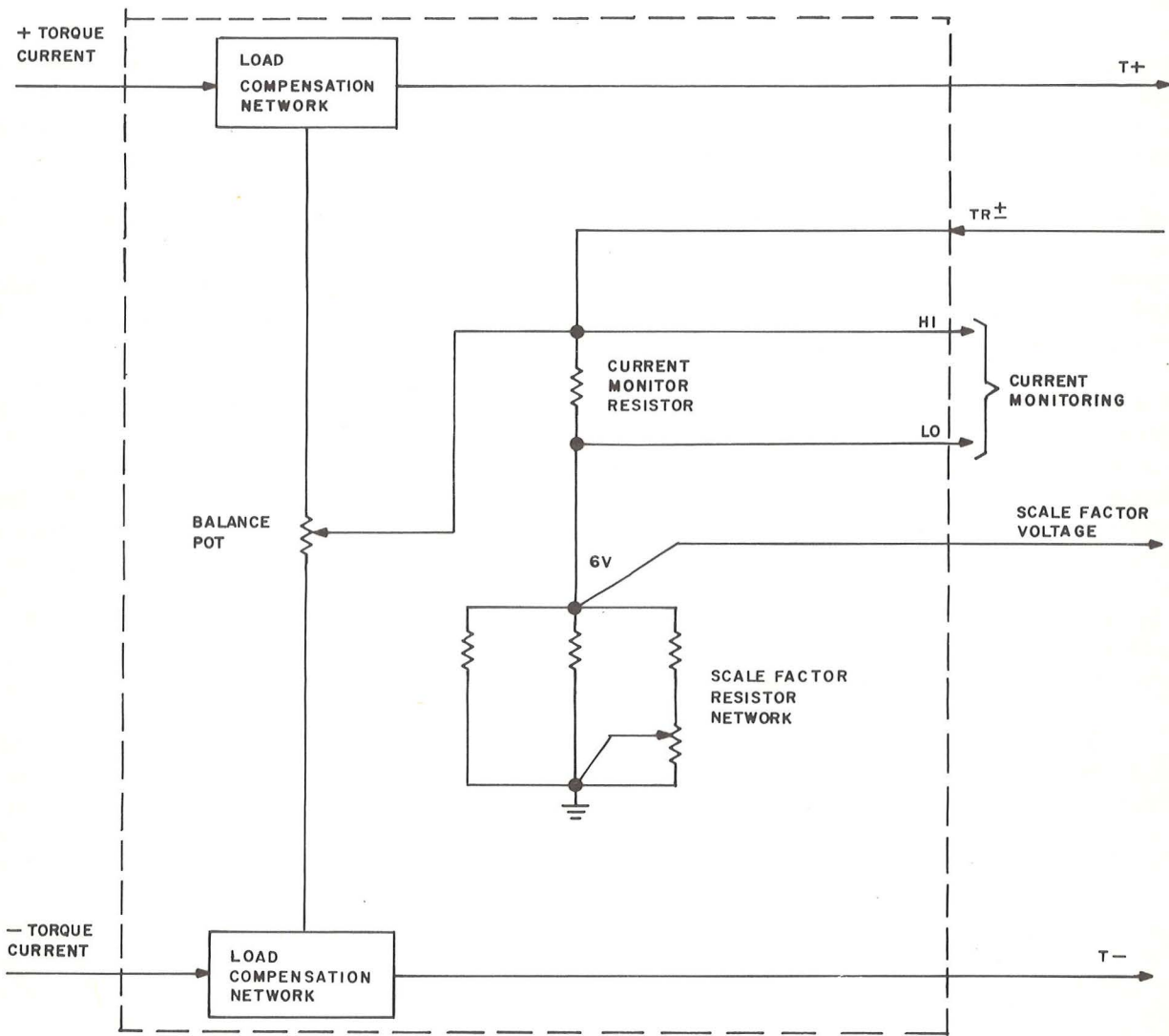


Figure 2-13. PIPA Calibration Module

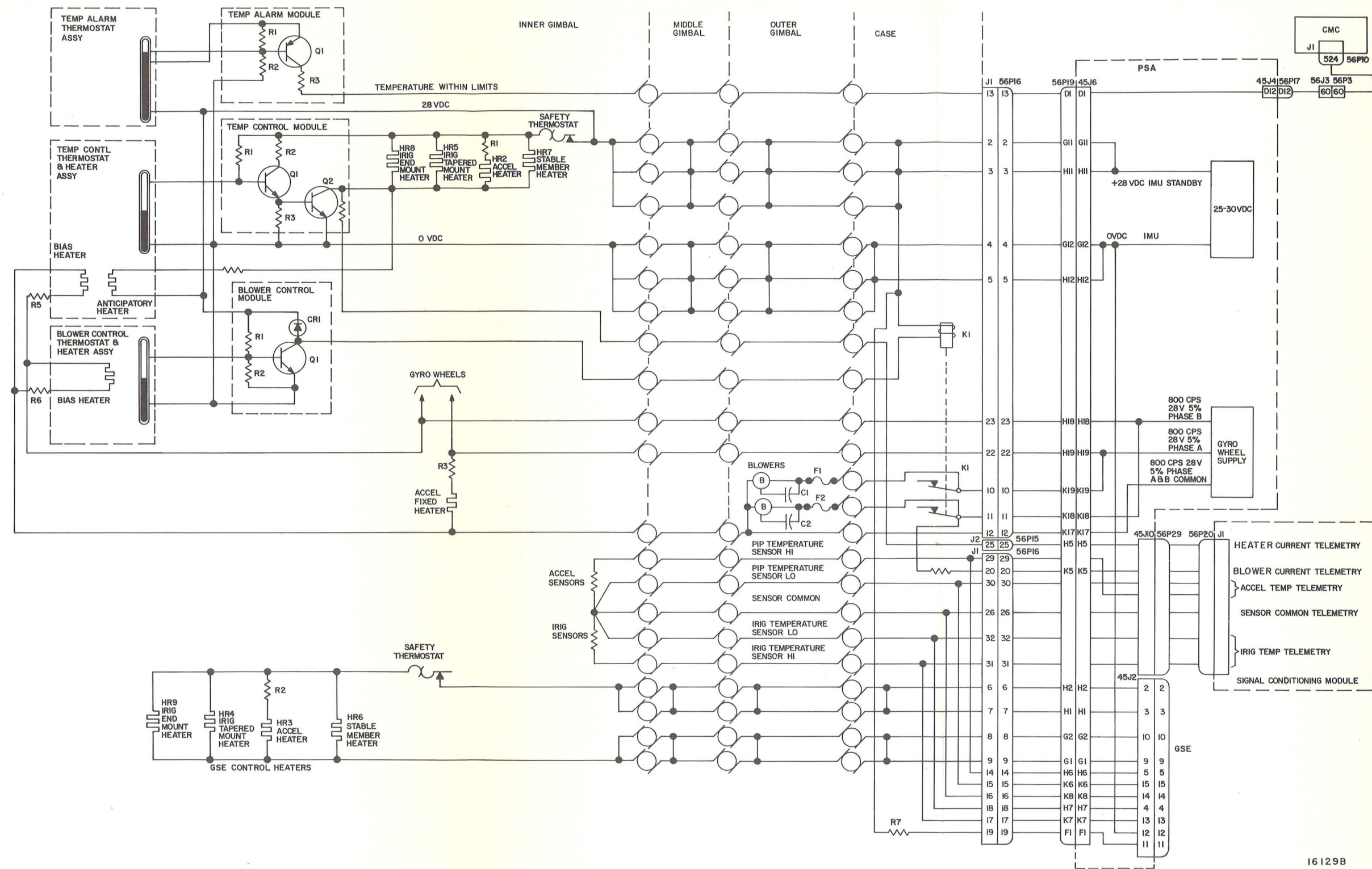
from the nominal 100 milliamperes will increase or decrease the scale factor resistor voltage and result in an output error signal from the dc differential amplifier. This error signal controls the current regulator in the binary current switch. The current regulator, which is in series with the 120 volt dc source and the ducosyn torque generator coils, will maintain the torque current at 100 milliamperes.

2-4.4 IMU TEMPERATURE CONTROL SYSTEM. The IMU temperature control system (figures 2-14 and 2-15) maintains the temperature of the stabilization gyros and accelerometers within the required temperature limits. The system supplies and removes heat to maintain the proper IMU temperature with minimum power consumption. Heat is removed by convection, conduction, and radiation. The natural convection used during IMU standby mode changes to blower controlled, forced convection during IMU operating modes. The IMU internal pressure is maintained between 3.5 and 15 psia to enable the required forced convection. To aid in removing heat, a water-glycol solution from the spacecraft coolant system at approximately 45 degrees Fahrenheit passes through the coolant passages in the IMU case.

2-4.4.1 Temperature Control Circuit. The temperature control circuit maintains the gyro and accelerometer temperature. The temperature control circuit consists of a temperature control thermostat and heater assembly, a temperature control module, three IRIG end mount heaters, three IRIG tapered mount heaters, two stable member heaters, and three accelerometer heaters. The thermostat and heater assembly is located on the stable member and contains a mercury-thallium thermostat, a bias heater, and an anticipatory heater. Except for the bias heater, all heaters (a total of 12) are connected in parallel and are energized by 28 volts dc through the switching action of transistor Q2, which completes the dc return path. The thermostat which senses the temperature of the stable member controls switching transistors Q1 and Q2.

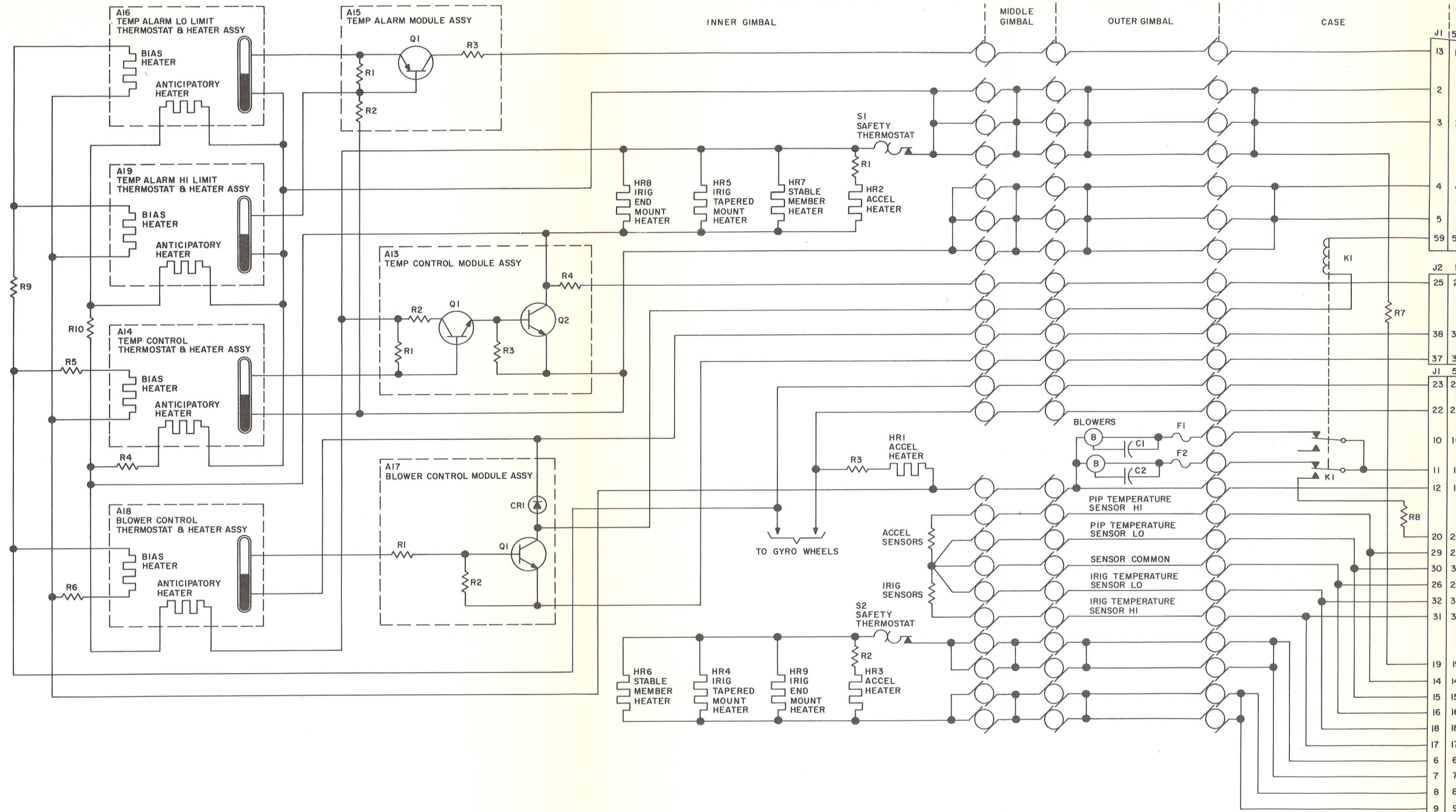
When the temperature falls below 130 (± 0.2) degrees Fahrenheit, the thermostat opens, transistors Q1 and Q2 conduct, and current flows through the twelve heaters. Because of the large mass of the stable member, its temperature will increase at a relatively slow rate as compared to the gyros which have a heater in each end mount. The anticipatory heater improves the response of the thermostat to insure that the magnitude of the temperature cycling of the gyros and the accelerometers is as small as possible. When the temperature rises above 130 (± 0.2) degrees Fahrenheit, the thermostat closes and the base of transistor Q1 is shorted to ground, cutting off transistors Q1 and Q2 and deenergizing the heaters. The thermostat has a 0.5 degree deadband which is the difference between contact closing with rising temperature and contact opening with falling temperature. The temperature control circuit will maintain the average of the gyro temperatures at 135 (± 1) degrees Fahrenheit and the average of the accelerometer temperatures at 130 (± 1) degrees Fahrenheit under normal ambient conditions. The temperature difference between the gyros and the accelerometers is adjusted by properly proportioning the amount of power in each heater. The balance is obtained by decreasing the current through each accelerometer heater by the proper selection of resistor R1.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM



16129B

Figure 2-14. IMU Temperature Control System for PGNCS Part Number 2015000-011



BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

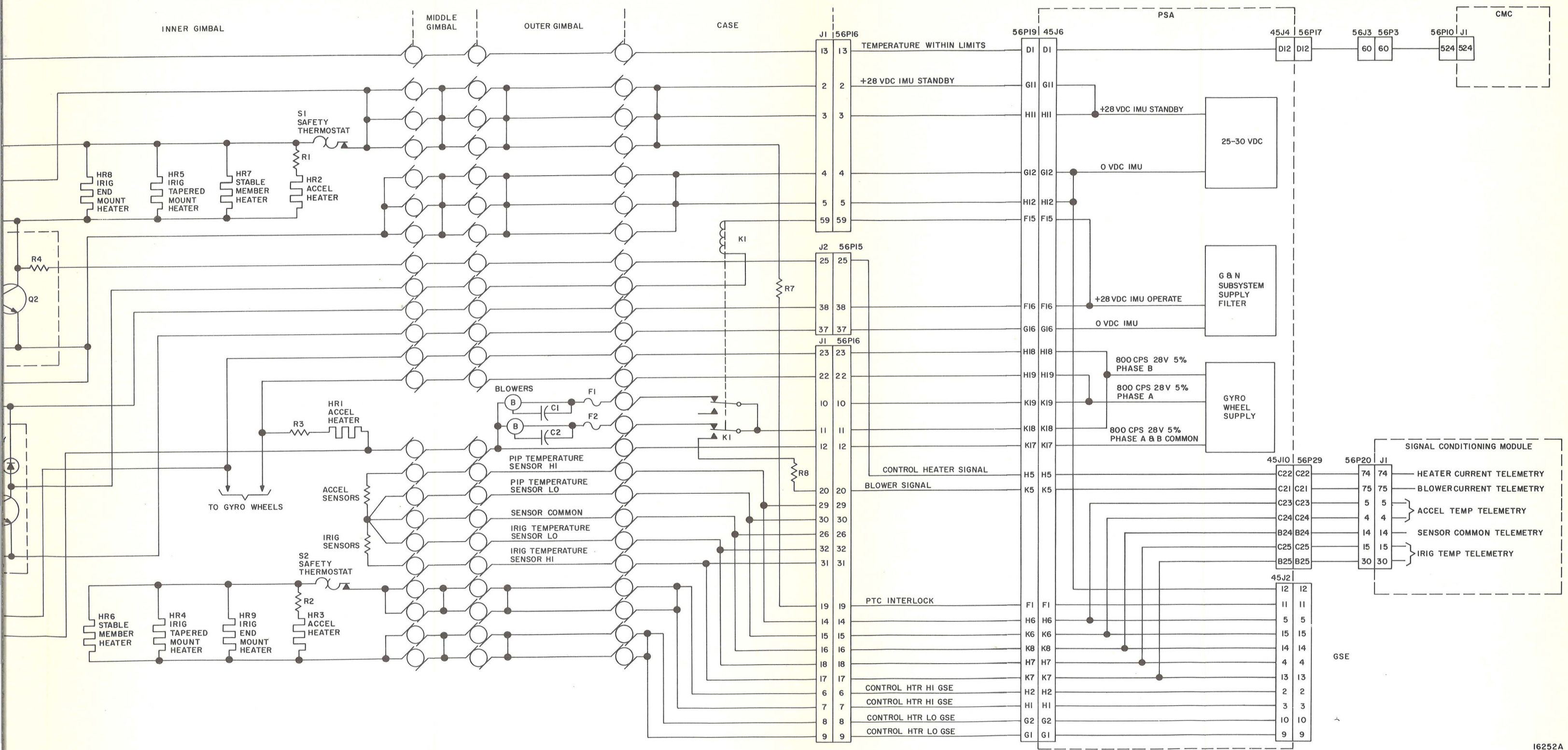


Figure 2-15. IMU Temperature Control System for PGNCS Part Number 2015000-021

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

During IMU operation, power is applied to the fixed accelerometer heaters to compensate for the additional heat supplied to the gyros by the gyro wheel motor heat dissipation. Power is also applied to a bias heater on the control thermostat. The bias heater supplies a fixed amount of heat to the control thermostat to maintain the proper absolute temperature level of the gyros and accelerometers. The amount of bias heat is determined by the value of R5. The power for the fixed accelerometer heaters and the thermostat bias heater is the -90 degree output from the 28 vac power supply which is also used for gyro wheel power.

The 28 vdc heater power is applied to the heaters through the contacts of a safety thermostat which provides protection against an extreme overheat condition in case a malfunction occurs in the temperature control circuit. The safety thermostat contacts open at 139.5 (± 3.0) degrees Fahrenheit and close at 137 (± 3) degrees Fahrenheit.

2-4.4.2 Blower Control Circuit for PGNCS Part Number 2015000-011. The blower control circuit (figure 2-14) maintains IMU heat balance by removing heat. The blower control circuit consists of a blower control thermostat and heater assembly, a blower control module assembly, two axial blowers, and a relay. Although the blower control circuit is active, it is not utilized in this system. The power inputs to the blowers by-pass the contacts of K1 to allow the blowers to operate continuously during IMU operate modes.

The blowers are supplied from the -90 degree output of the 28 volt, 800 cps, 5 percent power supply which also provides gyro wheel motor power. Fused phase shift networks are associated with each blower so that excitation and control current can be supplied from the same source.

2-4.4.3 Blower Control Circuit for PGNCS Part Number 2015000-021. The blower control circuit (figure 2-15) maintains IMU heat balance by removing heat. The blower control circuit consists of a blower control thermostat and heater assembly, a blower control module assembly, two axial blowers, and a relay. The contacts of the thermostat contained in the blower control thermostat and heater assembly close at 139 (± 0.2) degrees Fahrenheit and remain closed at higher temperatures. Resistor R6 is provided to limit the current through the bias heater in the blower control thermostat and heater assembly. The amount of heat supplied by the bias heater is a constant. If the duty cycle of the temperature control circuit exceeds 50 percent, enough additional heat will be provided by the anticipatory heater to increase the temperature of the blower control thermostat and heater assembly to 139 degrees Fahrenheit. When the thermostat contacts close, transistor Q1 conducts and relay K1 is energized to remove the power from the blowers. The normal duty cycle of the temperature control circuit, with the IMU in a 75 degree Fahrenheit ambient temperature, is approximately 15 to 20 percent. Under this condition the blowers will operate continuously. Only a very low ambient temperature will cause a blower off condition.

Power to the blowers is supplied from the -90 degree output of the 28 volt, 800 cps, 5 percent power supply which also provides gyro wheel motor power. Fused phase shift networks are associated with each blower so that excitation and control current can be supplied from the same source.

2-4.4.4 Temperature Alarm Circuit for PGNCs Part Number 2015000-011. The temperature alarm circuit which monitors the temperature control system, consists of a temperature alarm thermostat assembly and a temperature alarm module. If a high or low temperature is sensed by the temperature alarm thermostat (located on the stable member) a discrete is sent to the CMC and to the IMU auxiliary module. When the temperature is within the normal range, 28 volts dc is applied through the thermostat to the emitter of transistor Q1 causing the transistor to conduct through a grounding system in the CMC.

When the temperature falls below 126.3 (± 1.0) degrees Fahrenheit, 28 volts dc is removed from transistor Q1 causing it to stop conducting. When the temperature rises above 134.3 (± 0.2) degrees Fahrenheit, 28 volts dc is applied directly to the base as well as to the emitter of transistor Q1 causing it to stop conducting. Non-conduction of transistor Q1 signals the CMC of an alarm condition. There is no differentiation between a high or low temperature alarm. When the CMC senses a temperature alarm, it causes the TEMP and PGNCs lamps to light. When the IMU auxiliary module receives a temperature alarm, it sends the information to telemetry.

2-4.4.5 Temperature Alarm Circuit for PGNCs Part Number 2015000-021. The temperature alarm circuit which monitors the temperature control system, consists of a temperature alarm high limit thermostat and heater assembly, a temperature alarm low limit thermostat and heater assembly, and a temperature alarm module assembly. If a high or low temperature is sensed by the thermostats located on the stable member a discrete is sent to the CMC and to the IMU auxiliary module. When the temperature is within the normal range, the low limit thermostat contacts are closed and the high limit thermostat contacts are open. Transistor Q1 is then properly biased for conduction through a grounding system in the CMC.

At temperatures below 126.0 (± 0.2) degrees Fahrenheit both the low limit thermostat contacts and the high limit thermostat contacts are open. At temperatures above 134.0 (± 0.2) degrees Fahrenheit both the low limit thermostat contacts and the high limit thermostat contacts are closed. In either case, transistor Q1 is not able to conduct. Non-conduction of transistor Q1 signals the CMC of an alarm condition. There is no differentiation between a high or low temperature alarm. When the CMC senses a temperature alarm, it causes the TEMP and PGNCs lamps to light. When the IMU auxiliary module receives a temperature alarm, it sends the information to telemetry.

2-4.4.6 External Temperature Control. External temperature control of the IMU is provided by ground support equipment (GSE) control heater circuits in the IMU. The circuits are controlled externally to the airborne equipment by the portable temperature controller or the temperature monitor control panel of the optics inertial

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

analyzer (OIA). The GSE control heater circuitry consists of a safety thermostat, six gyro heaters, two stable member heaters, three accelerometer heaters, temperature indicating sensors, and an IMU standby power sensor which disables the GSE when airborne power is on. The temperature indicating sensors act as the control sensing element of the external control and indicating circuitry. The heaters are connected in parallel. The six gyro temperature indicating sensors (two in each gyro) are connected in series to sense the average temperature of the gyros. The three accelerometer temperature indicating sensors (one in each accelerometer) are connected in series to sense the average temperature of the accelerometers. All of the GSE control heater circuitry is electrically independent of the airborne temperature control system. The GSE heater control circuitry is not available as a backup system in flight.

2-4.5 COUPLING DATA UNIT. The coupling data unit (CDU) contains circuits to perform the following functions:

(1) Convert IMU gimbal resolver outputs into digital information which is stored in the read counter and transmitted to the CMC in the form of $\pm \Delta \theta G$ pulses.

(2) Accept $\pm \Delta \theta C$ pulses from the CMC, store these pulses in the binary error counter, and provide one dc and two ac signal outputs proportional to the stored pulses.

Any change in the IMU gimbal angles is sensed by the IMU gimbal resolvers (see figure 2-16) which generate angular information ($\sin \theta G$, $\cos \theta G$, $\sin 16 \theta G$, and $\cos 16 \theta G$) proportional to the IMU gimbal angles. This information is routed to the coarse and fine switching network which provides coarse and fine error signals to the digital-to-analog (D/A) converter as well as a ternary level signal to the read counter logic circuits. The read counter logic circuits provide up or down signals and drive pulses to increment or decrement the read counter as required to contain a digital representation of the gimbal angle. A coarse or fine system is used to increment or decrement the read counter. The coarse system is used if the IMU gimbal resolver signal is greater than 0.1 degree and causes the read counter to be incremented or decremented at a 12.8 kpps rate. The fine system is used if the IMU gimbal resolver signal is less than 0.1 degree and causes the read counter to be incremented or decremented at an 800 pps rate. The read counter can follow the IMU gimbal angles to within ± 20 arc seconds. The read counter transmits $\pm \Delta \theta G$ pulses (equivalent to approximately 40 arc seconds of gimbal angle rotation) to the CMC. The CMC utilizes these signals to determine the actual IMU gimbal angles and to display the angles on the DSKY.

During the IMU coarse align mode, the CMC compares the $\pm \Delta \theta G$ pulses from the read counter logic circuits (actual gimbal angles) with the CMC calculated (desired) gimbal angles and generates $\pm \Delta \theta C$ pulses which are routed to the error counter logic circuits. The CMC also supplies coarse align enable and enable error counter signals to the ISS moding logic to enable these circuits. The error counter logic circuits provide up or down signals and drive pulses to the error counter to cause it to be incremented or decremented as required. The D/A converter accepts ladder network

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

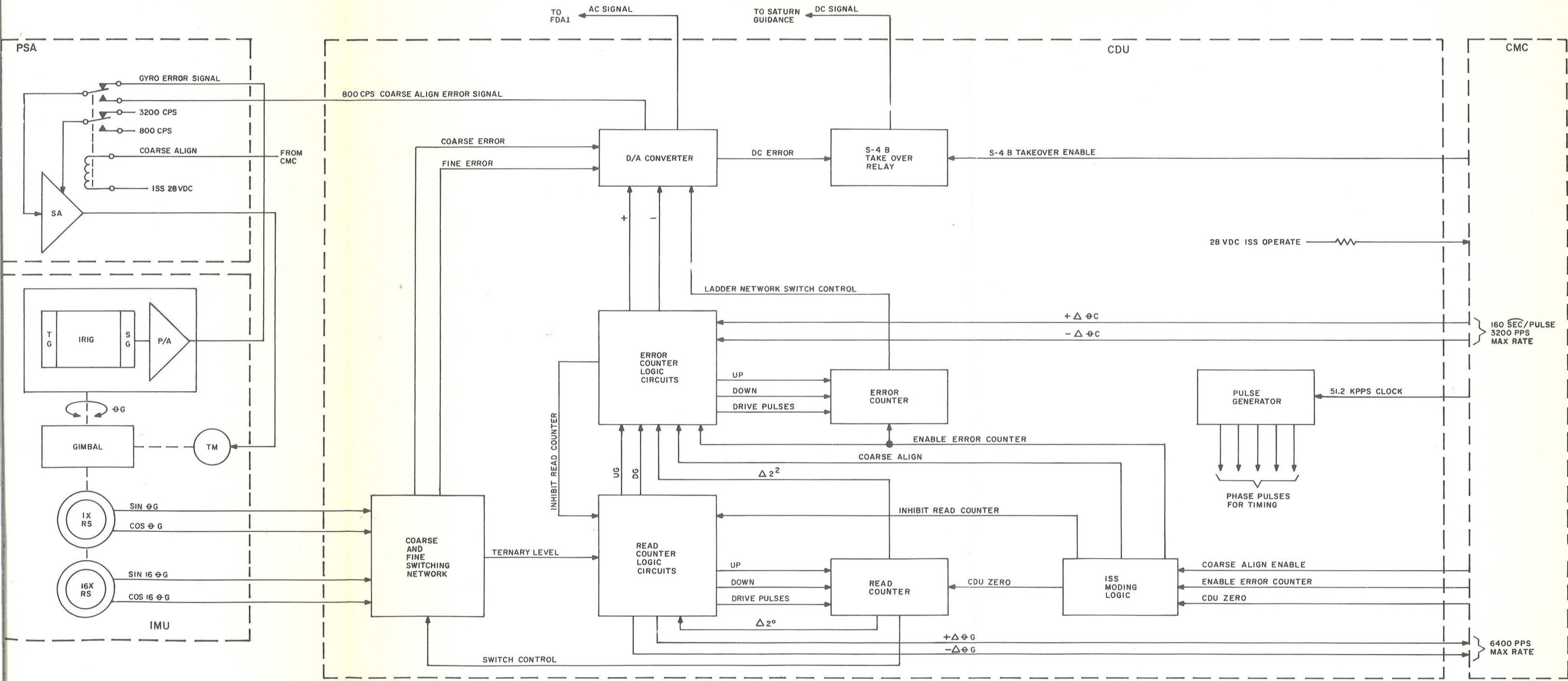


Figure 2-16. ISS-CDU Moding

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

switch control signals from the error counter and + or - control signals from the error counter logic circuits. The D/A converter provides an 800 cps coarse align error signal to the gimbal servo amplifiers to drive the gimbals to the desired angle. As the gimbals are positioned to the new angles, the change is detected by the resolvers which causes the read counter to be incremented or decremented in proportion to the change. A $\Delta 2^2$ signal is provided from the read counter to the error counter logic circuits to cause the error counter to be decremented when the read counter is incremented to null or to clear the error counter when the IMU gimbal is at the desired angle. The D/A converter also supplies signals to the FDAI and, if the S-4B take-over relay is enabled by the CMC, to the Saturn guidance system.

2-4.5.1 ISS CDU Discrettes. All CMC discrettes issued to the CDU to initiate and control the various ISS modes or functions are 0 (± 2) vdc, CMC ground, applied through a 2,000 ohm source impedance to the CDU mode module. ISS CDU discrettes supplied by the CMC are: CDU zero, enable error counter, coarse align enable, and S-4B take-over.

2-4.5.1.1 CDU Zero. The CDU zero discrete clears all three CDU read counters simultaneously. It also inhibits the transmission of incrementing pulses to the read counters for the duration of the discrete. The CDU zero discrete is present for as long as the read counters are to be held at zero. The minimum duration is approximately 400 milliseconds. The IMU is not changed by the CDU zero discrete.

2-4.5.1.2 Enable Error Counter. The enable error counter discrete enables all three error counters simultaneously which allows them to accept incrementing pulses ($\pm \Delta \theta C$) from the CMC. The error counters are normally cleared and inhibited. The enable error counter discrete is used in conjunction with the coarse align enable discrete during the coarse align mode, and the S-4B take-over discrete. The enable error counter discrete is used alone only when display of attitude error signals on the FDAI is required.

2-4.5.1.3 Coarse Align Enable. The coarse align enable discrete enables a relay driver which energizes the coarse align and demodulator reference relays located in the PSA. This connects the coarse align error signal to the gimbal servo amplifiers and changes the reference voltage for the demodulator in the gimbal servo amplifiers from 3,200 cps to 800 cps. The discrete also enables the digital feedback pulses from the read counter to the error counter. The presence of the coarse align enable discrete and the absence of the enable error counter discrete inhibit the incrementing pulses to the read counter.

2-4.5.1.4 S-4B Take-over. Prior to the CMC generating the S-4B take-over discrete, the CMC checks to see if the error counters are in use. If the error counters are in use, the CMC removes the enable error counter discrete and, after a 1/2-second delay, issues the S-4B take-over discrete. The CMC generates the enable error counter discrete 40 milliseconds after the S-4B take-over discrete and, after a 4 millisecond delay, can send $\pm \Delta \theta C$ pulses to the error counter. The S-4B take-over discrete energizes the control relays which connect the error counter D/A converter dc outputs to the Saturn guidance electronics for Saturn booster vehicle steering control. During

this mode, the digital feedback pulses from the read counter to the error counter are inhibited and the CMC monitors spacecraft position and velocity by use of the PIPA's and the ISS CDU read counters.

2-4.6 ISS MODES OF OPERATION. The ISS has five major modes of operation which are: IMU turn-on, ISS CDU zero, coarse align, S-4B take-over, and thrust vector control. Sub-modes are: IMU cage, attitude error indication, fine align, inertial reference, and attitude control. All ISS moding is initiated and controlled by CMC discrettes to the CDU. The CMC may send a single discrete, a combination of discrettes, or no discrete at all to select an ISS mode of operation.

2-4.6.1 IMU Turn-on. The IMU turn-on mode is a backup mode used to drive the gimbals to their zero position and hold them there during the IMU turn-on period. (See figure 2-17.) The IMU turn-on mode is enabled upon closure of the ISS OPERATE circuit breaker and allows a 90-second gyro run up period. The ISS OPERATE circuit breaker routes 28 vdc IMU operate power through the deenergized contacts of the ISS turn on control relay, located in the IMU auxiliary assembly module, to the cage relays. The 28 vdc IMU operate power is also routed through the same deenergized contacts to the CMC as a turn on delay request discrete. The cage relays energize and, in turn, cause the coarse align relays to be energized. The cage relays route the 1X gimbal resolver sine winding outputs through the contacts of the coarse align relays to the respective gimbal servo amplifiers. The gimbal servo amplifiers drive the gimbals until the resolver signals are nulled. The operation of the caging loops is discussed further in the IMU cage mode description.

Upon receipt of the ISS turn on delay request discrete, the CMC sends the ISS CDU zero discrete and the coarse align enable discrete to the CDU for a minimum period of 90 seconds. The CDU zero discrete clears the read counters and inhibits the incrementing pulses to the read counters. The coarse align enable discrete provides a redundant means of energizing the coarse align relays.

A second set of deenergized contacts on the ISS turn on control relay routes a ground to the time delay circuit of the pulse torque power supply which inhibits the operation of the power supply and thus prevents accelerometer pulse torquing during the 90 second turn on period. This allows time for the accelerometer floats to be centered and the gyro wheels to run up prior to torquing.

After the 90 second delay has been completed, the CMC sends the ISS turn on delay complete discrete. This discrete acts through a relay driver to energize and latch in the ISS turn on control relay. Energizing the ISS turn on control relay deenergizes the cage relay, removes the ISS turn on delay request discrete, and removes the inhibit from the pulse torque power supply. The computer program can then place the ISS in the inertial reference mode by removing both the CDU zero and the coarse align enable discrettes, or it can initiate the coarse align mode by removing only the CDU zero discrete and sending the ISS enable error counter discrete. The IMU turn on circuit will be reset whenever 28 vdc IMU operate power is turned off.

2-4.6.2 ISS CDU Zero. The purpose of the ISS CDU zero mode is to clear and inhibit the three ISS CDU read counters. The mode is initiated by the CMC sending the CDU zero discrete. The presence of the discrete is maintained for as long as the read counters are to be held at zero.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

2-4.6.3 Coarse Align Mode. The purpose of the coarse align mode is to change the orientation of the gimbals by CMC command. This is accomplished by feeding the CDU error counter CMC pulses equal to the required change in gimbal angles. The mode is initiated by the CMC sending the coarse align discrete (followed in 40 msec by the enable error counter discrete) to the three ISS channels of the CDU. The CMC, knowing the actual gimbal angle registered in the read counters, can calculate the desired amount of change in gimbal angles required to reposition the gimbals to the desired angles. The CMC converts this change into a number of $\pm\Delta\theta C$ pulses and sends these pulses to the error counters. The $\pm\Delta\theta C$ pulses are sent to the error counters at a rate of 3,200 pps in bursts of 60 milliseconds. Bursts of pulses are sent 480 milliseconds apart. Each $\Delta\theta C$ pulse is equal to a change in gimbal angle of 160 arc seconds. The error counter, having been enabled, accepts the pulses and counts up or down as necessary until all the pulses have been registered.

The ladder decoder in the D/A converter module supplies an 800 cps, amplitude modulated analog error signal equivalent to the digital information in the error counter. The ladder decoder signal is summed with a feedback signal and applied through a mixing amplifier, also located in the D/A converter module, to the gimbal servo amplifiers to drive the gimbals to the desired angles. The coarse align error signal output of the mixing amplifier is applied to the gimbal servo amplifiers through the contacts of the coarse align relays located in the PSA. The coarse align relays, which are energized by the coarse align enable discrete acting through a relay driver, switch the input of the gimbal servo amplifiers from the gyro preamplifiers to the coarse align error signal output of the ISS D/A converters. The demodulator reference relay is also energized by the coarse align enable discrete and switches the reference frequency of the demodulator in the gimbal servo amplifiers from 3,200 cps to 800 cps. The coarse align enable discrete also energizes a relay in the gimbal servo amplifiers which switches additional capacitance into the amplifier's compensation networks to tune them for 800 cps operation.

As the gimbals are driven, $\pm\Delta 2^2$ pulses, representing the change in actual gimbal angle, are generated by the read counter and applied to the error counter. The $\pm\Delta 2^2$ pulses are also equal to 160 arc seconds and decrease the $\pm\Delta\theta C$ pulses registered in the error counter. The error counter output to the ladder decoder, therefore, represents the difference between the desired amount of change in gimbal angle and the amount of change actually accomplished. The error counter reaches a null or reads zero, and the gimbals stop moving, when the actual gimbal angle has changed by an amount equal to the total value of the $\pm\Delta\theta C$ pulses sent by the computer to the error counter. The CMC checks the contents of the read counter 2.1 seconds after the CMC has sent the last coarse align torquing pulse to the error counter. If the gimbal angles are not within two degrees of the desired angles, the computer issues an alarm.

The rate at which the gimbals are driven is limited to prevent damage to the gyros and to assure that the read counter can track the gimbal angle accurately. The rate of gimbal movement is limited by feeding back the CDU fine error signal $\sin 16 (\theta - \psi)$ to the input of the mixing amplifier located in the D/A converter module. The coarse error signal is also fed back to the mixing amplifier but is used only during the turn-on mode. The fine error signal is out of phase with the output of the ladder decoder and has an amplitude proportional to the difference between the actual gimbal angle (θ) and the angle registered in the read counter (ψ). The fine error signal is applied through a voltage limiting circuit to the summing junction of the mixing amplifier where it is

summed with the 800 cps ladder decoder output signal. The D/A converter ladder decoder output is applied to the mixing amplifier through a scaling amplifier and a voltage limiting diode network. The scaling amplifier controls the signal gain to produce a scale factor of 0.3 volt rms per degree. The output of the mixing amplifier will be at a null when the D/A converter ladder decoder output, after limiting, is equal to the fine error feedback signal. The fine error signal will be a constant value only when the gimbal and the CDU are going at the same rate and with the gimbal angle leading the CDU angle. Since the CDU is constrained to counting at one of two speeds, the gimbals will be constrained to a rate equal to one of these two speeds. During the coarse align mode, the CDU is constrained to a high counting speed of 6.4 kpps and a low counting speed of 800 cps. At all other times, the high counting speed is 12.8 kpps.

If the gimbals are moving at a faster rate than the rate at which the CDU is counting, the fine error signal will increase causing a retarding torque to be developed by the gimbal servo amplifier. If the gimbals are moving at a rate slower than the rate at which the CDU is counting, the fine error signal will decrease causing the gimbal servo amplifier to apply an accelerating torque to the gimbals. By adjusting the gain of the fine error signal into the mixing amplifier, the gimbal drive rate is limited to either 35.5 degrees per second (6.4 kpps CDU counting rate) or 4.5 degrees per second (800 cps CDU counting rate).

2-4.6.4 S-4B Take-over. During the S-4B take-over mode, the CMC controls spacecraft steering by sending attitude error signals to the Saturn guidance system through the ISS CDU. This mode is a backup for the Saturn guidance system if the Saturn guidance system IMU fails during earth ascent, translunar injection, or earth orbit. The S-4B take-over mode is initiated by the astronaut activating a switch on the main D and C panel. Closure of this switch routes a +28 vdc discrete to the ISS CDU where it arms a set of relays which, when energized, connect the dc outputs of the D/A converters to the Saturn guidance system. The +28 vdc discrete is also routed to the CMC. Receipt of this discrete by the CMC signals the CMC to issue the S-4B take-over discrete. The S-4B take-over discrete energizes the control relays which connect the D/A converters dc outputs to the Saturn guidance system. The CMC issues the enable error counter discrete 40 milliseconds after the S-4B take-over discrete. With the error counters enabled and the D/A converters dc outputs connected to the Saturn guidance system, the CMC can send steering signals to the Saturn guidance system. The CMC calculates the amount of yaw, pitch, and/or roll required for proper spacecraft attitude and determines the amount of $\pm\Delta\theta C$ pulses to be sent to the error counters. The error counters receive the $\pm\Delta\theta C$ pulses and count up or down as required until all pulses have been received. The error counters send this digital information to the D/A converters where it is converted to a dc signal and sent to the Saturn guidance system for attitude control. The CMC monitors the spacecraft response to the attitude signals through the read counters and the PGNCS PIP's. During this mode, the digital feedback signals from the read counters to the error counters are inhibited.

Prior to the CMC issuing the S-4B take-over discrete, the CMC checks to see if the error counters are in use. If the error counters are in use, the CMC removes the enable error counter discrete which inhibits and clears the error counters. The S-4B take-over discrete will then be issued after a 1/2-second delay.

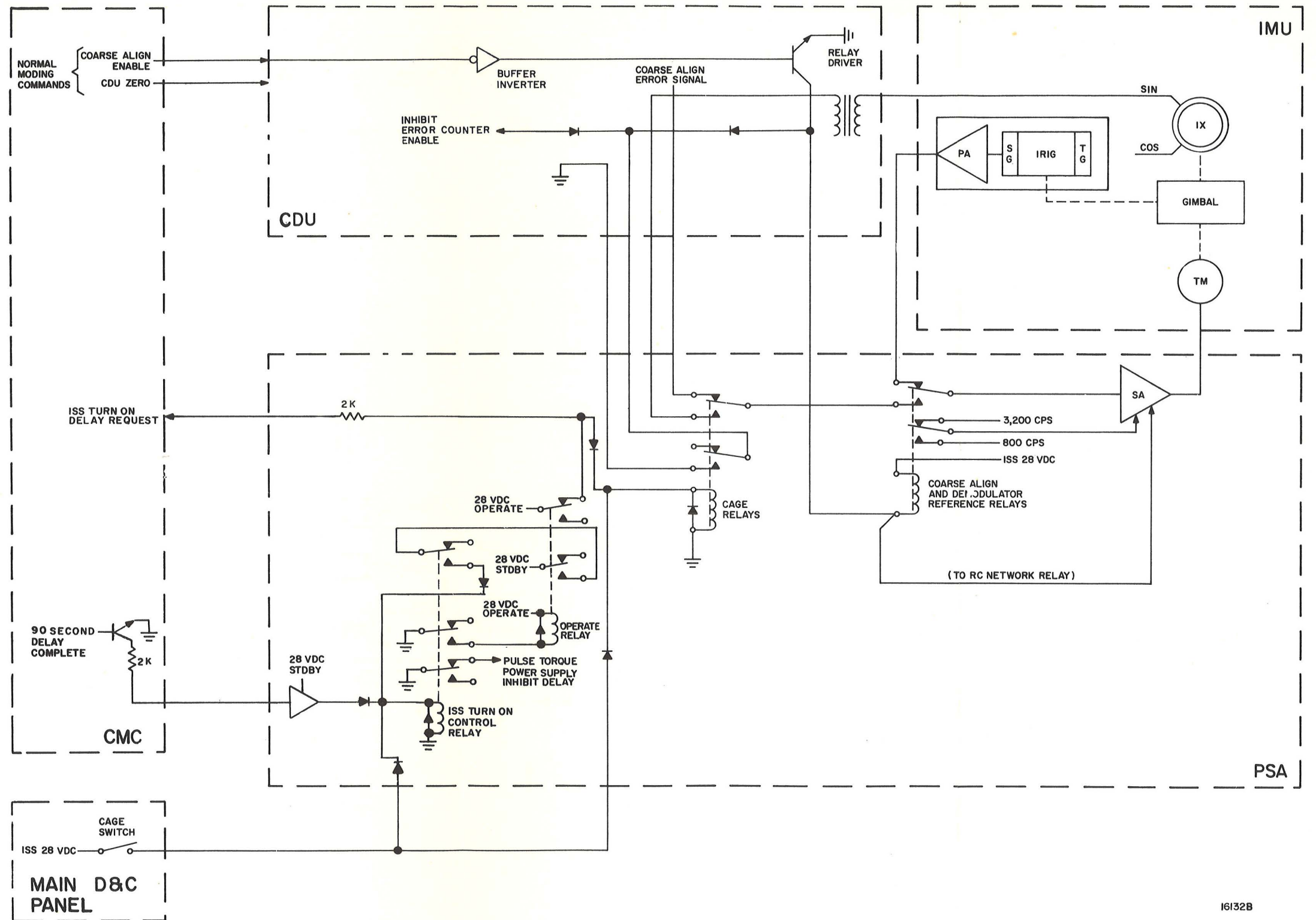
2-4.6.5 Thrust Vector Control. During the thrust vector control mode, the CMC controls the SPS engine gimbals by sending steering control signals to the SPS engine gimbal amplifiers through the optics CDU. The thrust vector control mode is initiated by the astronaut activating a switch on the main control panel. Closure of this switch routes a +28 vdc discrete to the optics CDU where it arms a set of relays which, when energized, connect the dc outputs of the optics CDU D/A converters to the SPS engine gimbal amplifiers. The +28 vdc discrete is also routed to the CMC. Receipt of this discrete by the CMC, signals the CMC to issue the thrust vector control discrete. The thrust vector control discrete energizes the control relays which connect the optics CDU D/A converters dc outputs to the SPS engine gimbal amplifiers. The CMC issues the optics D/A enable discrete 40 milliseconds after the thrust vector control discrete. With the optics error counters enabled and the optics CDU D/A converters dc outputs connected to the SPS engine gimbal amplifiers, the CMC can send steering control signals to position the SPS engine gimbals. The CMC calculates the amount of pitch or yaw required for proper trajectory and then determines the amount of $\pm\Delta\theta C$ pulses to be sent to the error counters. The error counters receive the $\pm\Delta\theta C$ pulses and count up or down as required until all pulses have been received. The error counters send this digital information to the D/A converters where it is converted to a dc signal and routed to the SPS engine gimbal amplifiers. The CMC monitors spacecraft response to the engine gimbal steering signals through the ISS CDU read counters and the PGNCS PIP's. During this mode, the digital feedback signals from the read counters to the error counters are inhibited.

Prior to the CMC issuing the thrust vector control discrete, the CMC checks to see if the optics CDU error counters are in use. If the error counters are being used, the CMC removes the optics D/A enable discrete which inhibits and clears the error counters. The CMC will then issue the thrust vector control discrete.

2-4.6.6 IMU Cage Mode. The IMU cage mode (figure 2-17) is an emergency backup mode which will allow the astronaut to recover a tumbling IMU by setting the gimbals to zero. During this mode, the 1X gimbal resolver sine winding outputs are fed through the CDU's to the gimbal servo amplifiers to drive the gimbals until the resolver signals are nulled.

The IMU cage mode is initiated by the astronaut pressing the IMU CAGE switch. The switch is held until the gimbals settle at the zero position (five seconds maximum). The gimbal position may be observed on the FDAI. The IMU CAGE switch routes a 28 vdc discrete signal to the CMC and to the cage relays located in the PSA. The cage discrete energizes the cage relays which in turn cause the coarse align relays, the demodulator reference relay, and a relay in the gimbal servo amplifiers to be energized. The relay in the gimbal servo amplifiers switches additional capacitance into the RC compensation networks to tune them for 800 cps operation. The demodulator reference relay changes the gimbal servo amplifier demodulator reference signal from 3,200 cps to 800 cps. The cage relays switch the 1X gimbal resolver sine winding outputs through the energized contacts of the coarse align relays into the corresponding gimbal servo amplifier inputs. The gimbal servo amplifiers drive the gimbals until the resolver signals are nulled which occurs at zero gimbal angle.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM



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Figure 2-17. IMU Cage Mode

Rev. J

2-41/2-42

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

Upon receipt of the IMU cage discrete, the CMC will discontinue sending the enable error counter discrete, the coarse align enable discrete, and the incrementing pulses to the CDU. The CMC will also discontinue sending torquing commands to the fine align electronics if any are in process.

After the IMU CAGE switch is released, the CMC will allow the read counters to settle, then will place the PGNCS in an attitude control mode. During the time the IMU cage discrete is present and while the read counters are settling, the NO REFERENCE lamp on the DSKY is lighted.

The IMU cage mode will also be entered automatically whenever the IMU is turned on. During the normal turn on sequence, the closure of the ISS OPERATE circuit breaker will route 28 vdc through the deenergized contacts of the operate relay to the cage relays and to the computer. The cage relays energize and cage the gimbals. After the 90 second turn on time delay has been completed, the CMC will (if it is on) send the ISS turn on delay complete discrete which will energize the ISS turn on control relay. A ground is supplied through the energized contacts of the ISS turn on control relay which energizes the operate relay which, in turn, deenergizes the cage relays. If, however, the CMC is off or in standby when the IMU is turned on, the ISS turn on control relay will remain deenergized and the ISS will remain in the IMU cage mode.

If the IMU cage mode is entered as a result of an IMU turn on with the CMC off or in standby, the ISS may be placed in the inertial reference mode by allowing 90 seconds for gyro runup, then pressing the IMU CAGE switch. The IMU CAGE switch will energize and latch the auto cage relay control which removes the ISS 28 vdc which had been energizing the cage relays. With the auto cage relay control latched, the cage relay will deenergize when the IMU CAGE switch is released. Deenergizing the cage relays causes the coarse align relays to be deenergized which connects the gyro error signals to the respective gimbal servo amplifiers. The stabilization loops will maintain the stable member inertially referenced to the orientation established by the caging loops.

2-4.6.7 Attitude Error Indication. The attitude error indication mode supplies attitude error signals to the FDAI. The attitude error indication mode is initiated by the CMC sending the enable error counter discrete to the CDU. The ISS read counters are enabled and register the IMU gimbal angles which are sent to the CMC as $\pm\Delta\theta_G$ pulses. The CMC calculates the difference between the actual gimbal angles registered in the read counters and the desired IMU gimbal angles and, if any difference exists, converts this difference into $\pm\Delta\theta_C$ pulses which are sent to the error counters. The error counter, having been enabled, accepts the pulses and counts up or down until all pulses are registered. The error counter output is applied to the D/A converters where it is converted into an ac signal proportional to the error counters contents and applied to the FDAI.

Any change in actual gimbal angles causes the read counters to increase or decrease proportionally. If the attitude error increases or decreases, the CMC will send additional $\Delta\theta_C$ pulses to decrease or increase, respectively, the error counter contents. The digital information in the error counters is converted into an 800 cps,

amplitude modulated, analog error signal by the ladder decoder in the DAC's. This ac signal, representing the difference between the actual gimbal angles and the desired gimbal angles, is applied through the scaling amplifiers to the FDAI.

2-4.6.8 Fine Align Mode. The purpose of the fine align mode is to reposition the stable member to a fine alignment by torquing the gyros. The fine align mode is actually a gyro torquing function accomplished during the inertial reference mode. The torquing current to the gyros is provided by the fine align electronics located in the PSA. The fine align electronics are enabled and controlled by CMC pulses sent directly to the fine align electronics. The CMC does not send command discrettes to the ISS CDU during this mode. The ISS is in the inertial reference mode prior to the enabling of the fine align electronics and returns to that mode when the fine align electronics are disabled. The computer accepts the angular data received from the optical measurements, transforms the S_t LOS, into stable member coordinates, and compares the desired coordinates with the actual coordinates. Differences are used to compute the angular rotation about each stable member axis which will move the stable member to the correct orientation. The fine align electronics torques the gyros on a time shared basis. The CMC sends four types of pulse trains to the fine align electronics. The first pulse train to be sent is the torque enable command which enables the fine align electronics. The second pulse train is a gyro select command which selects a particular gyro and the direction it is to be torqued by means of a switching network which closes the current path through the proper torque ducosyn coil. The third and fourth types of pulse trains are the torque set and torque reset commands which control a binary current switch to start and stop the current flow through the selected torque ducosyn coil. The amount of current flow through the torque ducosyn coils is precisely controlled at a fixed value. The amount of gyro torquing to be accomplished is determined by the amount of time torque current is applied, that is, the time duration between the receipt of the torque set and the torque reset commands. The factor is calculated by the CMC based on optical alignment measurements. The torque ducosyn displaces the gyro float causing the ducosyn signal generator to apply an error signal to the stabilization loop. The stabilization loops drive the gimbals to reposition the stable member. Upon completion of the torquing, the stable member remains fixed in inertial space, and in fine align mode, until the torque enable command is removed, after which the ISS remains in inertial reference mode until further change is commanded.

During the fine align mode, the ISS error counters remain cleared and inhibited. The read counters continue to repeat the gimbal angles and send angular data ($\pm\Delta\theta G$) to the CMC.

2-4.6.9 Inertial Reference Mode. The purpose of the inertial reference mode is to provide a coordinate reference system on which attitude and velocity measurements and calculations may be based. During the inertial reference mode, the stable member is held fixed with respect to inertial space by the stabilization loops. The ISS CDU read counters provide the CMC with changes in gimbal angles with respect to the stable member. The ISS is in the inertial reference mode during any operating period in which there is an absence of moding commands. During the inertial reference mode, the fine align electronics is inhibited and the ISS error counters are cleared and inhibited.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

2-4.6.10 Attitude Control. The attitude control mode, a PGNC mode, utilizes the digital autopilot to maintain spacecraft attitude. The ISS stabilization loops maintain an inertial reference from which spacecraft attitude can be determined. The spacecraft attitude is compared to a desired attitude by the digital autopilot. If an error exists, the RCS jets are used to change the spacecraft attitude. Attitude errors are displayed on the FDAI.

During the attitude control mode, the error counter is enabled. The computer sends the desired change in gimbal angle orientation to the error counters. The output of the error counters is developed into an analog error signal (attitude error signal) by the 800 cps ladder network and the output is fed to the FDAI. The computer develops discretes that control the firing of RCS jets and, therefore, change the orientation of the spacecraft. The change in spacecraft orientation is sensed by the IMU stabilization loops and results in changing IMU gimbal angles. The gimbal resolver output is sent to the FDAI to display total attitude and to the summing logic and coarse fine mixing circuit. Resulting digital information is sent to the read counter. The read counter supplies the changing gimbal angle pulses to the CMC. The CMC uses the information in determining when to stop firing the RCS jets. The FDAI error counter is pulsed to zero, nulling the attitude error signal on the FDAI.

2-5 OPTICAL SUBSYSTEM

The OSS allows direct visual sightings to be made and precision measurements to be taken on celestial objects by means of the SCT and SXT. The angular data developed is transferred by the CDU into the CMC, which uses this data to calculate spacecraft position and trajectory. In addition, the results obtained are used to align the IMU to a star framework.

Establishment of a LOS to selected celestial targets generally requires use of the SCT and/or SXT together with spacecraft positioning controls. The astronaut changes spacecraft attitude to direct the optics field of view toward the celestial target area. Direct viewing through the wide 60 degree field of the SCT enables target search and recognition. The SXT is then used to take precise measurements.

After target acquisition, the optical angular measurements and the time of sighting are transferred to the CMC. Data pertaining to the location of targets and programs for navigational calculations stored in the CMC are then compared. The results of the comparisons are used to align the IMU and determine spacecraft position and trajectory.

In case of failure in the optics electronics, the astronaut can operate the SCT manually with a universal tool and read the angles off the SCT counters. In such emergencies, the astronaut will calculate, with possible assistance from the ground, a navigational fix to determine position and required velocity corrections.

The OSS consists of an optical unit, two CDU channels, portions of the indicator control panel, and portions of the PSA. The indicator control panel contains the controls and indicators that are used to establish OSS operating modes.

Figure 2-18 is a functional block diagram of the OSS. The diagram shows the flow of data for the various modes of operation selected by mode switching and the functional relationship among the various blocks. Each functional block is described in subsequent paragraphs.

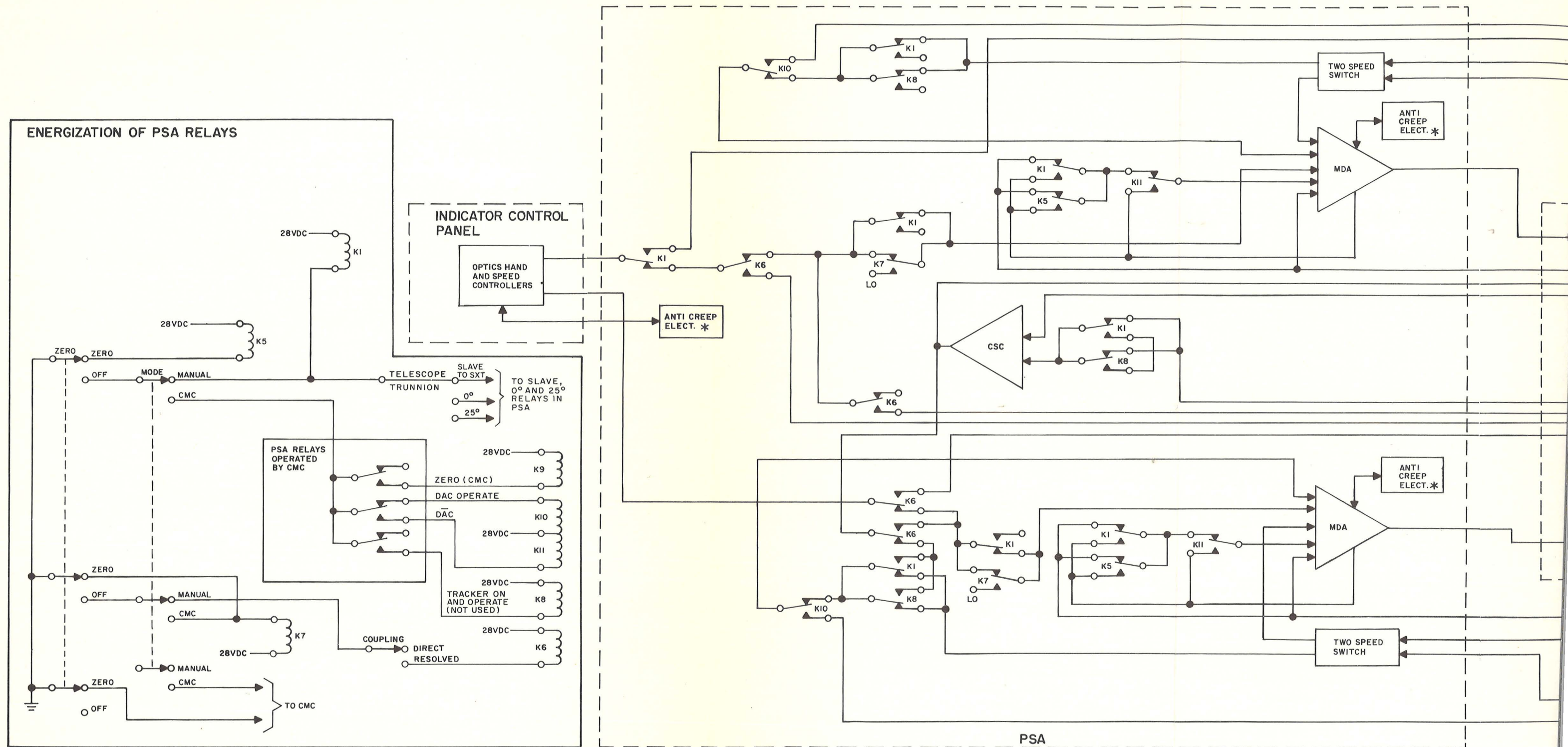
2-5.1 SCANNING TELESCOPE. The SCT is a single LOS, refracting type, low power instrument with 60 degrees instantaneous field of view. The SCT wide field of view is used for general celestial viewing and recognition of target bodies for sighting setups and measurements. In addition, it is used to track landmark points during earth and lunar orbits.

The SCT has two axes of rotational freedom, defined as the shaft and trunnion axes. Rotation about the shaft axis defines increments of shaft angle movement (A_S). Rotation about the trunnion axis defines increments of trunnion angle movement (A_t).

The prime element of the SCT is a double dove prism, (prism), mounted in the head assembly. The prism introduces the target image into the SCT optical system. The prism is positioned about the two axes interpreted as A_S and A_t . Mechanically, this is accomplished by rotation, through a differential, of the complete head assembly and driving of the prism on its mount. A differential gearing system enables independent positioning of A_S and A_t . The independent positioning enables the LOS established through the prism to traverse within the following limits: zero degrees (parallel to shaft axis) to ± 60 degrees at A_t elevation, unlimited 360 degrees in A_S . When the SCT is slaved to the SXT, rotation is limited to 270 degrees due to limit stops along the SXT shaft axis. Due to obstructions created by aperture limitation in the spacecraft frame, the LOS along the trunnion axis has a useful range of approximately 58 degrees. Therefore, the aggregate maximum field that can be viewed by rotating the trunnion and shaft within their respective limits is 116 degrees.

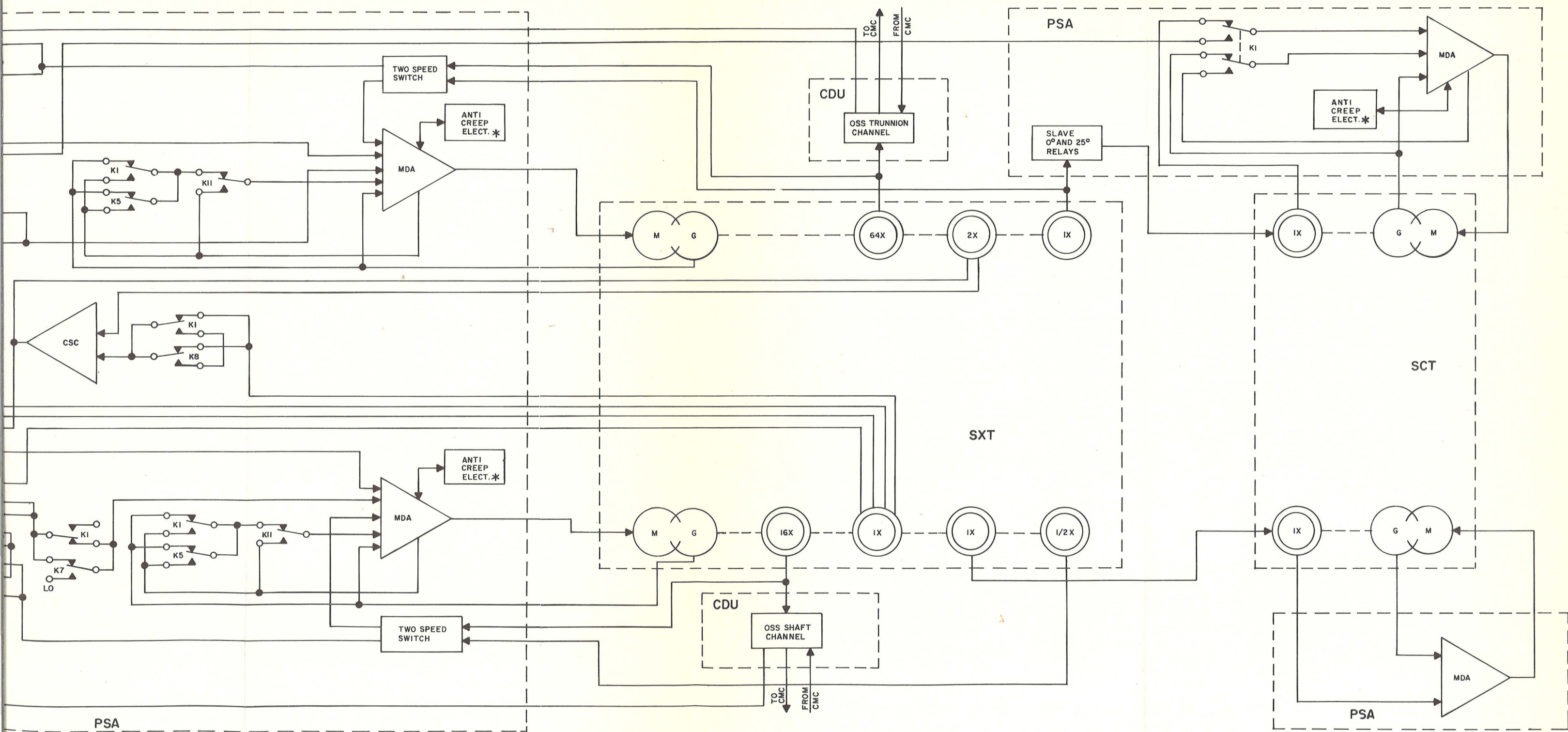
The optics hand controller controls the shaft and trunnion drive rates. The drive rates are integrated by SXT servos which position the SCT LOS. The shaft and trunnion angles can be read directly from the SCT by counters mounted in the face of the optical panel.

2-5.2 SEXTANT. The SXT is a highly accurate, dual LOS, electro-optical instrument with 28 power magnification and 1.8 degrees field of view. It is capable of sighting two celestial targets simultaneously and measuring the angle between them with 10 arc seconds accuracy. One LOS, called landmark LOS (LLOS) is fixed along the shaft axis normal to the local conical surface of the spacecraft. The LLOS is positioned by changes in spacecraft attitude. The other LOS, called star LOS (S_t LOS), has two degrees of rotational freedom about the shaft and trunnion axis. Variation about the trunnion axis is represented by movement of an indexing mirror. S_t LOS positioning is controlled by electro-mechanical integration loops consisting of servos, tachometers, and associated electronics. Control of the servos is determined by operating modes which are selected using indicator control panel switches. The S_t LOS movement is independent of the fixed LLOS. Measurements are made by first sighting into the SXT eyepiece and adjusting spacecraft attitude until the LLOS image is centered on the SXT reticle. Subsequent positioning of the S_t LOS to locate the star image coincident with the LLOS is done to satisfy requirements necessary for measurements to be taken between the two images. (Coincidence must be as near the reticle center as possible.) Positioning accuracies of the SXT trunnion and shaft axes are within 10 and 40 arc seconds, respectively.



*Removed by ECP 590, see table 3-IY for effectivity.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM



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Figure 2-18. OSS, Functional Block Diagram

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

2-5.3 SCT SERVO LOOPS. Shaft and trunnion servo loops are used to position the SCT. The shaft and trunnion servo loops function as single speed repeater servos. Trunnion servo loop operation is determined by mode switching.

2-5.3.1 SCT Shaft Servo Loop. A single speed positioning servo loop is used to rotate the SCT optics about the shaft axis. The SXT shaft 1X resolver generates an error signal proportional to the mechanical rotor displacement from electrical null. The electrical input to the SCT shaft 1X resolver is the sine-cosine output signal from the SXT shaft 1X resolver in the SXT shaft servo loop. The SCT resolver error signal is applied to the MDA summing network. Feedback from the motor-generator is applied through a feedback compensation network of the MDA summing network. The amplifier circuit generates the motor drive signals in the phase and magnitude required to drive the rotor of the SCT shaft 1X resolver to null and to position the SCT shaft at a new angle. The SCT shaft movement is thus slaved to SXT shaft movement.

2-5.3.2 SCT Trunnion Servo Loop. Rotation of the double dove prism and mount assembly about the trunnion axis is controlled by the SCT trunnion servo loop. This servo functions as a single speed, positioning servo. The SCT trunnion 1X resolver generates an error signal proportional to the mechanical rotor displacement from electrical null. The electrical input to the SCT trunnion resolver is determined by the configuration of the SLAVE, 0° and 25° relays. These relays are controlled by the position of the TELESCOPE TRUNNION switch on the indicator control panel.

When the TELESCOPE TRUNNION switch is in the 0° position, the 0° and SLAVE relays are energized. This disconnects the input sine-cosine signals from the SXT trunnion 1X resolver; applies a fixed 28 volt, 800 cps, 0 phase reference signal to the sine winding of the SCT trunnion 1X resolver; and shorts out the cosine winding to establish a zero electrical reference.

When the TELESCOPE TRUNNION switch is in the 25° position, the 25° and SLAVE relays are energized. This disconnects the input sine-cosine signals from the SXT trunnion 1X resolver; applies a fixed 28 volt, 800 cps, 0 phase reference signal to the sine winding of the SCT trunnion 1X resolver; and applies a fixed reference voltage from the secondary of a transformer to the cosine winding to establish an electrical reference which causes the SCT to move to a fixed 25° trunnion angle offset.

The error signal from the SCT trunnion 1X resolver is applied through relay K1 to the MDA summing network. The feedback from the motor-generator is also applied through relay K1 to the MDA summing network. The amplifier circuit generates the motor drive signals in the phase and magnitude required to drive the rotor of the SCT trunnion 1X resolver to a null and to position the SCT trunnion at a new angle.

2-5.4 SXT SERVO LOOPS. Shaft and trunnion loops are utilized to position the SXT. Either loop may function as a two-speed positioning servo or as an integrating servo loop. The loop function is determined by mode switching.

2-5.4.1 SXT Shaft Servo Loop. Rotation of the SXT index head about the shaft axis is controlled by the SXT shaft servo loop. This loop may function as either a precision two speed positioning servo loop or as an integrating servo loop. In manual and computer modes it is used as an integrating loop; in the zero optics mode the servo loop is set as a two speed, positioning servo.

When the ZERO switch is placed in ZERO position, the SXT shaft feedback relay (K5) and the shorting relay (K7) are energized. The output of the secondary winding of the 16X resolver (fine) and the output of the secondary winding of the SXT 1/2X resolver (coarse) is applied to the two speed switch. Relay K5 disconnects the direct tachometer feedback from the MDA input summing network and applies it through the compensation network as required for the positioning servo configuration.

Relay K7 places the rate input to the MDA summing network at zero. Consequently, the error input to the servo is taken from the two speed switch and is proportional to the SXT shaft displacement from the zero electrical reference. The zero reference for 1/2X coarse resolver is obtained from the 28 volt, 800 cps stator excitation supply. The error reference for the 16X fine resolver is more precise and is established by the 28 volt, 800 cps rotor excitation. The two speed servo loop drives the SXT gear train into accurate alignment with the zero reference.

In system modes of operation, other than zero optics and CMC, relays K5 and K7 are deenergized. Deenergizing these relays sets up the servo as an integrating loop. This places the two speed inputs to the MDA summing network at zero. Deenergizing relay K7 removes the zero input to the MDA summing network and switches in the rate error. The rate error is obtained from one of two drive rate sources depending on mode switching.

When the system is in manual resolved mode, the resolved input relay (K6) is energized. The optics hand controller drive rate signal is applied through closed contacts of the relay to the SXT shaft 1X resolver. The output of the SXT shaft 1X resolver goes to the cosecant function generator through relays K1 and K8 and to the MDA summing network through relay K7. The cosecant function generator output also goes to the 2X computing resolver in the SXT trunnion servo loop.

In the manual direct mode, all the relays are deenergized. The source of the drive rate error signal then is the optics hand controller. The signal is fed through the resolver input relay K6 and through contacts of K7 to the MDA summing network.

In the computer mode, the computer input relay (K10) is energized. This connects the CMC to the MDA summing network through the D/A converter in the CDU. Relay K7 disables manual control of the loop.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

2-5.4.2 SXT Trunnion Servo Loop. Rotation of the SXT indexing mirror about the trunnion axis is controlled by the SXT trunnion servo loop. This servo is similar to the SXT shaft servo in that mode relays select a precise two speed positioning servo loop in the zero optics mode and an integrating servo loop in the remaining modes. The same series of input relays switch the input to the MDA summing network from one of two sources: (1) the SXT trunnion hand controller during direct manual mode; (2) the resolved drive rate from the shaft 1X resolver during resolved manual mode. All relays are energized as explained previously.

SXT trunnion and shaft angles are obtained by interpreting displacement of the SXT indexing mirror about both axes. Two pancake resolvers, mechanically linked with the SXT indexing mirror and SXT head assembly, sense and transmit analog equivalents of SXT mirror displacements about trunnion and shaft axes.

2-5.5 MODES OF OPERATION. The OSS incorporates multiple modes of operation to facilitate the taking of precise measurements within short time increments. The specific mode selected effectively adapts the system to fit the needs of the astronaut under varying situations prior to sightings. The OSS operates in one of three prime modes and can be controlled in either of two control modes. Each is selected by switches on the indicator control panel. The modes are:

- (1) Zero Optics
- (2) Manual
 - (a) 0°
 - (b) 25°
 - (c) Slave to SXT
 - (d) Direct
 - (e) Resolved
- (3) Computer

2-5.5.1 Zero Optics. In zero optics mode, relays convert the SXT trunnion and shaft loops into two speed null sensing servos. The positioning servos then drive the SXT optics into alignment with zero reference inputs. This mode of operation is normally established as a part of a preliminary procedure for taking optical sightings.

The zero optics mode can be selected either manually or by the computer after the computer control mode of operation has been selected. The computer selects zero optics by energizing a relay which in turn energizes the two-speed switch. The computer is notified of the zero optics mode by a signal when the mode control switch is in the zero position.

Entering the zero optics mode causes the bank of SXT input relays to disconnect the SXT loops from all data source drive rate inputs. This switches in zero alignment reference inputs, and serves to mechanize each loop as a positioning servo. (See figure 2-19.)

In the SXT trunnion loop, the 1X and 64X resolver transmitter outputs are switched back to the loop MDA by a two-speed switch. These resolver output signals terminate at zero optics relay contacts whenever the loop functions as an integrating device with rate inputs. In zero optics mode, the relay contacts close and apply the output signals to a two-speed, coarse-fine switch. The resolvers are mechanized as control transformer error detectors for the trunnion integrating loop circuit and generate error signals proportional to the displacement of the mechanical shaft from the electrical zero reference. These error signals are used to drive the mechanical shaft until it is precisely aligned to the zero reference. The zero optics relay also reconnects the tachometer generator from direct to compensated feedback so that the servo characteristics assume the characteristics of a positioning servo.

The SXT trunnion loop accuracy is determined by the 64X resolver. However, since the 64X resolver passes through 64 nulls in 180 degrees of trunnion rotation, the 1X resolver output overrides the 64X resolver output and drives the SXT trunnion near zero. When the SXT trunnion angle is within 2.8 degrees of zero (5.625 degrees StLOS), the 64X resolver output is switched into the MDA to drive the SXT trunnion to zero. The SXT trunnion CDU read counter will then be cleared by the CMC CDU zero discrete.

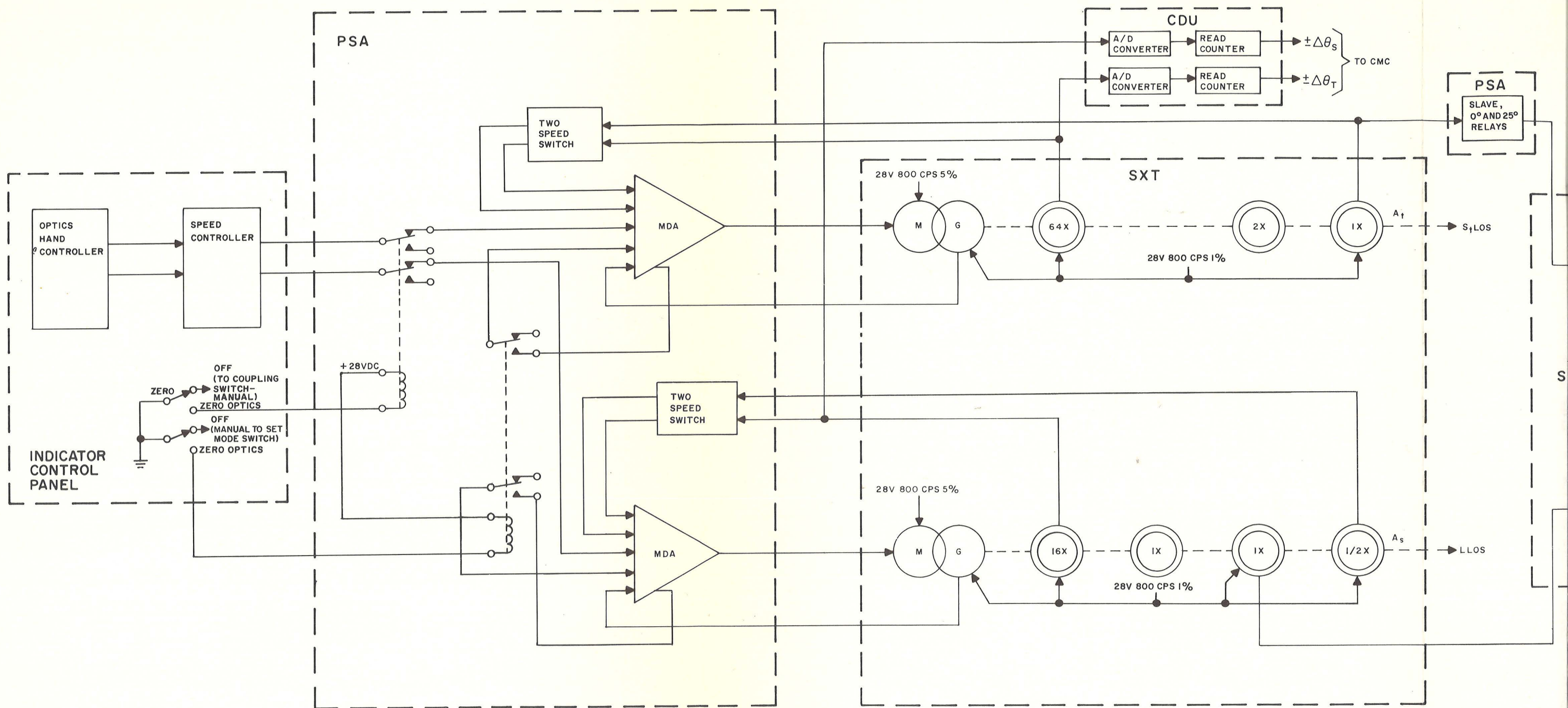
The SXT shaft integrating loop is mechanized identically to that of the trunnion loop. In the SXT shaft loop, the 1/2X and 16X resolvers serve as error detectors for the two-speed, shaft positioning servo.

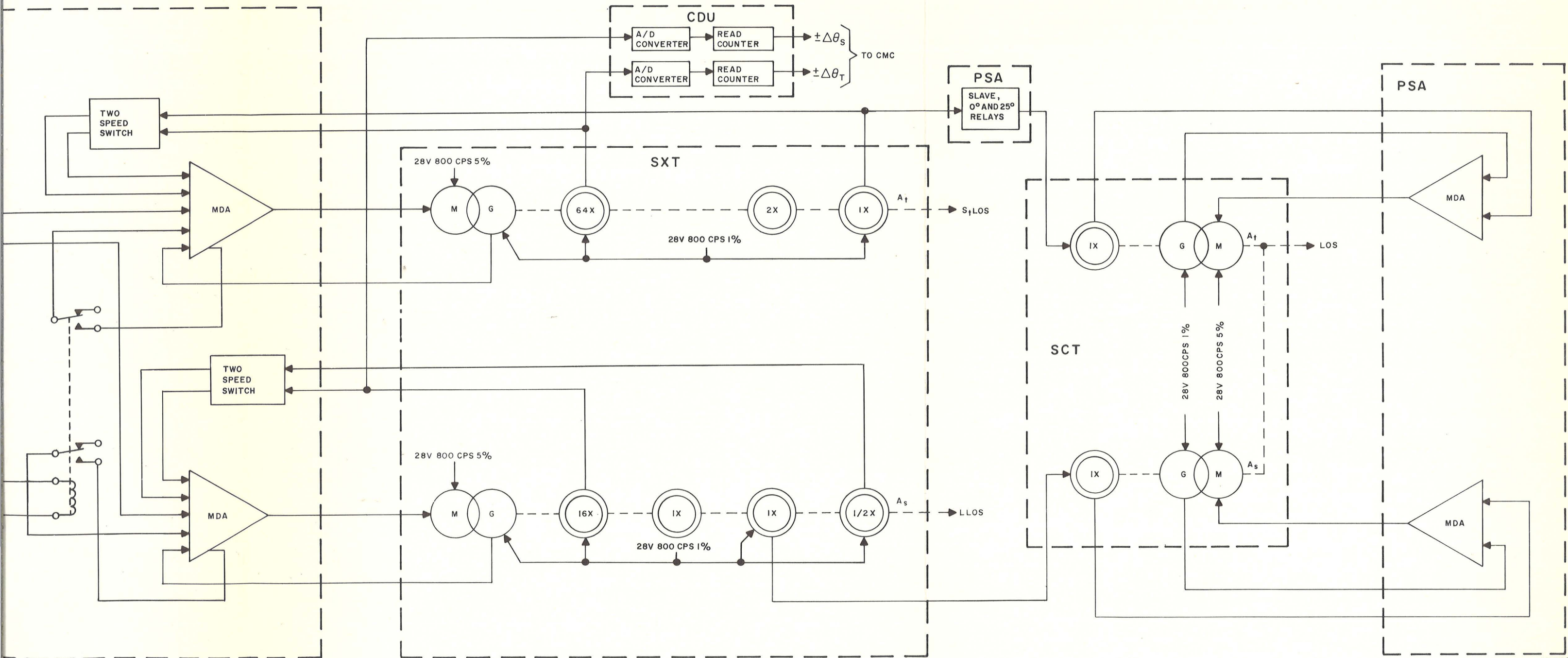
2-5.5.2 Manual Modes. In manual modes of operation, the direction and rate of image motion viewed on the SCT and SXT eyepieces are manually controlled. The optics hand controller and speed selector on the indicator control panel are used for manually acquiring the landmark and star targets. Displacement of the optics hand controller generates SXT trunnion and shaft angular drives (\dot{A}_t and \dot{A}_s).

The manual mode provides these three modes of operation associated with the SCT: 0°, 25°, and Slave to SXT.

2-5.5.2.1 0° Manual Mode. In this mode, 28 volt, 800 cps power is applied directly to the SCT trunnion 1X resolver cosine winding limiting the SCT to shaft motion only.

2-5.5.2.2 25° Manual Mode. In this mode, 28 volt, 800 cps power is applied through the 25° offset transformer to the sine winding of the SCT 1X resolver. The SCT trunnion is displaced 25 degrees from the zero position and is capable of scanning a 110 degree cone by rotation of the SCT shaft.





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Figure 2-19. Zero Optics and Manual Direct Modes

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

2-5.5.2.3 Slave to SXT Manual Mode. In this mode, the SCT trunnion is slaved directly to the SXT trunnion and the following two modes of operation associated with the SXT are provided: manual direct mode and manual resolved mode.

2-5.5.2.4 Manual Direct Mode. The manual direct mode permits image movement in a coordinate system. Figure 2-20 illustrates the direction of image movement as viewed in the SCT and SXT eyepieces using the optics hand controller. Characteristics of the typical image presentations shown in figure 2-20 are:

(1) Position 1. A landmark target is selected by manipulation of spacecraft attitude controls and centered on the SCT reticle. The star target at this point is not in the SCT field of view.

(2) Position 2. By selecting the offset 25° mode, the landmark target is displaced to the edge of the SCT field of view. The selected star target is now positioned in the SCT field of view.

(3) Position 3. The SCT reticle is displaced in shaft to align the star image on the SCT reticle reference line. The included angle between landmark and star image, as measured on the SCT reticle reference line, must be less than 50° to maintain images in the SXT reticle. The SXT trunnion is displaced an amount equal to the included angle.

(4) Position 4. The SXT trunnion angle is driven to the included angle to acquire the star target. When viewed in the SXT eyepiece, both landmark and star images now appear in the field of view. The star image may appear to be misaligned with respect to the shaft axis reticle reference.

(5) Position 5. Fine alignment of the star image with the shaft reticle reference is accomplished by positioning the optics hand controller.

(6) Position 6. The star image is displaced in trunnion angle toward the center of the landmark target aligned on the center of the reticle.

(7) Position 7. Super-imposition of the star target on the landmark target is accomplished and the MARK button depressed.

In manual direct mode, the SXT trunnion and shaft servos are mechanized as rate integrating servo loops. The trunnion and shaft angular drive rate signals (A_t and A_s) are fed through switching relays directly into the MDA of their respective servos.

The trunnion drive rate signal (A_t) causes the trunnion servo, located in the SXT optics, to drive the SXT indexing mirror around the trunnion axis in a direction and at a rate proportional to the direction and displacement of the optics hand controller. Similarly, the shaft drive rate signal (A_s) causes the shaft to rotate the entire SXT optics around the shaft axis.

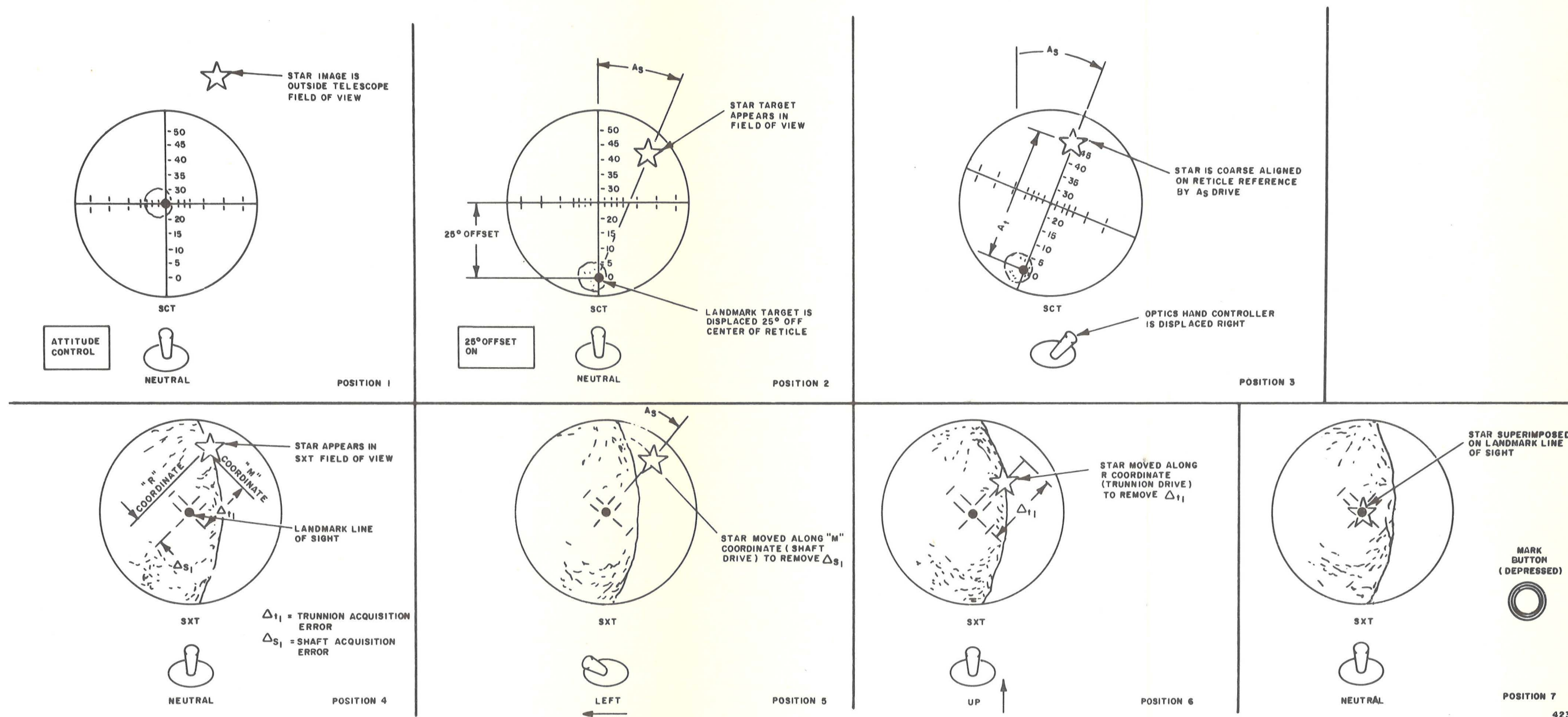


Figure 2-20. Image Presentations, Manual Direct Mode

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

The sin and cos error signals from the SXT trunnion 64X resolver and the SXT shaft 16X resolver are applied through the CDU A/D converter to the read counters. The $\pm\Delta\theta_T$ and $\pm\Delta\theta_S$ pulses from the read counters are sent to the CMC representing the amount of SXT trunnion and shaft angle rotation. The CMC can use this data to display the SXT angles on the DSKY and to determine spacecraft position and trajectory.

During the direct manual mode, the anticreep electronics removes B+ from the MDA's when the optics hand controller output (either trunnion or shaft) drops below 500 mv. This prevents the optics from drifting when the optics hand controller is at null. The anticreep electronics has been removed by ECP 590. See table 3-IY for effectivity.

2-5.5.2.5 Manual Resolved Mode. The manual resolved mode permits independent movement of the image coordinates relative to a rectangular system. Figure 2-21 illustrates the direction of image movement in the SXT eyepiece using the optics hand controller. Characteristics of the typical SXT image presentations shown in figure 2-21 are:

(1) Position 1. With the landmark and star images coarse aligned by the SCT, both targets are presented in the SXT field of view. At this point, the image is not aligned with the SXT reticle.

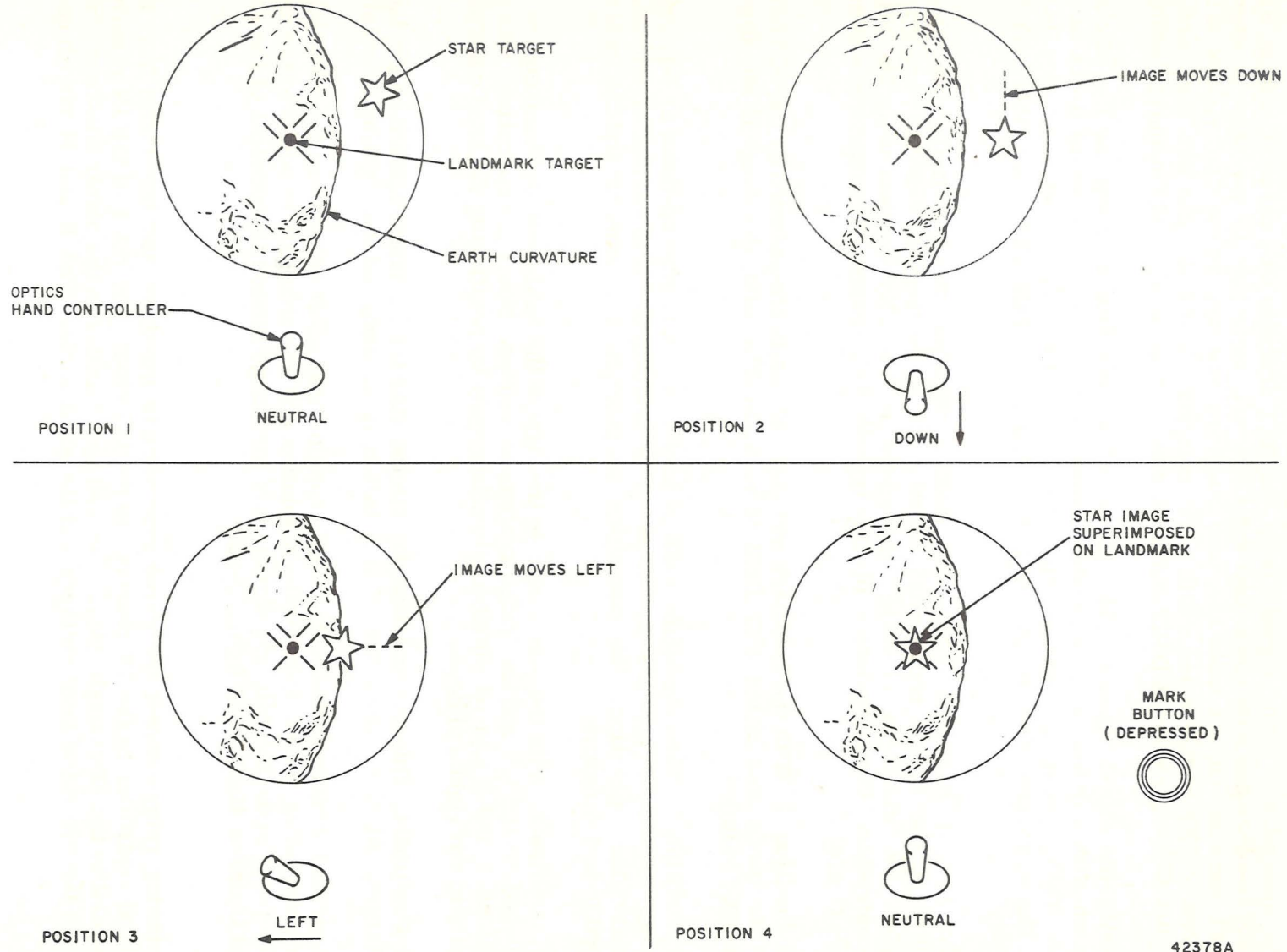
(2) Position 2. When the optics hand controller is positioned up-down or left-right, the image movement appears to follow or is coordinated with controller motion. In this particular operation, the controller is positioned downward and the image is displaced in that direction.

(3) Position 3. With the star target to the right of the landmark, the hand controller is displaced to the left and the image movement follows. Note: the operations covered in positions 2 and 3 can be obtained simultaneously by positioning the hand controller diagonally to the right and down.

(4) Position 4. This presentation depicts the star target superimposed on the landmark target. At this time, the MARK button is pressed and the sighting completed.

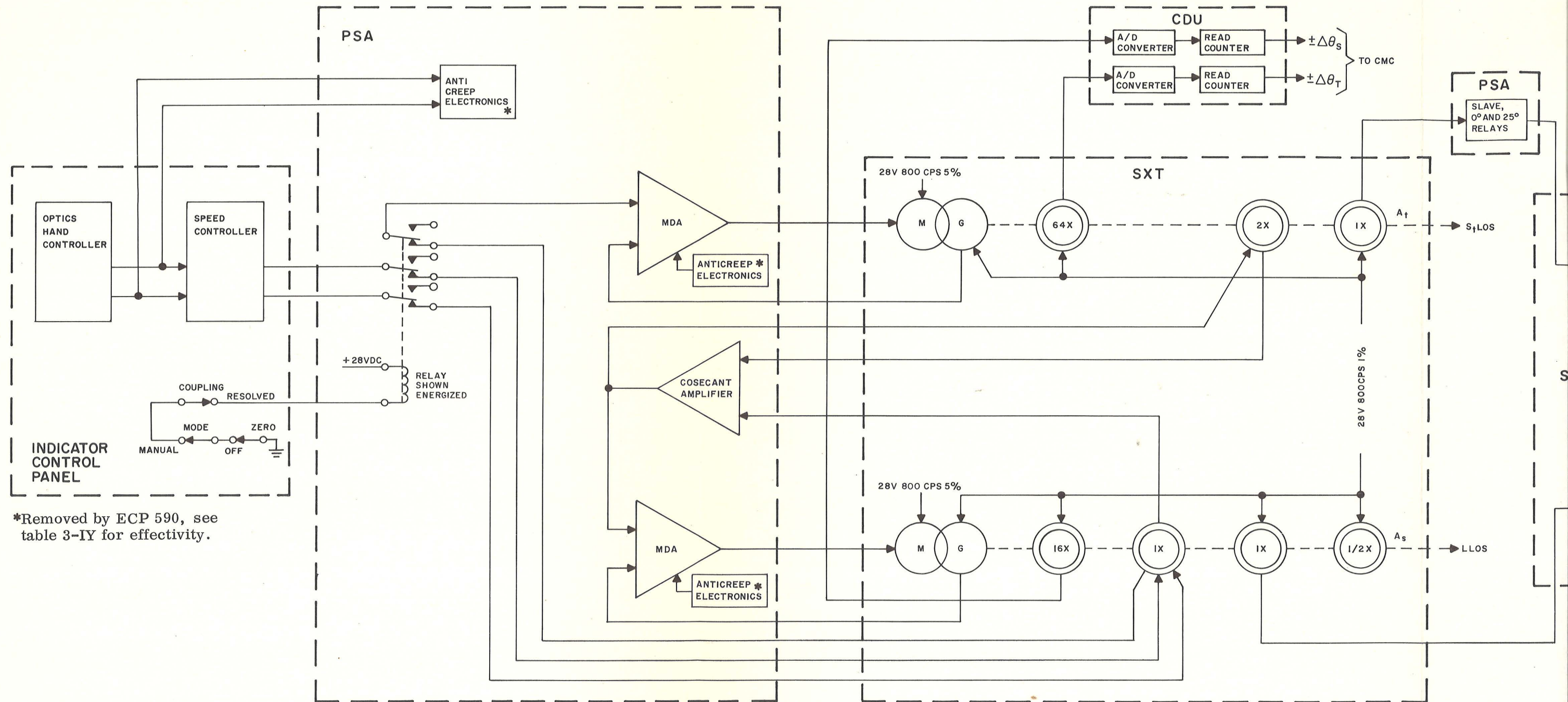
In manual resolved mode, the SXT trunnion and shaft servos are mechanized as rate or integrating loops. Drive rate resolution and variable gain is added to the shaft servo loops to make the rate and direction of the image motion independent of the shaft axis and trunnion angle. (See figure 2-22.)

Switching relays direct the drive rate signals (up-down and left-right) from the optics hand controller to the 1X resolver in the SXT shaft. The rotor of the 1X resolver is positioned to the shaft angle (A_S). The voltages induced in the stator windings are a function of the two rotor input voltages and the shaft angle. The R and M coordinates



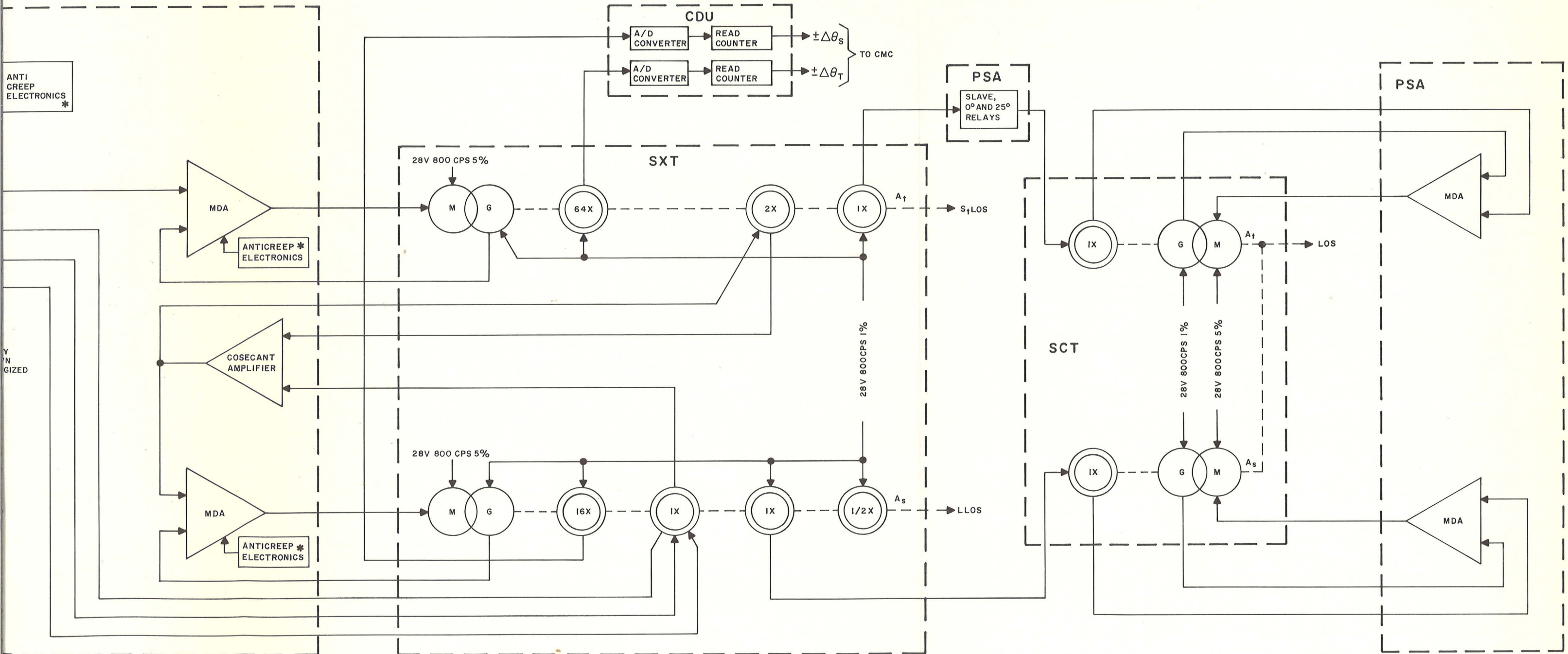
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Figure 2-21. SXT Image Presentations, Manual Resolved Mode



*Removed by ECP 590, see table 3-IY for effectivity.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM



16146D

Figure 2-22. Manual Resolved Mode

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

drive rate signals are resolved by the 1X resolver into shaft and trunnion signals. The trunnion signal is fed directly to the MDA while the shaft signal is fed through the cosecant amplifier to the MDA. This servo causes the SXT indexing mirror to rotate around the trunnion axis at a rate and direction proportional to the drive rate signal causing the image to move up or down for an up-down drive rate signal.

The left or right drive rate signal from the 1X resolver is fed to the SXT shaft servo through a cosecant amplifier which is connected in the input circuit by switching relays. This amplifier functions as a variable gain device between trunnion angles of 10 to 60 degrees. The amplifier employs a 2X computing resolver in the SXT trunnion servo as a minor loop. The effect of this cosecant generator is to make the rate of image motion (angular velocity) independent of changes in trunnion angle, by decreasing the shaft speed as trunnion angle increases. The reduction in shaft speed enables a constant rate of trunnion peripheral velocity to be achieved regardless of trunnion angle. This insures that the optics LOS does not sweep past the target before recognition.

One input to the summing network of the cosecant amplifier is the drive rate along the Y axis, which is obtained after resolution by the 1X resolver in the shaft servo. The motion along this axis increases at a rate proportional to the sine of the trunnion angle (A_t). To compensate for this effect, an inverse sine factor (the cosecant) is introduced. A computing resolver in the SXT trunnion gear train is employed to yield the function $0.5C \sin A_t$ to compensate for the gear train ratio. The controlled electrical variable (C) drives the SXT shaft integrating loop so that its angular velocity is independent of the trunnion angle. In this way, a uniform sweep rate along the Y axis is achieved throughout the range of trunnion angles. The resolution of the drive rates onto the Y axis is used as a direct input to the shaft integrating servo loop.

During the manual resolved mode, the anticreep electronics removes B+ from the MDA's when the optics hand controller output (both trunnion and shaft) drops below 500 mv. This prevents the optics from drifting when the optics hand controller is at null. The anticreep electronics has been removed by ECP 590. See table 3-IY for effectivity.

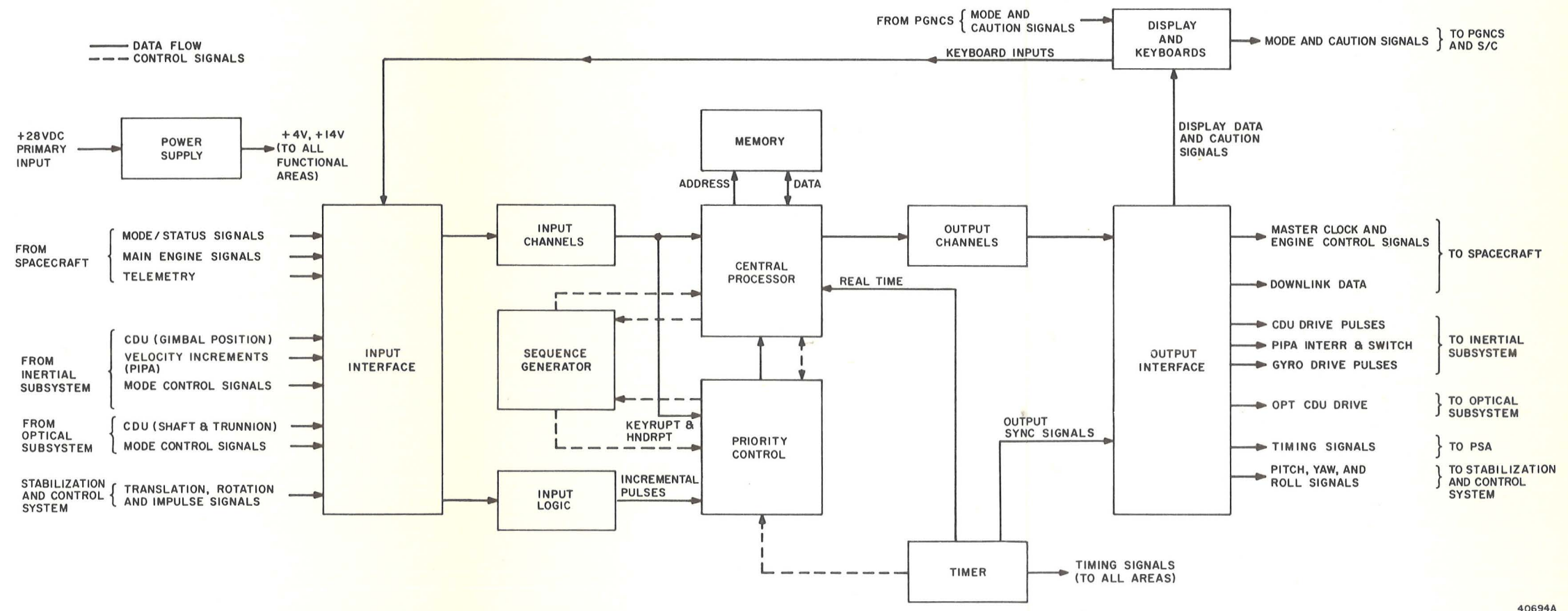
2-5.5.3 Computer Mode. The computer mode is initiated from the indicator control panel by the astronaut. The computer then controls and drives the OSS loops. The computer can initiate zero optics, and computer driving. During the computer mode, the zero optics mode functions the same as in the manual mode. During computer driving mode, the CMC issues the enable error counter discrete which enables the error counters to receive $\pm\Delta\theta_{CT}$ and $\pm\Delta\theta_{CS}$ pulses from the CMC. These pulses represent the desired angle of the optics as computed by the computer. The output of the error counter is fed to the D/A converters where it is converted to an ac signal and used to position the SXT loops through the MDA. As the SXT is driven, the CDU read counters receive the trunnion and shaft resolver error signals and register the SXT angles. A feedback circuit from the read counter to the error counter reduces the error counter contents as the read counter increases and nulls the error counter and SXT stops driving. The CMC receives the $\pm\Delta\theta_T$ and $\pm\Delta\theta_S$ pulses from the read counter and determines when the optics are at the desired angles.

2-6 COMPUTER SUBSYSTEM

The CSS is the control and processing center of the PGNCS. It consists of the CMC, a main panel DSKY and a navigation panel DSKY. The CSS processes data and issues discrete outputs and control pulses both to the PGNCS and to the other Apollo spacecraft systems. The CMC is a parallel digital control computer with many features of a general purpose computer. As a control computer, the CMC aligns the IMU, positions the optics (SXT), and issues control commands to subsystems of the spacecraft. As a general purpose computer, the CMC solves the guidance and navigation equations required for the round trip to the moon. In addition, the CMC monitors the operation of the spacecraft, including the CSS.

The main functions of the CMC (see figure 2-23) are implemented through the execution of the programs stored in memory. Programs are written in a machine language called basic instructions. A basic instruction contains an operation (order) code and a relevant address. The order code defines the data flow within the CMC, and the relevant address selects the data that is to be used for computations. The sequence generator controls data flow. The order code of each instruction is entered into the sequence generator, and the sequence generator produces a different sequence of control pulses for each instruction. Each instruction is followed by another instruction. In order to specify the sequence in which consecutive instructions are to be executed, the instructions are normally stored in successive memory locations. By adding the quantity one to the address of an instruction being executed, the address of the instruction to be executed next is derived. Execution of an instruction is complete when the order code of the next instruction is transferred to the sequence generator and the relevant address is in the central processor.

The central processor consists of several flip-flop registers. It performs arithmetic operations and data manipulations on information accepted from memory, the input channels, and priority control. Arithmetic operations are performed using the ONE's complement number system. Values of 14 bits, excluding sign, (up to 28 bits during double precision operations) are processed with an additional bit produced for overflow or underflow. All operations within the central processor are performed under control of pulses generated by the sequence generator (indicated by dashed lines in figure 2-23). In addition, all words read out of memory are checked for correct parity, and a parity bit is generated for all words written into memory within the central processor. Odd parity is used; therefore, all words stored in memory contain an odd number of ONE's including the parity bit. The central processor also supplies data and control signals through the output channels and interface to the various spacecraft subsystems.



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Figure 2-23. Computer Subsystem Block Diagram

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

The CMC, part number 2003100, has provision for ten program interrupts. These ten interrupts are: T6 RUPT, T5 RUPT, T3 RUPT, T4 RUPT, KEYRUPT 1, KEYRUPT 2, UPRUPT, DOWNRUPT, RADAR, and HAND CNTRL RUPT. The T6 RUPT through T4 RUPT programs are initiated by the CMC. The DOWNRUPT program is initiated at the completion of every parallel-to-serial conversion for downlink operation. The remaining priority programs are initiated by external inputs to the CMC. The KEYRUPT programs are initiated when a DSKY pushbutton is depressed or when priority control receives a MARK signal (discrete bit) from the OSS to indicate a sighting. The UPRUPT program is initiated when a complete UPLINK word is received. The HAND CNTRL RUPT program is initiated as soon as the hand controller is moved out of detent by the astronaut.

The CMC, part number 2003200, has ten program interrupt conditions. These ten interrupts are T6 RUPT, T5 RUPT, T3 RUPT, T4 RUPT, KYRPT 1, KYRPT 2, MKRPT, UPRUPT, DLKRPT, and HNRPT. The T6 RUPT through T4 RUPT conditions are internal interrupts initiated by the CMC. The KYRPT conditions are initiated when a DSKY pushbutton is depressed. A MARK signal (discrete bit) from the OSS, indicating a sighting, initiates MKRPT. This interrupt shares the same level of priority as KYRPT 2. UPRUPT indicates the completion of an uplink word. HNRPT is initiated as soon as the hand controller is moved out of detent by the astronaut.

Before a priority program can be executed, the current program must be interrupted: however, certain information about the current program must be preserved. This includes the program counter and any intermediate results contained in the central processor. Priority control produces an interrupt request signal, which is sent to the sequence generator. This signal, acting as an order code, causes the execution of an instruction that transfers the current contents of the program counter and any intermediate results to memory. In addition, the control pulses transfer the priority program address in priority control to the central processor, and then to memory via the write lines. As a result, the first basic instruction word of the priority program is entered into the central processor from memory, and execution of the priority program is begun. The last instruction of each priority program restores the CMC to normal operation, provided no other interrupt request is present. This instruction transfers the previous program counter and intermediate results from their storage locations in memory back to the central processor.

Certain data pertaining to the flight of the spacecraft is used to solve the guidance and navigation problems required for the round trip to the moon. This data, which includes real time, acceleration, and IMU gimbal angles, is stored in memory locations, called counters. The counters are updated as soon as new data becomes available. An incrementing process is implemented, by priority control between the execution of basic instructions, and changes the contents of the counters. Data inputs to priority control are called incremental pulses. Each incremental pulse produces a counter address and a priority request. The priority request signal is sent to the sequence generator, where it functions as an order code. The control pulses produced by the sequence generator transfer the counter address to memory through the write lines of the central processor. In addition, the control pulses enter into the central processor the contents of the addressed counter to be incremented.

Real time plays a major role in solving guidance and navigation problems. Real time is maintained within the CMC in the main time counter of memory. The main time counter provides a 745.65 hour (approximately 31 days) clock. Incremental pulses are produced in the timer and sent to priority control to increment the main time counter. The Apollo mission requires that the CMC clock be synchronized with the KSC clock. The CMC time is transmitted once every second by downlink operation for comparison with the KSC clock.

Incremental transmissions occur in the form of pulse bursts from the output channels to the CDU, the gyros, the RCS of the spacecraft and to the radar. The number of pulses and the time at which they occur are controlled by program. Discrete outputs originate also in the output channels under program control. These outputs are sent to the DSKY's and various other subsystems. Continuous pulse trains originate in the timing output logic for synchronization of other systems.

The uplink word from the spacecraft telemetry system is supplied as an incremental pulse input to priority control. As this word is received, priority control produces the address of the uplink counter in memory and requests the sequence generator to execute the instructions which perform the serial-to-parallel conversion of the input word. When the serial-to-parallel conversion is completed, the parallel word is transferred to a storage location in memory by the uplink priority program. The uplink program also retains the parallel word for subsequent downlink transmission. Another program converts the parallel word to a coded display format and transfers the display information to the DSKY's.

The downlink operation of the CMC is asynchronous with respect to the spacecraft telemetry system. The telemetry system supplies all the timing signals necessary for the downlink operation. These signals include start, end, and bit sync pulses.

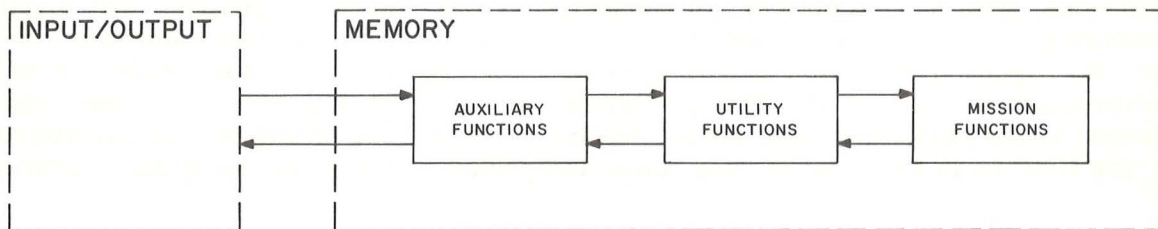
The CSS contains two identical DSKY's. With either keyboard, the astronaut can load information into the CMC, retrieve and display information contained in the CMC, and initiate any program stored in memory. A keycode is assigned to each keyboard pushbutton. When a keyboard pushbutton on either DSKY is depressed, the keycode is produced and sent to an input channel. A signal is also sent to priority control, where it produces both the address of a priority program stored in memory and a priority request signal, which is sent to the sequence generator. This results in an order code and initiates an instruction for interrupting the program in progress and executing the KEYRUPT priority program stored in memory. A function of this program is to transfer the keycode, temporarily stored in an input channel, to the central processor, where it is decoded and processed. A number of keycodes are required to specify an address, or a data word. The program initiated by a keycode also converts the information from the DSKY keyboard to a coded display format. The coded display infor-

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

mation is transferred by another program to an output channel, and sent to the display portion of each DSKY. The display notifies the astronaut that the keycode was received, decoded, and processed properly by the CMC.

2-6.1 PROGRAMS. A CMC program performs such functions as solving guidance and navigation problems, testing the operation of the PGNCS and monitoring the operation of the spacecraft. Such a program consists of a group of program sections that are classified according to the functions they perform. These functions are defined as mission functions, auxiliary functions, and utility functions. (See figure 2-24.)

2-6.1.1 Mission Functions. Mission functions are performed by program sections that implement operations concerned with the major objectives of the Apollo mission. These operations include erecting the IMU stable member and aligning it to a desired heading while the spacecraft is located on the ground. The stable member is also aligned each time the ISS is re-energized during a flight and after prolonged use. In addition, the mission functions include computation of spacecraft position and velocity during coasting periods of the flight by solution of second-order differential equations which describe the motions of a body subject to the forces of gravity.



40383

Figure 2-24. Program Organization

2-6.1.2 Auxiliary Functions. Auxiliary functions are executed at the occurrence of certain events, requests, or commands. These functions are performed by program sections that provide a link between CMC and other elements of the PGNCS. This link enables the CMC to process signals from various devices and to send commands for control and display purposes. In addition, the auxiliary functions implement many and varied operations within the CMC in support of the mission functions.

2-6.1.3 Utility Functions. Utility functions are performed by program sections that coordinate and synchronize CMC activity to guarantee orderly and timely execution of required operations. These functions control the operation of the mission functions and schedule CMC operations on either a priority or a real-time basis. The utility functions also translate interpretive language to basic machine language which allows complex mathematical operations such as matrix multiplication, vector addition, and dot product computations to be performed within the framework of compact routines. In addition, the utility functions save the contents of registers A and Q during an interrupt condition and enable data retrieval and control transfer between isolated banks in the fixed-switchable portion of fixed memory.

2-6.2 MACHINE INSTRUCTIONS. The CMC has three classes of machine instructions - regular, involuntary, and peripheral (table 2-I). Regular instructions are programmed and are executed in whatever sequence they have been stored in memory. Involuntary instructions (with one exception) are not programmable and have priority over regular instructions. One involuntary instruction may be programmed to test computer operations. No regular instruction can be executed when the computer forces the execution of an involuntary instruction. The peripheral instructions are used when the CMC is connected to the peripheral equipment. During the execution of any peripheral instruction, the CMC is in the monitor stop mode and cannot perform any program operation.

Table 2-I. Instruction Classes

Class	Type	Control
Regular	Basic Extracode Channel Special	Program
Involuntary	Interrupt Counter	Priority
Peripheral	Keyboard Tape	Operator

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

2-6.2.1 Regular Instructions. Four types of instructions comprise the regular instruction class. They are the basic, channel, extracode, and special instructions. Basic instructions are the most frequently used instructions. The instruction words stored in memory are called basic instruction words. They contain an order code field and an address field. Special instructions have predefined addresses and order codes whereas basic instructions have only predefined order codes. The special instructions are used to control certain operations in the computer. For example, one special instruction is used to switch the computer to the extend mode of operation. This mode extends the length of the order code field and converts basic instruction words to channel or extracode instruction words. Channel instructions can only be used with input/output channel addresses. Extra code instructions perform the more complex and less frequently used arithmetic operations.

Regular instructions can also be functionally subdivided into the following:

- (1) Sequence changing.
- (2) Fetching and storing.
- (3) Modifying.
- (4) Arithmetic and logic.
- (5) Input/output.
- (6) Editing.

The sequence changing instructions alter the sequence in which the instructions stored in memory are executed. One group, called transfer control instructions, changes the program path as defined by the programmer. The other group, called decision making instructions, branches to alternate program paths in response to predefined conditions.

The fetching and storing instructions move data, without alteration, from one location to another. One group called copy instructions provides a nondestructive transfer of data from memory to the central processor. Another group called exchange instructions transposes data between memory and the central processor. One instruction provides a nondestructive transfer of data from the central processor to memory.

The modifying instructions alter the next instruction to be executed by changing the contents of the order code field, address field, or both.

The arithmetic and logic instructions perform numerical computations. One group called the basic arithmetic instructions performs addition, subtraction, multiplication, and division in the ONE's complement number system. Another group called the add and store instructions performs single or double precision addition, and transfers the

resultant from the central processor to memory. The incrementing instructions increment a signed quantity, increment its absolute value, or diminish its absolute value by one. One instruction performs subtraction in the TWO's complement number system for angular data and one instruction performs the Boolean AND operation.

The input/output or channel instructions link the interface circuits to the central processor. One group called read instructions transfers the total or partial contents of any channel (register) location to the central processor either directly or accompanied by the Boolean AND, OR, or EXCLUSIVE OR operation. Another group of instructions transfers all new or partially new information to any channel location in the same manner.

The editing or special instructions are address-dependent and control the operation of the program. One special instruction, as mentioned previously, controls the extend mode of operation. Other instructions prevent a program from being interrupted and shift and cycle data to the left or right.

2-6.2.2 Involuntary Instructions. The involuntary instruction class contains two types of instructions - interrupt and counter. The interrupt instructions use the basic instruction word format just as the regular instructions do. However, the interrupt instructions are not entirely programmable. The contents of the order code field and the address field are supplied by computer logic rather than the program. The counter instructions have no instruction word format. Signals which function as a decoded order code specify the counter instruction to be executed and the computer logic supplies the address. The address for these instructions is limited to one of 29 counter locations in memory.

There are two interrupt instructions. One instruction initializes the computer when power is first applied and when certain program traps occur. The other interrupt instruction is executed at regular intervals to indicate time, receipt of new telemetry or keyboard data, or transmission of data by the computer. This interrupt instruction may be programmed to test the computer.

There are several counter instructions. Two instructions will either increment or decrement by one the content of a counter location using the ONE's complement number system. Two other instructions perform the same function using the TWO's complement number system. Certain counter instructions control output rate signals and convert serial telemetry data to parallel computer data.

2-6.2.3 Peripheral Instructions. There are two types of peripheral instructions. One type deals with memory locations and the other type deals with channel locations. The peripheral instructions are not used when the computer is in the spacecraft. They are used when the computer is connected to peripheral equipment during subsystem and preinstallation system testing. The peripheral instructions are not programmable and are executed when all computer program operations have been forcibly stopped. These

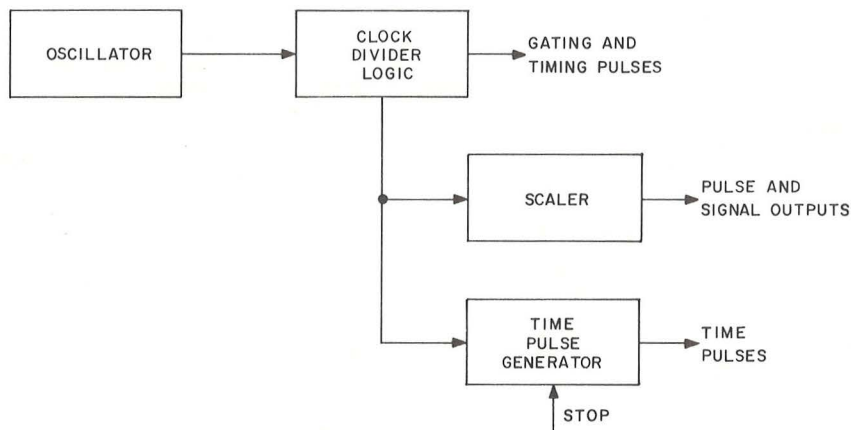
BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

instructions are used to read and load any memory or channel location, and to start the computer program at any specified address. The peripheral instructions and counter instructions are processed identically.

2-6.3 TIMER. The timer generates the timing signals required for operation of the CMC and is the primary source of timing signals for all spacecraft systems.

The timer is divided into the areas indicated in figure 2-25. The master clock frequency is generated by an oscillator and is applied to the clock divider logic. The divider logic divides the master clock input into gating and timing pulses at the basic clock rate of the computer. Several outputs are available from the scaler (which further divides the divider logic output into output pulses and signals for gating) to generate rate signal outputs and for the accumulation of time. Outputs from the divider logic also drive the time pulse generator which produces a recurring set of time pulses. This set of time pulses defines a specific interval (memory cycle time) in which access to memory and word flow take place within the computer.

The start-stop logic senses the status of the power supplies and specific alarm conditions in the computer, and generates a stop signal which is applied to the time pulse generator to inhibit word flow. Simultaneously, a fresh start signal is generated which is applied to all functional areas in the computer. The start-stop logic and subsequent word flow in the computer can also be controlled by inputs from the Computer Test Set (CTS) during pre-installation systems and subsystem tests.



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Figure 2-25. Timer Block Diagram

2-6.4 SEQUENCE GENERATOR. The sequence generator carries out the instructions stored in memory. The sequence generator processes instruction codes and produces control pulses which regulate the data flow of the computer. The control pulses are responsible for performing the operations assigned to each instruction in conjunction with the various registers in the central processor and the data stored in memory.

The sequence generator (figure 2-26) consists of the order code processor, command generator, and control pulse generator. The sequence generator receives order code signals from the central processor and priority control. These signals are coded by the order code processor and supplied to the command generator. The special purpose control pulses are used for gating the order code signals into the sequence generator at the end of each instruction.

The command generator receives instruction signals from priority control and peripheral equipment and coded signals from the order code processor. The command generator determines the input signals and produces instruction commands which are supplied to the control pulse generator.

The control pulse generator receives twelve time pulses from the timer. These pulses occur in cycles and are used for producing control pulses in conjunction with the instruction commands. There are five types of control pulses: read, write, test, direct exchange, and special purpose. Information in the central processor is transferred from one register to another by the read, write, and direct exchange control pulses. The special purpose control pulses regulate the operation of the order code

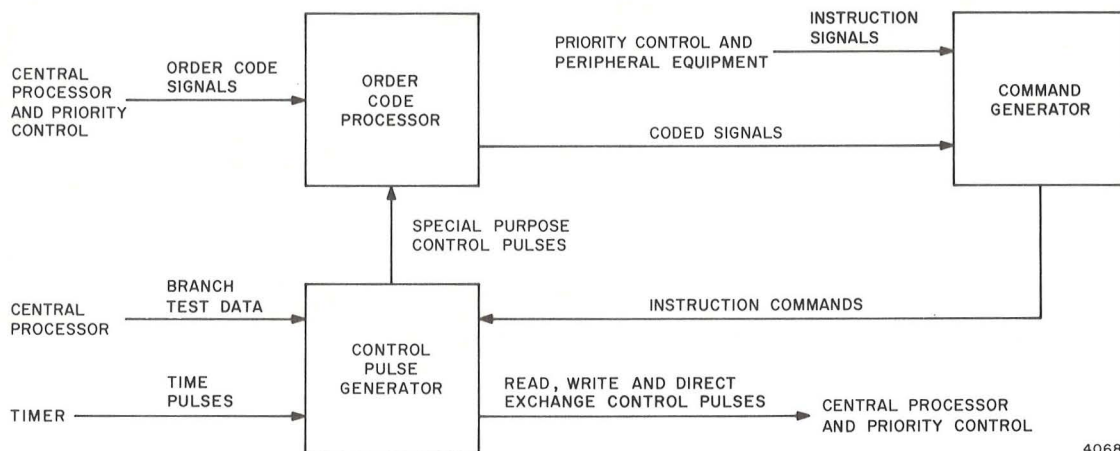


Figure 2-26. Sequence Generator Block Diagram

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

processor. The test control pulses are used within the control pulse generator. The branch test data from the central processor changes the control pulse sequence of various instructions.

2-6.5 CENTRAL PROCESSOR. The central processor, figure 2-27, consists of the flip-flop registers, the write, clear, and read control logic, write amplifiers, memory buffer register, memory address register and decoder, and the parity logic. All data and arithmetic manipulations within the CMC take place in the central processor.

Primarily, the central processor performs operations indicated by the basic instructions of program stored in memory. Communication within the central processor is accomplished through the write amplifiers. Data flows from memory to the flip-flop registers or vice-versa, between individual flip-flop registers, or into the central processor from external sources, In all instances, data is placed on the write lines and routed to a specific register or to another functional area under control of the write,

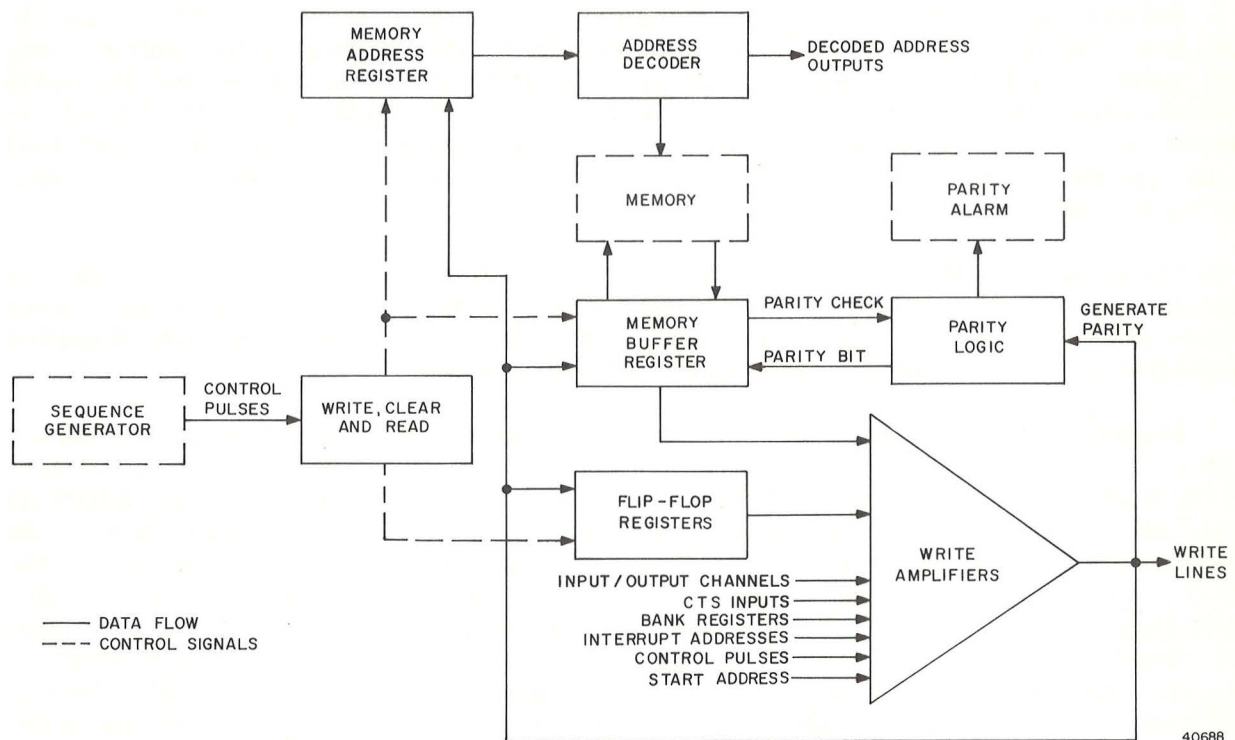


Figure 2-27. Central Processor Block Diagram

clear, and read logic. This logic section accepts control pulses from the sequence generator and generates signals to read the content of a register onto the write lines, and write this content into another register of the central processor or to another functional area of the CMC. The particular memory location is specified by the content of the memory address register. The address is fed from the write lines into this register, the output of which is decoded by the address decoder logic. Data is subsequently transferred from memory to the memory buffer register. The decoded address outputs are also used as gating functions within the CMC.

The memory buffer register buffers all information read out or written into memory. During readout, parity is checked by the parity logic and an alarm is generated in case of incorrect parity. During write-in, the parity logic generates a parity bit for information being written into memory. The flip-flop registers are used to accomplish the data manipulations and arithmetic operations. Each register is 16 bits or one computer word in length. Data flows into and out of each register as dictated by control pulses associated with each register. The control pulses are generated by the write, clear, and read control logic.

External inputs through the write amplifiers include the content of both the erasable and fixed memory bank registers, all interrupt addresses from priority control, control pulses which are associated with specific arithmetic operations, and the start address for an initial start condition. Information from the input and output channels is placed on the write lines and routed to specific destinations either within or external to the central processor. The CTS inputs allow a word to be placed on the write lines during system and subsystem tests.

2-6.6 PRIORITY CONTROL. Priority control is related to the sequence generator in that it controls all involuntary or priority instructions. Priority control processes input-output information and issues order code and instruction signals to the sequence generator and twelve-bit addresses to the central processor.

Priority control (figure 2-28) consists of the start, interrupt, and counter instruction control circuits. The start instruction control initializes the computer if the program works itself into a trap, if a transient power failure occurs, or if the interrupt instruction control is not functioning properly. The computer is initialized with the start order code signal, which not only forces the sequence generator to execute the start instruction, but also resets many other computer circuits. When the start order code signal is being issued, the T12 stop signal is sent to the timer. This signal stops the time pulse generator until all essential circuits have been reset and the start instruction has been forced by the sequence generator. The computer may also be initialized manually when connected to the peripheral equipment and placed into the monitor stop mode. In this mode, the time pulse generator is held at the T12 position until the monitor stop signal is released.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

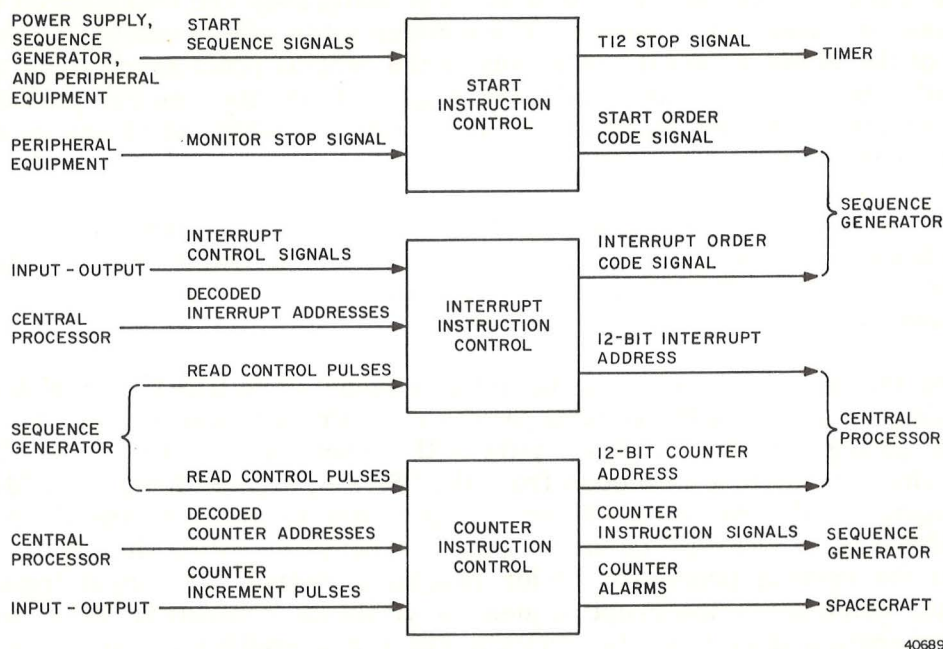


Figure 2-28. Priority Control Block Diagram

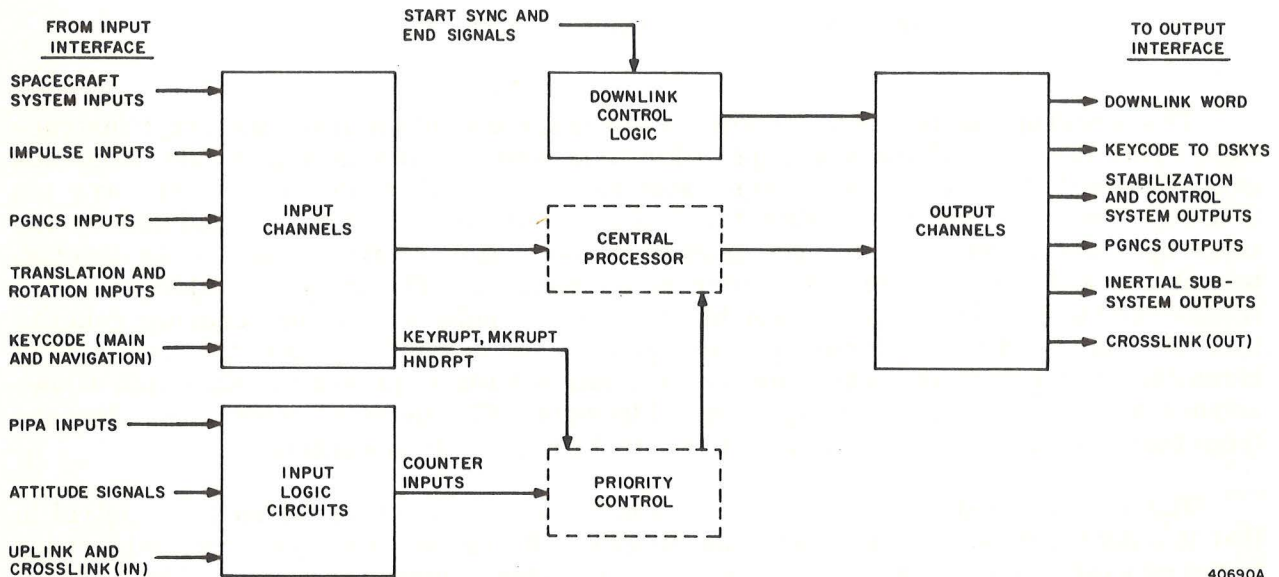
The interrupt instruction control can force the execution of the interrupt instruction. This is done with the interrupt order code signal which is sent to the sequence generator and the twelve-bit address sent to the central processor. There are ten addresses each of which accounts for a particular function that is regulated by the interrupt instruction control. The interrupt instruction control links the keyboards, telemetry, and time counters to program operations. The interrupt addresses are transferred to the central processor by read control pulses from the sequence generator. The source of the keyboard, telemetry, and time counter inputs is the input-output circuits. The interrupt instruction control has a built-in priority chain which allows sequential control of the ten interrupt addresses. The decoded interrupt addresses from the central processor are used to control the priority operation.

The counter instruction control is similar to the interrupt instruction control in that it links input-output functions to the program. It also supplies twelve-bit addresses to the central processor and instruction signals to the sequence generator. The instruction signals cause a delay (not an interruption) in the program by forcing the sequence

generator to execute a counter instruction. The addresses are transferred to the central processor by read control pulses. The counter instruction control also has a built-in priority of the 29 addresses it can supply to the central processor. This priority is also controlled by decoded counter address signals from the central processor. The counter instruction control contains an alarm detector which produces an alarm if an incremental pulse is not processed properly.

2-6.7 INPUT-OUTPUT. The input-output section accepts all inputs to, and routes to other systems all outputs from, the computer. The input-output section, (figure 2-29) includes the interface circuits, input and output channels, input logic, output timing logic, and the downlink circuits.

Most of the input channels and the output channels are flip-flop registers similar to the flip-flop registers of the central processor. Certain discrete inputs are applied to individual gating circuits which are part of the input channel structure. Typical inputs to the channels include keycodes from the DSKY's, signals from the PGNCs proper and other spacecraft systems as shown. Input data is applied directly to the input channels; there is no write process as in the central processor. However, the data is read out to the central processor under program control. The input logic circuits accept inputs which cause interrupt sequences within the computer. These incremental inputs (acceleration data from the PIPA's etc.) are applied to the priority control circuits and subsequently to associated counters in erasable memory.



40690A

Figure 2-29. Input-Output Block Diagram

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

Outputs from the computer are placed in the output channels, and are routed to specific systems through the output interface circuits. The operation is identical to that in the central processor. Data is written into an output channel from the write lines and read-out to the interface circuits under program control. Typically, these outputs include outputs to the stabilization and control system, the DSKY's, the PGNCS, etc. The downlink word is also loaded into an output channel, and routed to the spacecraft telemetry system by the downlink circuits.

The output timing logic gates synchronization pulses (fixed outputs) to the PGNCS and the spacecraft. These are continuous outputs since the logic is specifically powered during normal operation of the computer and during standby.

2-6.8 MEMORY. Memory (figure 2-30) consists of an erasable memory with a storage capacity of 2,048 words, and a fixed core rope memory with a storage capacity of 36,864 words. Erasable memory is a random-access, destructive-readout storage device. Data stored in erasable memory can be altered or updated. Fixed memory is a nondestructive storage device. Data stored in fixed memory is unalterable since the data is wired in and readout is nondestructive.

Both memories contain magnetic-core storage elements. In erasable memory, the storage elements form a core array; in fixed memory, the storage elements form three core ropes. Erasable memory has a density of one word per 16 cores; fixed memory has a density of eight words per core. Each word is located by an address.

In fixed memory, addresses are assigned to instruction words to specify the sequence in which they are to be executed, and blocks of addresses are reserved for data, such as constants and tables. Information is placed into fixed memory permanently by weaving patterns through the magnetic cores. The information is written into assigned locations in erasable memory with the CTS, the DSKY's, uplink, or program operation.

Both memories use a common address register (register S) and an address decoder in the central processor. When register S contains an address pertaining to erasable memory, the erasable memory cycle timing is energized. Timing pulses sent to the erasable memory cycle timing then produce strobe signals for the read, write, and sense functions. The erasable memory selection logic receives an address and a decoded address from the central processor and produces selection signals which permits data to be written into or read out of a selected storage location. When a word is read out of a storage location in erasable memory, the location is cleared. A word is written into erasable memory through the memory buffer register (register G) in the central processor by a write strobe operation. A word read from a storage location, is applied to the sense amplifiers. The sense amplifiers are strobed and the information is entered into register G of the central processor. Register G receives information from both memories.

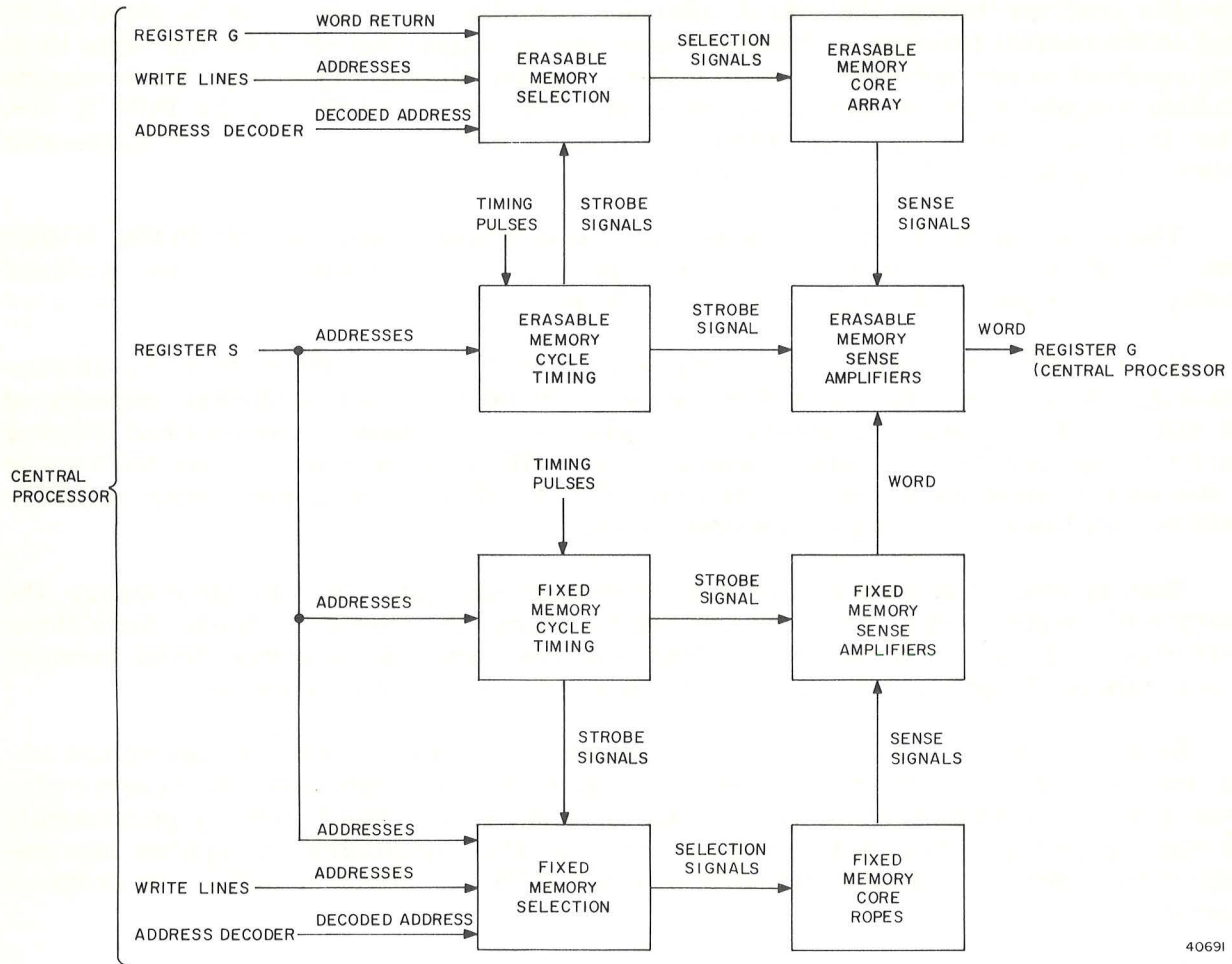


Figure 2-30. Memory Block Diagram

The address in register S energizes the fixed memory cycle timing when a location in fixed memory is addressed. The timing pulses sent to the fixed memory cycle timing produce the strobe signals for the read and sense functions. The selection logic receives an address from the write lines, a decoded address and addresses from register S, and produces selection signals for the core rope. The content of a storage location in fixed memory is strobed from the fixed memory sense amplifiers to the erasable memory sense amplifiers and then entered into register G of the central processor.

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

2-6.9 POWER SUPPLIES. The two power supplies (figure 2-31) furnish operating voltages to the CMC and the DSKY's. Primary power of 28 vdc from the spacecraft is applied to both power supplies. Regulator circuits maintain a constant output of (+4 volts) and (+4 volts switched) from one supply, and (+14 volts) and (+14 volts switched) from the other. The regulator circuits are driven by a sync signal input from the timer, each power supply having a different sync frequency. During system and subsystem test, inputs from the CTS can be used to simulate power supply failures.

The standby mode of operation is initiated by pressing the standby (STBY) button on the DSKY. During standby, CMC is put into a RESTART condition and the switchable +4 and +14 voltages are switched off thus putting the CMC into a low power mode where only the timer and a few auxiliary signals are operative.

The voltage alarm circuits monitor the +28, +14, and +4 volt outputs and produce a CMC restart signal (fresh start) should any of the voltages deviate from nominal by more than a predetermined amount. The oscillator alarm produces a CMC restart

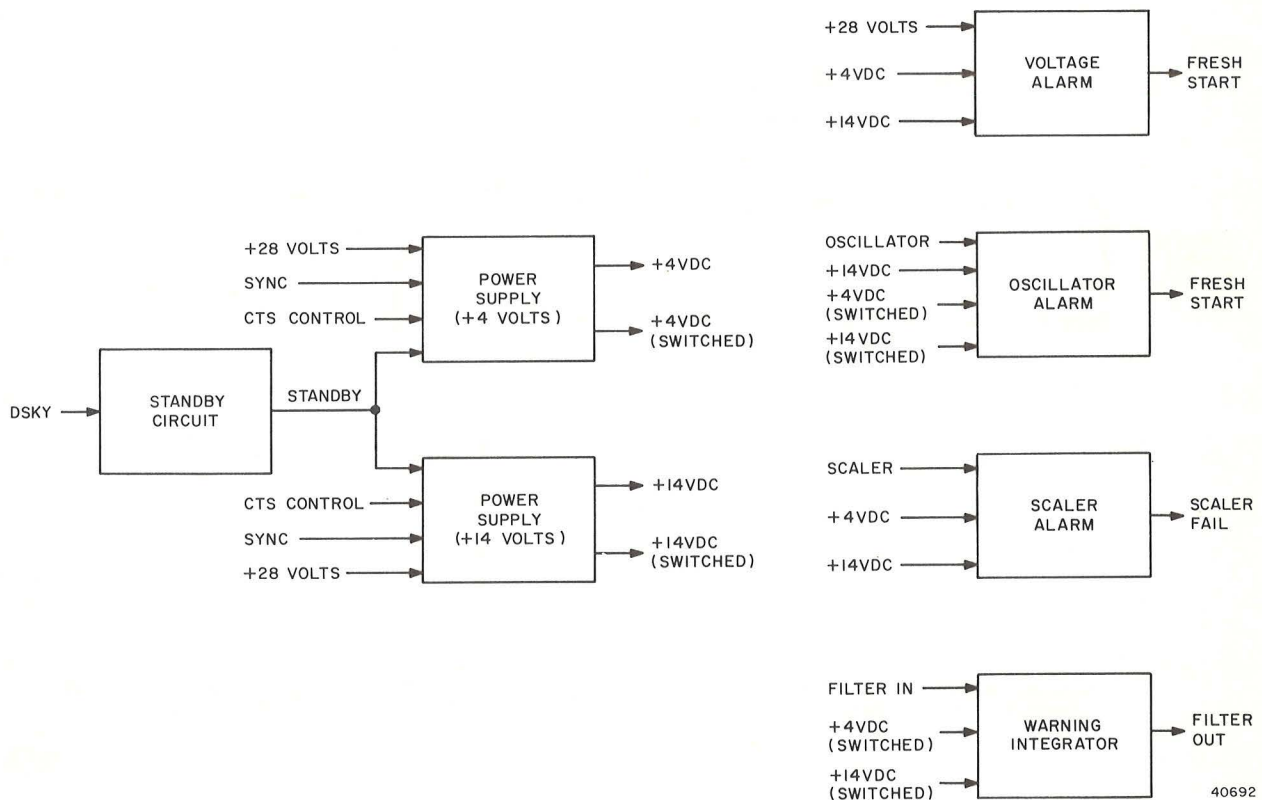


Figure 2-31. Power Supplies Block Diagram

signal (fresh start) if the oscillator fails or if the CMC is in the standby mode. The scaler alarm circuit monitors the scaler output of the timer and generates fail signal if the scaler output fails. The warning integrator monitors certain operations and generates a CMC warning signal (filter out) if these operations are frequently repeated or prolonged.

2-6.10 DISPLAY AND KEYBOARD. Two identical and interchangeable DSKY's are associated with the CMC: one on the main D and C panel designated the main DSKY, and the second on the lower D and C panel of the PGNCS designated the navigation DSKY.

Each DSKY (figure 2-32) consists of a keyboard, relay matrix with associated decoding circuits, displays, mode and caution circuits, and power supply. The keyboard,

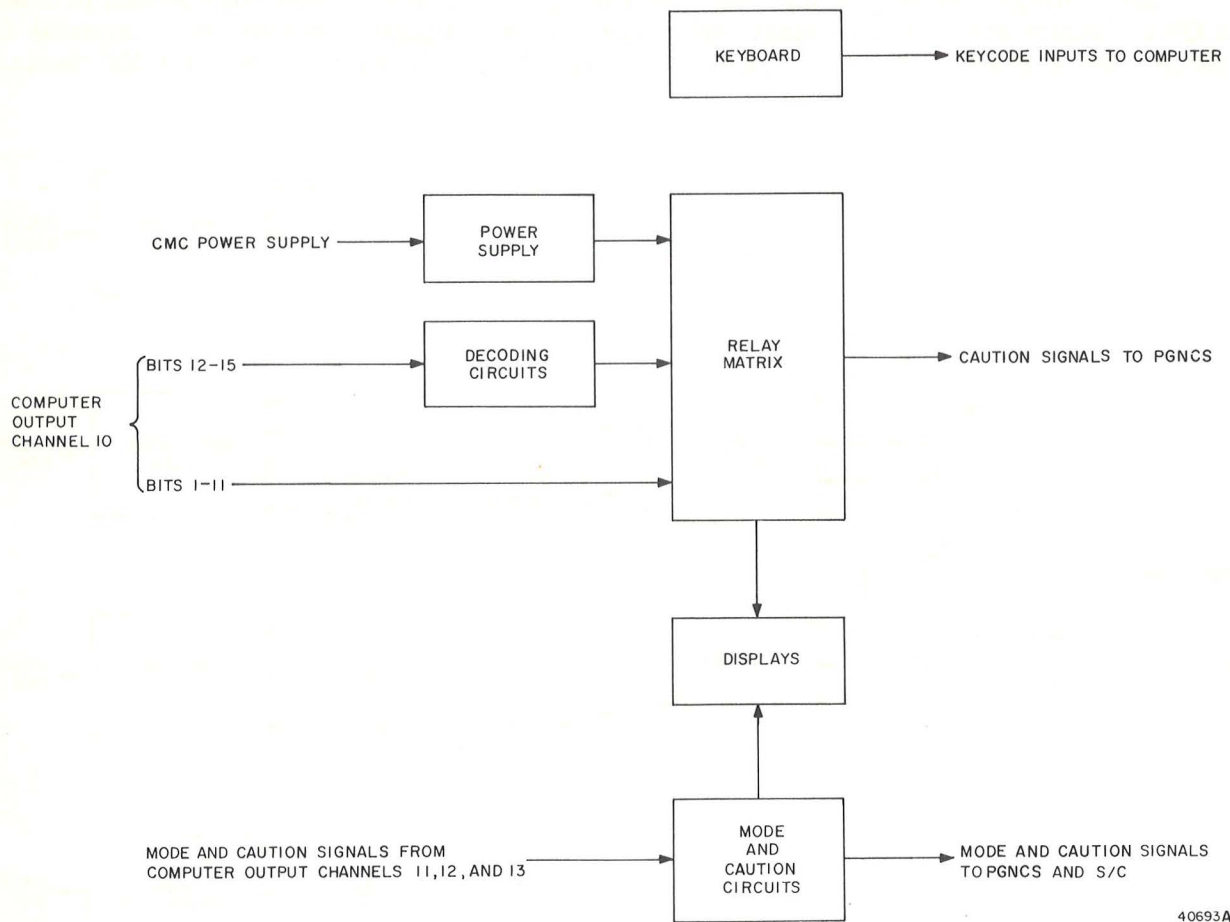


Figure 2-32. Display and Keyboard Block Diagram

BLOCK II PRIMARY GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

which contains several numerical, sign, and other control keys, allows the astronaut to communicate with the CMC. The inputs from the keyboard are entered into an input channel and processed by the CMC.

The inputs entered from the keyboard, as well as other information, appear on the displays after processing by program. The display of information is accomplished through the relay matrix. A unique code for the characters to be displayed is formed by 15 bits from output channel 10 in the CMC. Bits 12 through 15 are decoded by the decoding circuits, and, along with bits 1 through 11, energize specific relays in the matrix which causes the appropriate characters to illuminate. The information displayed is the result of a keycode punched-in by the astronaut, or is computer-controlled information. The display characters are formed by electroluminescent segments which are energized by a voltage from the power supply routed through relay contacts. Specific inputs from the PGNCS are also applied, through CMC, to certain relays in the matrix through output channel 10 of the CMC. The resulting relay-controlled outputs are caution signals to the PGNCS.

The mode and caution circuits accept direct input signals from channels 11, 12, and 13, without being decoded. The resulting outputs can give an indication to the astronaut on the DSKY and/or route the output signal to the PGNCS and spacecraft.