BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Feasibility Study for Simplified Apollo Guidance - Case 310

DATE: March 29, 1968

FROM: R. V. Sperry

MEMORANDUM FOR FILE

On February 13, 1968, Dr. G. E. Mueller asked Bellcomm, TRW (Houston) and IBM (Houston) to "determine the feasibility of meeting the minimum requirements for carrying out the lunar mission by developing a simplified program for the Apollo guidance computer". Attachment 1 is a copy of a letter from Dr. Mueller to Dr. Gilruth, identifying the study.

Presentations were made to the Software Task Force on March 1 and this memorandum contains copies of the viewgraphs presented by Bellcomm (Attachment 2) and supporting memoranda giving details of the study.

The study had two major topics:

1. can the program organization be changed to use a single program interrupt and

2. can the program be simplified so as to require no more than one half the fixed and erasable memory of the computer.

The first topic, use of a single interrupt is discussed in Attachment 3. It is concluded that a program organization with a rigid sequence of computations, using a single timer interrupt to turn off RCS jets and to control the basic cycle of operations is possible. No hardware changes are required and the present counter interrupts should be kept and used.

The second part of the study, to simplify the program to use no more than one half of the available memory in the computer is discussed in Attachments 4 through 12.

To simplify the program extensive use is made of the ground computing facilities and the up-data links, and more crew participation is assumed. The powered flight programs were simplified, navigation simplified and essentially all targeting transferred to the ground. A memory budget was estimated based on the assumed simplifications and compared with the MIT memory budgets and counts for Colossus Rev. 135 and Sundance Rev. 255.

In addition, the listings for Sundisk Rev. 269 and Sunburst Rev. 113 were used to help establish the one half memory budget.

The rationale for the reduction of computer memory in the utility and service routines is discussed in Attachment 4. Essential features here are the removal of all restart programming and extensive use of single precision calculations.

The power flight routines to be used by the command module are discussed in Attachment 5. These routines are reduced to single external ΔV routine, the Lambert Steering is dropped. A single prethrust program will be used and except for a limited on-board capability to return to earth, all targets are externally generated. The orbital integration routines will be dropped and only the Kepler Conic routine will be retained. The platform alignment routines are unchanged.

Attachment 6 gives a simplified entry guidance which has as its basis, work concerned with present entry backup modes. It is simplified to two phases: a constant drag phase and a final phase similar to the present MIT final phase. The constant drag phase selects a constant deceleration, depending upon the range and flight path angle. The final phase will correct for errors committed during the constant drag phase and, therefore, achieve the same overall accuracy. However, the range capability will be reduced from the present requirement of 1,000 nautical miles to about 700 nautical miles.

A simplified CSM autopilot is discussed in Attachment 7. This autopilot takes advantage of the improved knowledge of spacecraft characteristics that are currently available and MIT is proposing to use to update their design. The simplified approach accepts the current performance and uses the better spacecraft bending data to remove features of the current design. For example, the thrust misalignment corrector loop would be dropped. Other simplifications include the use of the same filter for CSM/LM and CSM alone, the use of a simplified RCS digital autopilot and the use of the same autopilot for entry as is used in the CSM-RCS autopilot.

Attachment 8 gives a simplified LM digital autopilot which is an outgrowth of several years work at Bellcomm in evaluation of LM autopilot designs. Simplification is achieved by using first order differences to estimate rate and a simplified pulse ratio modulator for RCS jet control. The descent engine gimbal is not used for control of the attitude on the LM but is driven to direct the thrust through the center of gravity by a very simple law.

In both the LM and CSM autopilots no predictive equations are used, only simple feed-back control. Under nominal conditions, these simplified autopilots will use more RCS fuel than the present MIT designs. However, for off-nominal conditions in which the predictive model is degraded, the simple feed-back control systems uses less fuel. The pointing accuracy is not noticeably changed.

Attachments 9 and 10 discuss simplified LM descent strategies. The strategy of Attachment 9 was used for the estimation of computer memory shown. This strategy is very similar to MIT proposals prior to the inclusion of the landing point designator. In addition, manual control beyond lo-gate is assumed; the current landing point designator, rate of descent mode, automatic landing mode are eliminated. Simple redesignation to the right or left is retained during the visibility phase. Velocity updating from the landing radar to the LM guidance computer is dropped but the velocity displays are retained and the crew uses these to correct the velocity errors after manual take over at lo-gate. The lo-gate point is moved back to about 500 ft. altitude and 1,200 ft. uprange from the landing point. An even further simplified descent philosophy is discussed in Attachment 10. It involves even more crew participation but requires a larger landing ellipse and was, therefore, not considered in the word count.

Attachment 11 gives the simplified LM ascent and rendezvous strategy. The powered ascent is simplified, to consist of a pitch polynomial followed by cross-product steering near the end of the burn. Targeting for CSI and CDH will be done by charts and not included in the computer. Targeting for TPI can be done by the computer, using the Kepler Conic routine with ignition based on elevation angle. TPF will be done manually. A reshaped concentric flight plan is proposed to allow more complete ground support.

Attachment 12 gives a LM rescue strategy which requires no on-board targeting for the CSM.

Attachment 13 gives a simple navigation technique that can be used for the rendezvous navigation by the LM or CSM and can also be used for return to earth navigation in the event of communication failures. The technique uses constant weighting factors and drops the recursive filter technique currently implemented.

The final Attachment (#14) is a letter from W. H. Hittinger to Dr. Mueller which points out the considerable impact on MSC to support such a second software development, some impact on MIT and an impact on crew training. He also summarizes Bellcomm's position in the last paragraph:

"Thus the Bellcomm position is that a backup program meeting your guidelines is technically feasible. Because of the impacts noted above we cannot assess the total value to the space program of going ahead with such a development."

R. V. Sperry

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Attachments

ATTACHMENT 1

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D.C. 20546

IN REPLY REFER TO: M

February 14, 1968

Dr. Robert R. Gilruth
Director
Manned Spacecraft Center, NASA
Houston, Texas 77058

Dear Bob:

As you know, the Software Review Board that I have established at the request of Sam Phillips has been meeting for the past several weeks. One area of concern that the Board has identified is the level of sophistication and complexity inherent in the present MIT programs.

In order to better understand the limitations and the possibilities available to the carrying out of the lunar mission, using the present computer system with a different programming approach, I have asked the support contractors, whom Chris Kraft and Sam Phillips have assigned to the Software Board, to determine the feasibility of meeting the minimum requirements for carrying out the lunar mission by developing a simplified program for the Apollo guidance computer. In particular, I have asked them to determine the feasibility and the cost and schedule implications of carrying out the mission using half the fixed and erasable memory of the computer, using only one interrupt and otherwise trading off simplicity in the program for minor increases in required propellants.

In carrying out this task, IBM (Houston), TRW (Houston), and Bellcomm will require some discussion and direction from your staff, including Flight Operations, Flight Crew, and Guidance and Control. Would you please provide the necessary support so that this work can be completed by March 1, 1968.

Sincerely,

George

George E. Mueller
Associate Administrator
for Manned Space Flight

ATTACHMENT 2

Bellcomm, Inc.

FEASIBILITY STUDY
SIMPLIFIED APOLLO GUIDANCE

- NO MORE THAN HALF AVAILABLE MEMORY
- ONE INTERRUPT
- NO HARDWARE CHANGES

3/1/68

FEASIBILITY INVOLVES

- PROGRAM REORGANIZATION - ONLY ONE INTERRUPT REQUIRES RIGID SEQUENCE OF COMPUTATIONS

- PROGRAM SIMPLIFICATION - HALF MEMORY REQUIRES SIMPLER CONCEPTS, FEWER ALTERNATIVES

INTERRUPT STRUCTURE OF AGC

29 COUNTER INTERRUPTS - NOT PROGRAMMABLE, REQUIRE 1 MACHINE CYCLE TIME (12 μ SEC.), NOT UNDER COMPUTER PROGRAM CONTROL, CANNOT BE INHIBITED. EXAMPLES: PIPA COUNTS, CDU PULSES

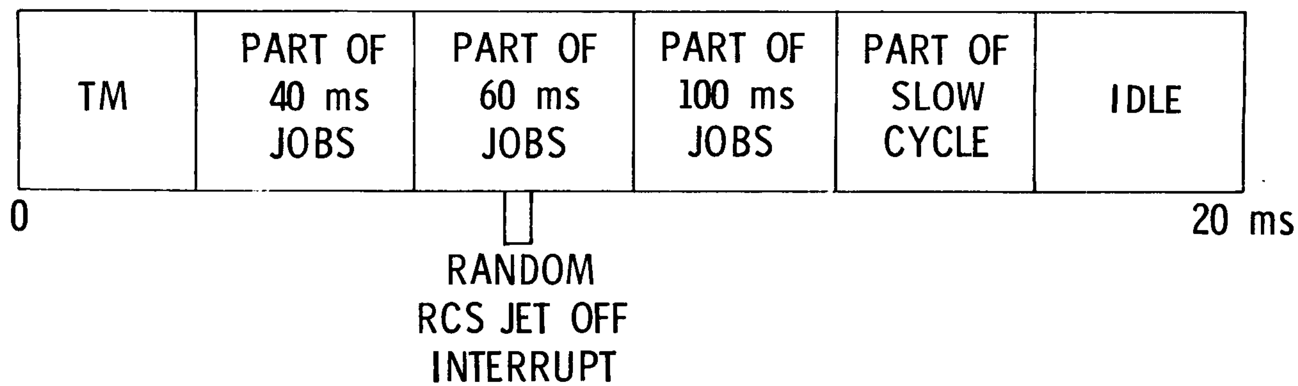
11 PROGRAM INTERRUPTS - 4 CLOCKS, 2 DSKY, 1 UPLINK, 1 DOWNLINK, 1 RADAR, 1 HAND CONTROLLER, 1 RESTART. ALL BUT RESTART CAN BE INHIBITED, BUT ONLY AS A GROUP. SERVICING AN INTERRUPT REQUIRES A MINIMUM OF 4 MACHINE CYCLES (46.8 μ S)

TIMING DEMANDS

- RCS JET OFF COMMANDS AT GRANULARITY OF 1 ms
- DOWN TELEMETRY NEEDS NEW DATA EVERY 20 ms
- SEXTANT MARKS NEED CHECKING ABOUT EVERY 40 ms
- DSKY, HAND CONTROLS NEED CHECKING ABOUT EVERY 100 ms
- DAP CYCLE TIME ABOUT 60 ms
- HOUSEKEEPING CYCLE IS 100 ms
- GUIDANCE, NAVIGATION CYCLE ABOUT 2 SEC. IN POWERED FLIGHT, SLOWER IN COASTING FLIGHT
- UPLINK GIVES NEW DATA EVERY 100 ms

PROGRAM ORGANIZATION

- ENDLESS PROGRAM CYCLE REPEATS EVERY 20 ms ON TIMER INTERRUPT
- TIMER ALSO USED FOR RCS JET OFF INTERRUPT



- ABILITY TO STORE TELEMETRY DATA FOR NEXT 100 ms VERY DESIRABLE HARDWARE CHANGE

COMPUTER MEMORY ESTIMATES

	CMC		LGC	
PROGRAMS AND ROUTINES	MIT BUDGET	B/C BUDGET	MIT BUDGET	B/C BUDGET
UTILITY AND SERVICE	12679	7700	13848	8000
DIGITAL AUTO PILOT	5671	2000	4325	2000
BASIC MATH	4203	1600	4052	1400
TARGETING	1520	0	1390	0
NAVIGATION	2570	500	1600	300
POWERED FLIGHT	2390	1300	3048	1200
ALIGNMENT	927	700	671	400
MISSION CONTROL	4535	2000	4479	2200
MISC.	1234	1100	1354	1100
	35729	16900	34767	16600

SIMPLIFICATION REQUIRES

- INCREASED GROUND PARTICIPATION
- INCREASED CREW PARTICIPATION

COMMAND MODULE MEMORY ESTIMATES
(COLOSSUS REV. 135)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
UTILITY AND SERVICE ROUTINES			
PINBALL INC. NOUN TABLES	3000	2875	2000
INTERPRETER + ETC.	2262	2198	1200
T4 RUPT	875	789	800
DISPLAY INTERFACE	700	678	100
SYSTEMS TEST	650	633	---
IMU MODE SWITCHING	580	573	580
EXTENDED VERBS	550	526	500
DOWN LINK + LISTS	500	371	350
FRESH START & RESTART	420	396	270
RESTART ROUTINES & TABLES	400	374	---
SXT MARK	353	358	250
EXECUTIVE	328	332	500
SELF CHECK	310	314	310
PROGRAM SELECT	300	268	---
IMU COMPENSATION	250	246	100
WAIT LIST	240	240	240
RTB OP CODES	200	147	100
PHASE TABLE MAINT.	183	183	---
MISC.	578	586	360
SUBTOTAL	12679	12087	7660 (7700)

COMMAND MODULE MEMORY ESTIMATES (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
AUTOPILOT AND MANEUVER PROGRAMS			
DIGITAL AUTO PILOTS			
ENTRY	850	824	2000
CSM DAP	3421	3386	
INERTIA CALL	---	100	0
KALCMANU	710	728	50
ATTITUDE MANEUVER	60	53	0
CREW MANEUVER	16	11	0
VECPOINT	130	130	130
REND. FINAL ATTITUDE	30	23	30
MIDDLE GIMBAL DISPLAY	75	64	0
CM BODY ATTITUDE	200	195	0
SUBTOTALS	5592	5414	2210 (2000)

COMMAND MODULE MEMORY ESTIMATES
(COLOSSUS REV. 135)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
BASIC MATH ROUTINES			
IN-FLIGHT ALIGNMENT ROUTINES	287	225	225
POWERED FLIGHT SUBROUTINES	156	156	156
CSM GEOMETRY	210	254	250
TIME OF FREE FALL	300	266	50
CONIC SUBROUTINES	1050	1082	800
ORBITAL INTEGRATION	1400	1446	0
PERIAPO	100	78	0
LATITUDE, LONGITUDE, ALTITUDE	170	159	0
INITIAL VELOCITY	175	175	0
LUNAR AND SOLAR EPHEMERIS	75	71	75
PLANETARY INERTIAL ORIENTATION	280	259	0
SUBTOTAL	4203	4171	1556 (1600)
TARGETING ROUTINES			
TRANSFER PHASE INITIATION SEARCH	320	304	0
RETURN TO EARTH	1200	888	0
SUBTOTAL	1520	1192	0

COMMAND MODULE MEMORY ESTIMATES (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
NAVIGATION ROUTINES			
MEASUREMENT INCORPORATION	350	333	20
PREFERRED TRACKING ATTITUDE	310	280	310
LUNAR ORBIT NAVIGATION	750	780	50
LUNAR LANDMARK SELECTION	190	192	10
RENDEZ TRKG SGHTING MARK + BACKUP	90	77	90
RENDEZ TRKG DATA PROCESS + BACKUP	400	434	20
LANDMARK TABLE	150	150	0
CISLUNAR NAVIGATION	330	534	0
SUBTOTAL	2570	2780	500 (500)

COMMAND MODULE MEMORY ESTIMATES (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
POWERED GUIDANCE ROUTINES			
SERVICER	500	401	500
DESIRED THRUST DIRECTION	350	338	238
CROSS PRODUCT STEERING	140	128	100
VG CALCULATION	100	99	0
TIME OF BURN CALCULATION	80	68	0
INITIAL VG	20	17	20
ENTRY GUIDANCE	1200	1169	400
SUBTOTAL	2390	2220	1258 (1300)

COMMAND MODULE MEMORY ESTIMATES (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
ALIGNMENT ROUTINES			
COARSE ALIGN	75	73	75
FINE ALIGN	90	90	90
AUTO OPTICS	155	151	155
SIGHTING MARK	50	49	50
STAR DATA TEST	65	57	65
GYRO TORQUING	40	35	40
PICK-A-PAIR	125	125	0
STAR CATALOG	223	223	113
ALTERNATE LOS SIGHTING MARK	54	54	54
OPTICS CALIBRATION	50	42	50
SUBTOTAL	927	899	692 (700)

COMMAND MODULE MEMORY ESTIMATES (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
MISSION CONTROL PROGRAMS			
PREL OR SERV INITIALIZATION	38	38	38
PREL OR SERV GYROCOMPASSING	397	349	200
OPTICAL VERIF OF GYROCOMP	130	131	0
EARTH ORBIT INSERTION MONITOR	380	376	325
TRANSLUNAR INJECTION	250	---	150
TPI SEARCH	150	---	0
RENDEZVOUS NAVIGATION	160	118	160
GROUND TRACK DETERMINATION	50	66	0
ORBITAL NAVIGATION	100	141	0
CISLUNAR MIDCOURSE NAVIGATION	100	---	0
TPI PRETHRUST	450	138	0
TPM PRETHRUST	100	502	0
RETURN TO EARTH	250	240	50
SOR PRETHRUST	50	-	0

COMMAND MODULE MEMORY ESTIMATES (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
MISSION CONTROL PROGRAMS (CONTINUED)			
SOM PRETHRUST	50	---	0
SPS THRUSTING	560	563	422
RCS THRUSTING	150	128	150
THRUST MONITOR	100	72	0
IMU ORIENT DETERM + BACKUP	265	259	265
IMU REALIGN AND BACKUP	260	240	260
MANEUVER TO CM/SM SEP ATT	320	308	0
CM/SM SEP + PRE-ENT MANEUVER	90	82	0
ENTRY INITIALIZATION	25	17	0
POST	20	6	0
UPCONTROL	20	---	0
BALLISTIC	20	---	0
CONST D PHASE	---	---	50
FINAL PHASE	40	31	0
LM TPI SEARCH	10	---	0
SUBTOTAL	4535	3805	2020 (2000)

COMMAND MODULE MEMORY ESTIMATES (CONTINUED)

PROGRAMS	BUDGET	ACTUAL	BC BUDGET
MISCELLANEOUS PROGRAMS AND ROUTINES			
UPDATE PROGRAM	320	310	320
RENDEZVOUS OUT-OF-PLANE DISPLAY	120	---	120
EXTERNAL DELTA V PRETHRUST	250	244	50
GENERAL LAMBERT MANEUVER	120	---	0
S-BAND ANTENNA DISPLAY	85	80	85
ORBIT PARAMETER DISPLAY	115	290	290
RENDEZVOUS PARAM DISPLAY ROUTINE 1 + 2	100	158	100
TARGET DELTA V	100	108	100
CMC/LSC CLOCK SYNCHRONIZATION	24	26	24
SUBTOTAL	1234	1216	1089 (1100)

LUNAR MODULE MEMORY ESTIMATES
(SUNDANCE REV. 255)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
UTILITY AND SERVICE ROUTINES			
PINBALL	3000	2891	2000
INTERPRETER	2262	2198	1200
T4 RUPT	850	822	800
DISPLAYS	700	678	100
SYSTEMS TEST	675	707	0
IMU MODE SWITCHING	580	573	580
EXTENDED VERBS	600	555	500
DOWN LINK	500	357	350
FRESH START-RESTART	420	398	270
RESTART TABLES	350	287	0
PHASE TABLE MAINTENANCE	183	183	0
AOT MARK	381	386	250
EXECUTIVE	328	332	500
SELF CHECK	310	314	310
PROGRAM SELECT	300	291	0
IMU COMPENSATION	270	268	100
WAIT LIST	240	240	240
RTB OP CODES	200	147	100
RADAR RUPTS	230	181	0
COAS BACKUP MARK	50	15	15
DAP DATA LOAD	160	146	50
RADAR SUBROUTINES	710	737	550
MISC	474	554	340
SUB TOTALS	13773	13260	8255 (8000)

LUNAR MODULE MEMORY ESTIMATES (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
AUTOPILOT AND MANEUVER PROGRAMS			
DIGITAL AUTO PILOT	3250	2701	1604
KALC MANU	680	675	50
ATTITUDE MANEUVER	60	101	0
CREW DEFINED MANEUVER	18	11	0
VECPOINT	130	130	130
REND. FINAL ATTITUDE	30	---	30
BALL ANGLE DISPLAY	82	59	0
MIDDLE GIMBAL DISPLAY	75	64	0
SUBTOTALS	4325	3741	1814 (2000)

LUNAR MODULE MEMORY ESTIMATE (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
BASIC MATH ROUTINES			
IN-FLIGHT ALIGNMENT ROUTINES	250	225	225
POWERED FLIGHT SUBROUTINES	156	156	156
LM GEOMETRY	96	99	99
TIME OF FREE FALL	300	266	0
CONIC SUBROUTINES	1050	1082	800
ORBITAL INTEGRATION	1400	1453	0
PERIAPO	100	78	0
LATITUDE, LONGITUDE, ALTITUDE	170	159	0
INITIAL VELOCITY	175	175	0
LUNAR AND SOLAR EPHEMERIS	75	126	75
PLANETARY INERTIAL ORIENTATION	280	259	0
SUBTOTAL	4052	4078	1355 (1400)

LUNAR MODULE MEMORY ESTIMATE (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
TARGETING ROUTINES			
PREDICTED LAUNCH TIME	650	293	0
COELLIPTIC SEQUENCE INITIATION	440	649	0
CONSTANT DELTA ALTITUDE	300	---	0
SUBTOTAL	1390	942	0
NAVIGATION ROUTINES			
MEASUREMENT INCORPORATION	350	333	40
PREFERRED TRACKING ATTITUDE	50	37	30
RENDEZVOUS NAVIGATION	450	---	50
LUNAR SURFACE NAVIGATION	100	517	0
RR SEARCH, DESIGNATE AND READ	650	592	200
SUBTOTAL	1600	1479	320 (300)

LUNAR MODULE MEMORY ESTIMATE (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
POWERED GUIDANCE ROUTINES			
SERVICER	850	867	500
DESIRED THRUST DIRECTION	230	117	150
CROSS PRODUCT STEERING	75	60	60
VG CALCULATION	100	99	0
TIME OF BURN CALCULATION	105	118	0
DESCENT GUIDANCE	923	913	400
ASCENT GUIDANCE	700	514	100
TRIMGIMB	65	58	0
SUBTOTAL	3048	2746	1210 (1200)

LUNAR MODULE MEMORY ESTIMATE (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
ALIGNMENT ROUTINES			
COARSE ALIGN	75	61	75
FINE ALIGN	80	102	90
AUTO OPTICS	55	65	55
STAR DATA TEST	56	41	45
GYRO TORQUING	40	27	40
PICK-A-PAIR	142	130	0
STAR CATALOG	223	223	113
SUBTOTAL	671	649	418 (400)

LUNAR MODULE MEMORY ESTIMATE (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
MISSION CONTROL PROGRAMS			
PREDICTED LAUNCH TIME	200	---	0
ASCENT GUIDANCE	200	160	100
RENDEZVOUS NAVIGATION	200	138	140
GROUND TRACK DETERMINATION	50	66	0
LUNAR SURFACE NAVIGATION	20	18	0
PREFERRED TRACKING ATTITUDE	70	54	50
CSI PRETHRUST	70	93	0
CDH PRETHRUST	100	134	0
TPI PRETHRUST	460	123	0
TPM PRETHRUST	95	516	0
SOR PRETHRUST	50	---	0
SOM PRETHRUST	50	---	0

LUNAR MODULE MEMORY ESTIMATE (CONCLUDED)

PROGRAM	BUDGET	ACTUAL	BC ESTIMATE
MISSION CONTROL PROGRAMS (CONT'D)			
DPS THRUST	900	975	500
RCS THRUST	150	42	50
APS THRUST	30	18	30
SEPARATION MONITOR	50	---	0
THRUST MONITOR	70	53	0
IMU ORIENTATION	210	253	210
IMU REALIGNMENT	154	158	150
LUNAR SURFACE ALIGN	500	503	350
LANDING BRAKING	200	137	180
LANDING APPROACH	150	---	130
LANDING (AUTO)	100	---	0
LANDING (ROD)	100	---	0
LANDING (MANUAL	100	---	100
DPS ABORT	100	140	100
APS ABORT	100	122	100
SUBTOTAL	4479	3703	2240 (2200)

LUNAR MODULE MEMORY ESTIMATE (CONTINUED)

PROGRAM	BUDGET	ACTUAL	BC BUDGET
MISCELLANEOUS PROGRAMS AND ROUTINES			
AGS INITIALIZATION	100	114	100
RENDEZVOUS OUT-OF-PLANE DISPLAY	120	111	120
UP DATE PROGRAM	320	310	320
EXTERNAL DELTA V PRETHRUST	150	97	50
GENERAL LAMBERT MANEUVER	120	93	0
RR/LR SELF TEST	102	144	102
LR SPURIOUS RETURN TEST	---	---	---
S-BAND ANTENNA DISPLAY	105	99	105
ORBIT PARAMETER DISPLAY	120	290	120
RENDEZVOUS PARAMETER DISPLAY	90	158	90
TARGET DELTA V	100	108	100
CMC/LSC CLOCK SYNCHRONIZATION	27	26	27
SUBTOTAL	1354	1550	1134 (1100)

LV MONITOR

- FDAI, DSKY DISPLAY
- DROP ATTITUDE ERROR NEEDLES

CSM POWERED FLIGHT

- EXTERNAL ΔV, GROUND TARGET
- LM TARGETS LM RESCUE IF NEEDED
- ON-BOARD TARGETING FOR ABORT IF NEEDED

NAVIGATION

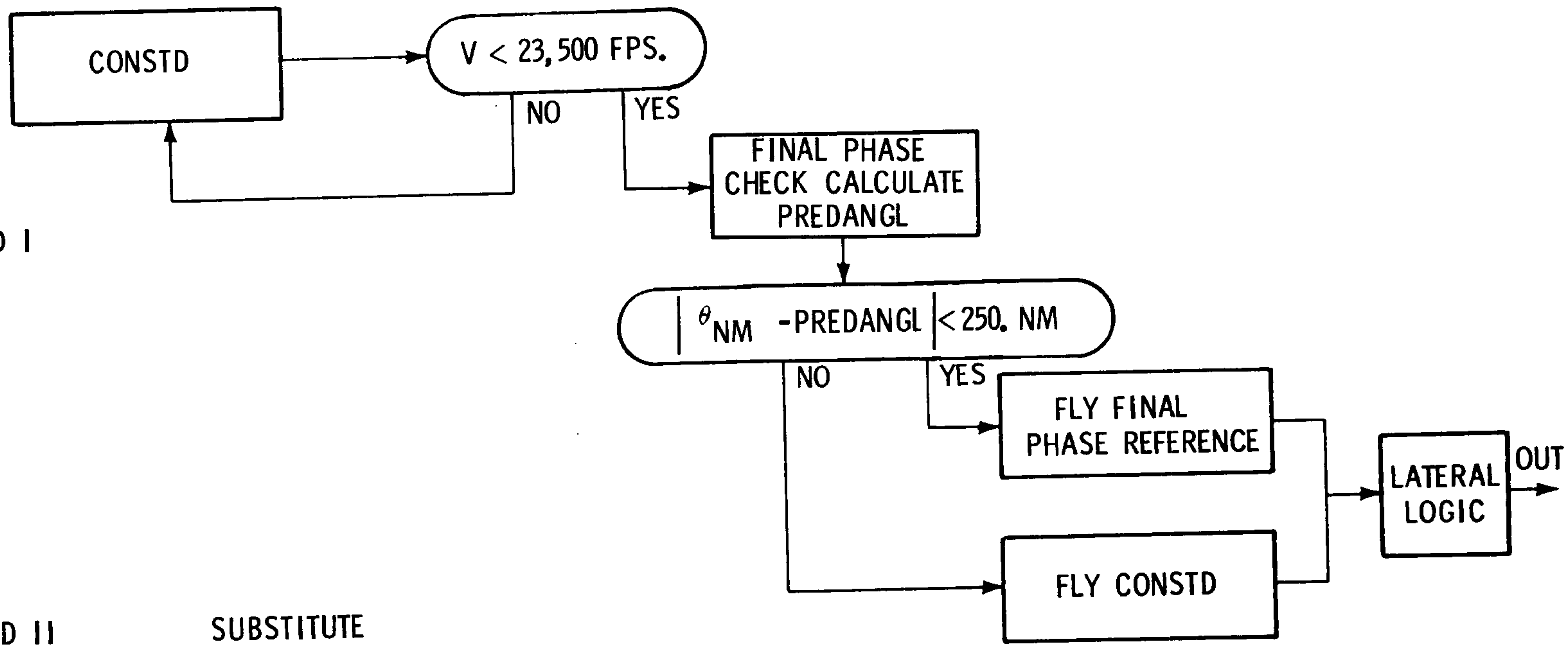
- LANDMARK PROCESSING ON GROUND
- USE SIMPLE FILTER (NOT KALMAN) FOR NAVIGATION
- DROP ORBITAL INTEGRATION AND MOST DOUBLE PRECISION
- PLATFORM ALIGNMENTS UNCHANGED

SIMPLE ENTRY

- CONSTANT DRAG - LEVEL DEPENDS ON RANGE AND FLIGHT PATH ANGLE (AS IN PRESENT BACK UP PROCEDURES)
- USE MIT's FINAL PHASE
- NOT THOROUGHLY VERIFIED

CONSTANT-G ENTRY GUIDANCE

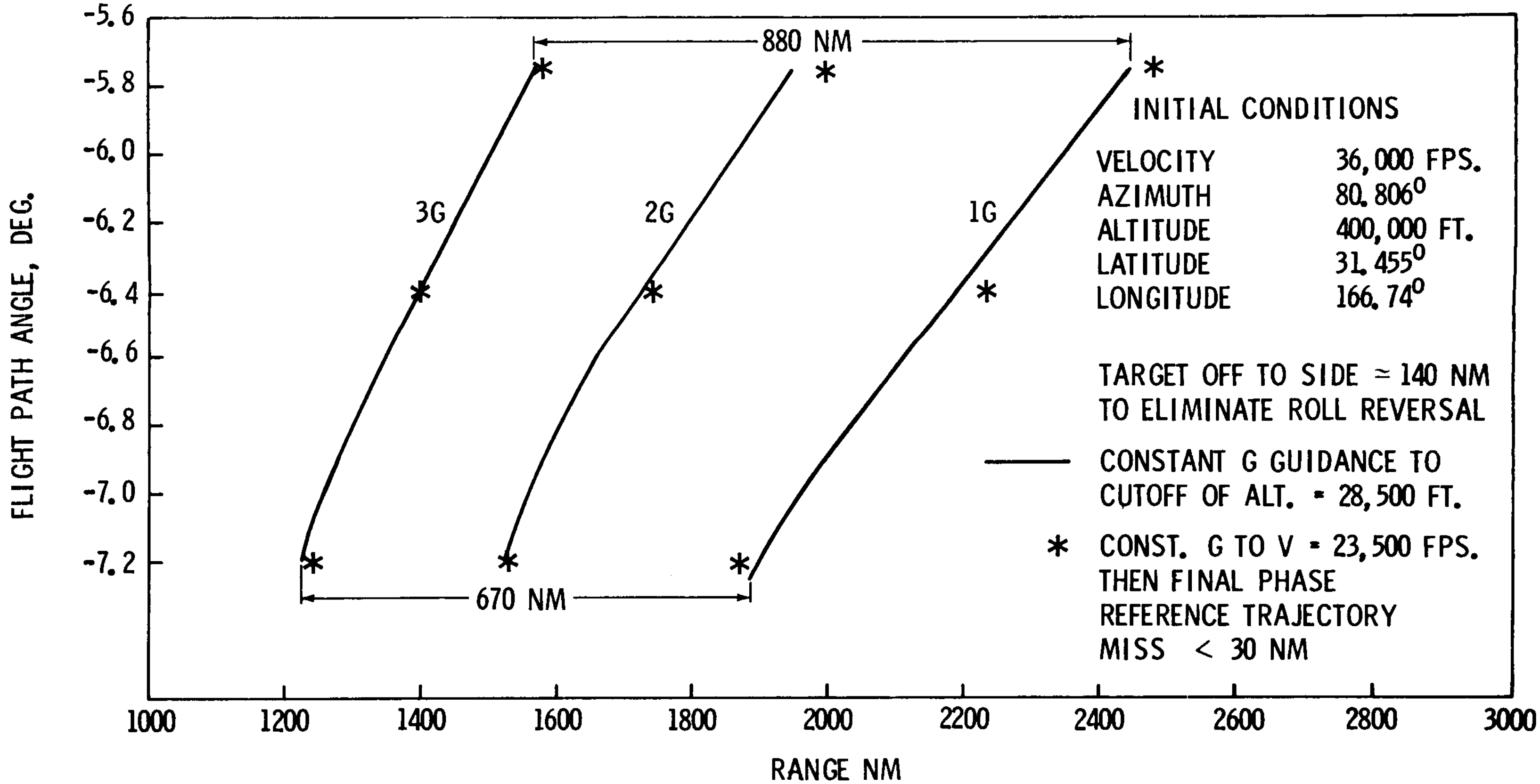
METHOD I



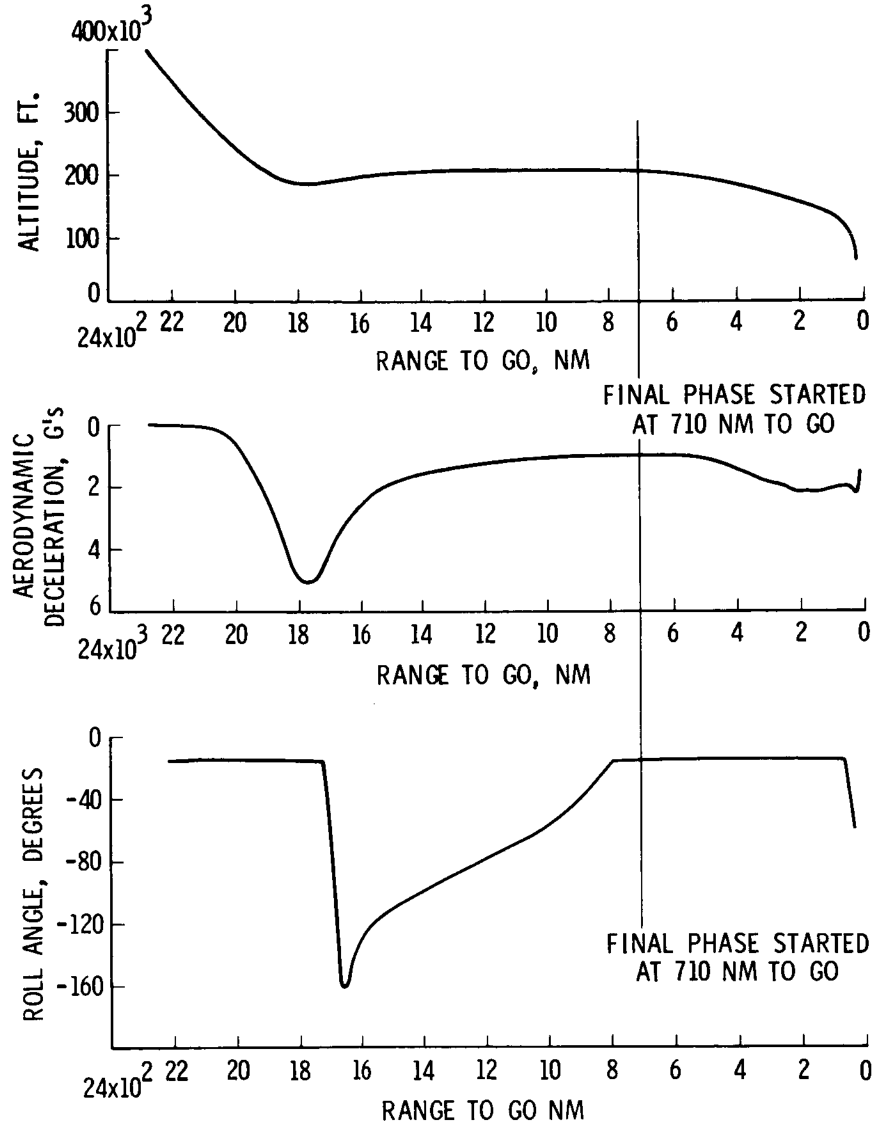
METHOD II SUBSTITUTE

$\theta_{NM} < 700.$ NM FOR V < 23,500. FPS.

ENTRY GUIDANCE USING CONSTANT G



CONSTANT G - FINAL PHASE REFERENCE TRAJECTORY ENTRY GUIDANCE



G LEVEL 1 G	INITIAL LAT. 31.45^{0} N	RANGE-TO-GO 2271.5 NM
$\gamma = -6.4^{0}$	INITIAL LONG. 166.74^{0}	FINAL MISS 8. NM
V = 36,000 FPS.	AZIMUTH 80.806^{0}	

NO LATERAL LOGIC IN GUIDANCE. TARGET OFF TO SIDE

CSM DIGITAL AUTO PILOTS

CSM-TVC DAP

- ELIMINATE THRUST MISALIGNMENT CORRECTOR LOOP
- IMPLEMENT SINGLE SIXTH ORDER FILTER FOR BOTH CSM AND CSM/LM
- IMPLEMENT FILTER IN CASCADED NODAL FORM - SINGLE PRECISION
- USE CONSTANT GAIN FOR CSM-SWITCH FOR CSM/LM

CSM-RCS DAP

- USE SIMPLE DIGITAL VERSION OF PULSE RATIO MODULATOR

CM ENTRY DAP

- EXTERNAL ATMOSPHERIC - SAME AS CSM-RCS DAP
- INTRA-ATMOSPHERIC - USE ROLL PORTION OF CSM-RCS DAP WITH PITCH AND YAW RATE DAMPING

NOT THOROUGHLY VERIFIED

LM DIGITAL AUTOPILOTS

RCS - DAP

- FIRST ORDER DIFFERENCE TO ESTIMATE RATE
- USE DIGITAL VERSION OF PULSE RATIO MODULATOR
- JET SELECT LOGIC IN PITCH, YAW AND ROLL

TRIM-GIMBAL CONTROL

- CENTERS THRUST VECTOR THROUGH THE CENTER OF GRAVITY

NOT THOROUGHLY VERIFIED

COMPARISON MIT-LAD AUTOPILOTS

	MIT-DAP (SUNDANCE)	LAD-DAP
1. RCS AUTOPILOT		
a. STATE ESTIMATOR	NON-LINEAR FIRST ORDER FILTER COMBINED WITH A UP-DOWN COUNTER ESTIMATION OF RATE AND ACCELERATION	FIRST ORDER DIFFERENCE TO COMPUTE RATE ONLY
REQUIRED PARAMETERS	SUCCESSIVE ATTITUDE READOUTS, INERTIAS, TORQUE OF RCS JETS, MOMENT OFFSET, RATE AT n-1, TRIM GIMBAL DISPLACEMENT, THRUST OF DPS ENGINE, LEVER ARM WEIGHTING FACTORS	SUCCESSIVE ATTITUDE READOUT
b. COMPUTATION OF JET FIRING TIME	RUFLAW AND TJETLAW FOR 1. DRIFTING FLIGHT 2. POWERED FLIGHT WITH TRIM GIMBAL CONTROL 3. POWERED FLIGHT WITHOUT TRIM GIMBAL CONTROL	COMPUTE ATTITUDE AND RATE ERROR USE Ton-Toff CHARACTERISTIC TO COMPUTE Ton (JET FIRING TIME) AND Toff (JET OFF-TIME)
REQUIRED PARAMETERS	RUFLAW RATE AND ATTITUDE LIMITS NET AND OFFSET ACCELERATION TJET AT n-1 INERTIA, TORQUE OUTPUT OF RCS JETS, PHASE PLANE PARAMETERS	RATE GAIN Ton CONSTANT, Toff CONSTANT, Ton-Toff CHARACTERISTIC PARAMETERS

COMPARISON MIT-LAD AUTO PILOT (CONTINUED)

		MIT-DAP (SUNDANCE)	LAD-DAP
c.	JET SELECT LOGIC	IN JET-AXES (45^0 AXES)	IN PITCH, YAW AND ROLL AXIS
	REQUIRED PARAMETERS	DECISION FOR A 1,2, OR 4 JET FIRING BASED UPON MAGNITUDES OF NET ACCELERATIONS	DECISION FOR A 1,2, OR 4 JET FIRING BASED UPON MAGNITUDES OF ATTITUDE AND RATE ERRORS
2.	TRIM GIMBAL CONTROL	TIME-OPTIMAL BANG-BANG CONTROL LAW, TRIES TO CONTROL VEHICLE ATTITUDE	REGULATOR: CENTERS THRUST THROUGH C.G.
	REQUIRED PARAMETERS	INERTIA, THRUST OF DPS ENGINE LEVER ARM, ACCELERATION, RATE AND ATTITUDE ERRORS	SUCCESSIVE VEHICLE RATES

LM DESCENT

- T_{IG} BY GROUND
- LINEAR GUIDANCE LAW WITH HIGH THRUST THROTTLE CONTROL IN BRAKING
- ALTITUDE UPDATE ONLY
- AUTO TO LOW-GATE, MANUAL AFTER
- REDESIGNATE BY YAW CONTROL ONLY
- DROP ROD, AUTO LAND, LPD, VELOCITY UPDATE
- NOT THOROUGHLY VERIFIED

PROPOSED SIMPLIFICATION OF LM DESCENT GUIDANCE

REDUCTION IN CAPABILITY

1.) DELTA V PENALTY FOR SIMPLIFICATION - SMALL

2.) ACCURACY

- a.) NO REDUCTION IN VISIBILITY IN APPROACH PHASE
- b.) NO INCREASE IN LANDING ELLIPSE SIZE
- c.) INCREASE IN LANDING VELOCITY

UNCERTAINTY	(3σ VALUES)
UNCERTAINTY IN RADIAL RATE	12.0 FT/SEC.
UNCERTAINTY IN DOWN RANGE VELOCITY	12.0 FT/SEC.
UNCERTAINTY IN OUT-OF-PLANE VELOCITY	12.0 FT/SEC.
RSS	20.7 FT/SEC.

3.) FUNCTION - NO AUTOMATIC REDESIGNATION OF LANDING SITE CAPABILITY

LM ASCENT

- T_{IG} BY GROUND, CHART BACKUP
- POLYNOMIAL PITCH, GROUND OR CHART AZIMUTH CROSS PRODUCT GUIDE TOWARD END
- NO LUNAR SURFACE NAVIGATION
- NOT THOROUGHLY VERIFIED

LM RENDEZVOUS

- CSI, CDH BY CHARTS
- TPI ON-BOARD TARGETING
- TPF MANUAL
- ALL BURNS EXTERNAL ΔV
- SIMPLE RELATIVE NAVIGATION

SIMPLE NAVIGATION

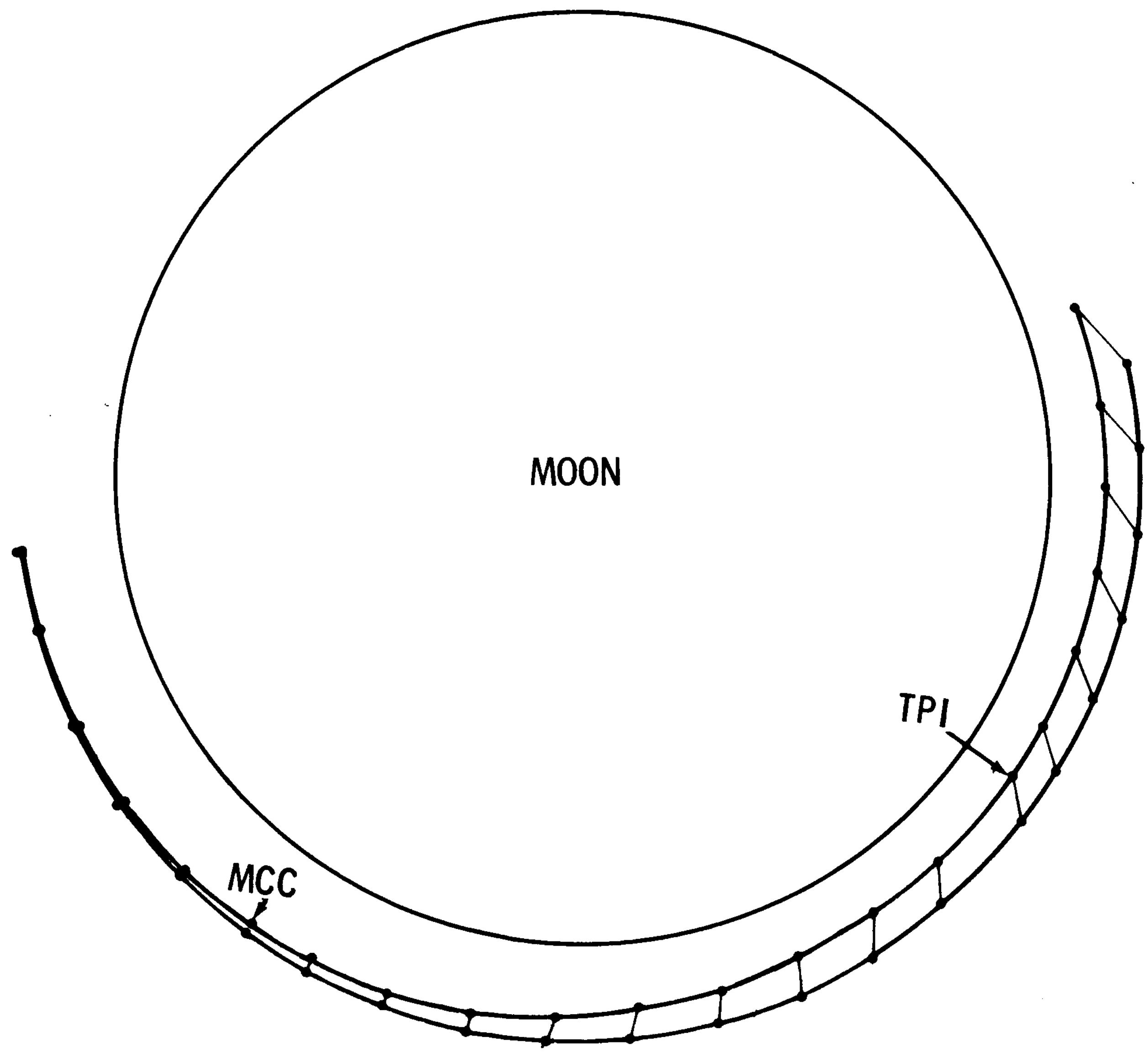
$$\hat{R}_{REL\,+} = \hat{R}_{REL\,-} + 0.5\left(\tilde{R}_{REL} - \hat{R}_{REL\,-}\right)$$

($\sim$ = MEASURED, $\wedge$ = ESTIMATE)

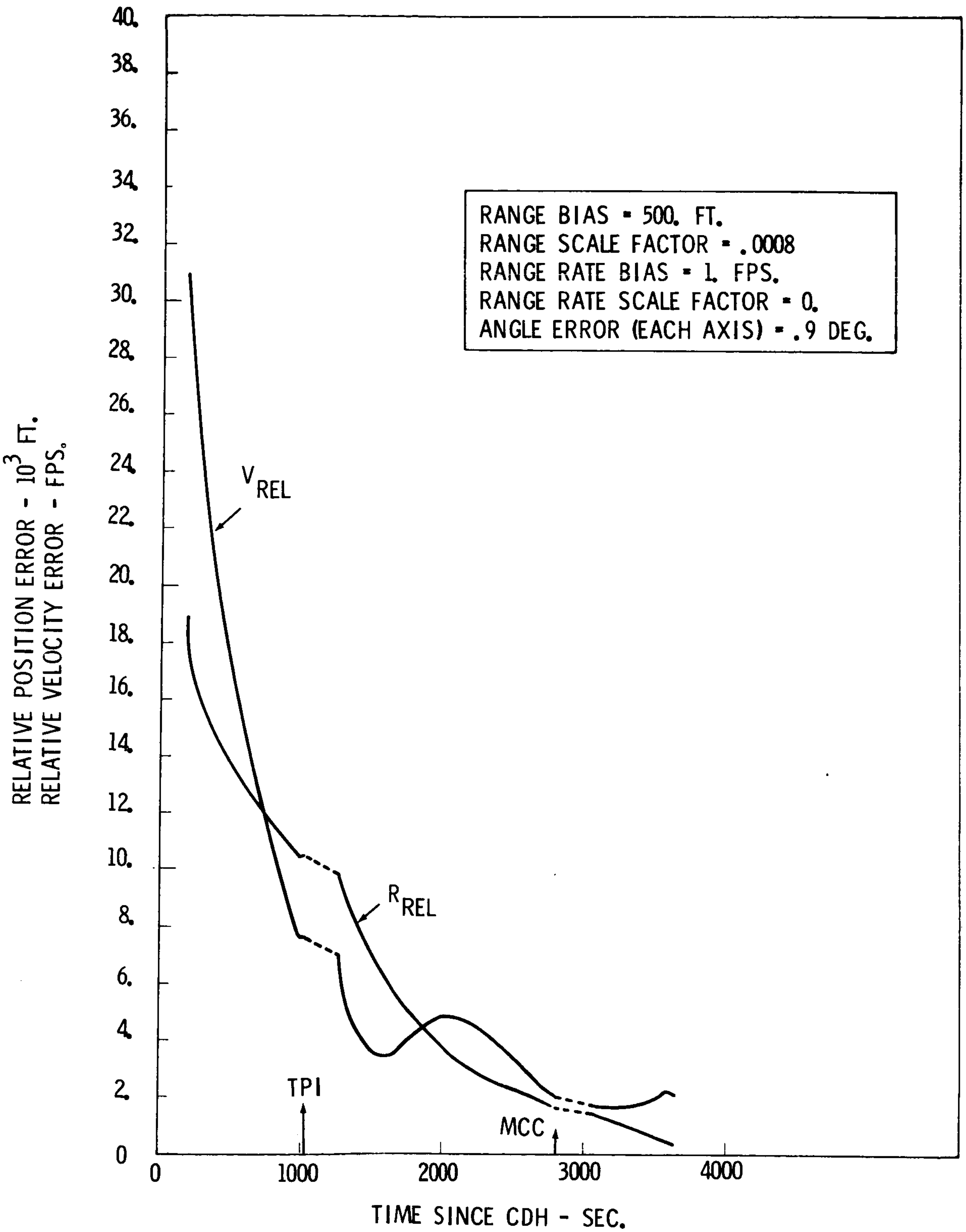
$$\hat{V}_{REL\,+}(t) = \hat{V}_{REL\,-}(t) + \frac{0.1}{\Delta t}\left\{\left(\tilde{R}_{REL} - \hat{R}_{REL\,-}\right)_{t} - \left(\tilde{R}_{REL} - \hat{R}_{REL\,+}\right)_{t-\Delta t}\right\}$$

$$\hat{V}_{\substack{REL\\LOS}\,+} = \hat{V}_{\substack{REL\\LOS}\,-} + 0.5\left(\tilde{V}_{\substack{REL\\LOS}} - \hat{V}_{\substack{REL\\LOS}\,-}\right)$$

TRAJECTORY USED IN NAVIGATION STUDY



POSITION & VELOCITY ERROR AFTER NAV. UPDATES
PNGCS, RENDEZVOUS RADAR, R & $\dot{R}$



CONCLUSION

HALF SIZE PROGRAM IS TECHNICALLY FEASIBLE

- PROVIDES BACKUP USING SIGNIFICANTLY DIFFERENT APPROACH
    - STRUCTURED TO MAKE VERIFICATION EASIER
    - SIZED TO PERMIT ADDITIONAL FUNCTIONS REQUIRED FOR LATER MISSIONS
- MINOR HARDWARE CHANGES WOULD MAKE JOB EASIER

IMPACT SUMMARY

• ΔV BUDGET	-	ESSENTIALLY UNCHANGED
• RCS BUDGET	-	INCREASED BY USE OF MANUAL CONTROL AND NON-PREDICTIVE AUTO PILOTS
• ACCURACY	-	LUNAR LANDING, EARTH ENTRY ESSENTIALLY UNCHANGED
• RTCC	-	PROCESS SEXTANT LANDMARK TRACKING. COMPENSATE FOR CONIC NAVIGATION. REPROGRAM TO DUPLICATE FLIGHT PROGRAM
• AGC	-	HARDWARE CHANGES DESIRABLE - CHANGE PROGRAM INTERRUPTS INTO DISCRETES, STORE TELEMETRY DATA
• CREW	-	RETRAIN OR TRAIN DIFFERENT CREW

PROGRAM

- APPROXIMATELY 200 MAN YEARS EFFORT
- APPROXIMATELY 2 YEARS DEPENDING ON PEOPLE
- IMPACT ON MIT
- IMPACT ON MSC

ATTACHMENT 3

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Organization of Half Size AGC Computer Program - Case 310

DATE: March 27, 1968

FROM: W. G. Heffron

MEMORANDUM FOR FILE

Introduction

The short feasibility study recently conducted by Bellcomm, Inc. for the Apollo Guidance Software Task Force, NASA, had two goals. One was to use only half the available memory, the other to use only one interrupt. This memorandum reports on the one interrupt part of the study.

By way of definition, an interrupt is a signal to the computer to stop activity in a routine, branch to another routine, perform necessary computations and then return to the original routine.

Present or Interrupt Structure of the AGC

Figure 1 notes that the Apollo Guidance Computer has two types of interrupts, which are distinctly different.

Counter Interrupts are special hardware features of the AGC. As an example, when an integrating accelerometer outputs a pulse representing some 0.1 fps sensed velocity change, a counter interrupt occurs which changes the contents of an AGC memory work by one count. Later, when programmed to do so, the AGC routine can examine this word to determine how much velocity change has been sensed. Counter interrupts are a hardware feature, not a software feature, and involve the computer only in that a memory word is used and in that one machine cycle is used to bump the word content by one count when the interrupt occurs. Programmers do not need to know about counter interrupts.

The alternative to a counter interrupt is an accumulating register external to the AGC, which is read (serially, usually) by the AGC when programmed to do so. The data is then transferred into an AGC memory word. This alternative thus requires as much AGC storage if not more, as much time if not more (especially for serial transfer), and additional hardware. It was concluded, therefore, that these interrupts should be used and be considered only as a hardware feature of the AGC, not as true interrupts.

Program Interrupts are signals, response to which must be programmed. For example, to turn the engine off at a precise time, the AGC software sets an internal clock to a value equal to the time-to-go. The clock counts down to zero, and zero causes an interrupt. The program branches to a location in memory uniquely associated with this particular interrupt, determines that "engine off" is the proper response, issues the command, and then returns to the original routine in progress before the interrupt.

In some cases, it is understood that the interrupt routine does even less than this before returning to the routine in progress. Instead of actually doing the special computations, the task is entered into the waitlist (see next paragraph) as a job waiting to be done.

Present AGC program design depends greatly on these program interrupts. Various tasks are assigned priority levels from 1 to 32 and at frequent convenient times, the software routine in progress is changed, becoming the one having the current greatest priority (until a higher priority job enters the queue). For example, when a specific task, e.g. navigation, is to begin two seconds later, this time is entered into a waitlist, an interrupt occurs two seconds later and the navigation is executed as soon as its priority is the greatest in the then current queue of tasks awaiting execution.

Timing Demands

An alternative program organization is to check frequently enough if it is time to do something, rather than doing it at "exactly" the required time. Timing demands are shown in Figure 2, and are the basis for the discussion which follows.

The ability to turn RCS jets off with a time granularity of 1 ms is highly desirable because fuel can be saved in that way. But checking every 1 ms to see if it is time to do so takes too much time from other activities. This function is then the task which must be kept as an interrupt.

Down telemetry can accept new data (two words) every 20 ms. The rate used to be slower, and this speed was recognized as necessary and implemented. It should therefore be kept, although it is somewhat more often than desirable from a programming viewpoint. And the return to the telemetry routines should be fairly synchronous at this rate.

Sextant marks should be recognized and serviced within some 40 ms of their occurrence, if the sextant data is to be useful in lunar landmark tracking.

The Digital Autopilot Routines require new computations about every 60 ms. Service should be fairly synchronous, since the digital filters in the autopilot depend upon a constant time interval between gimbal angle readings.

DSKY (Display and Keyboard) entries and hand controls should be serviced within some 100 ms of their occurrence, if crew members are not to notice a delay in response to their actions.

Uplink telemetry should be serviced within about 100 ms of the arrival of the uplink information. Communication system performance, not the AGC, determines this rate.

The G & N housekeeping cycle is about 100 ms and should be maintained. This includes such operations as Inertial measuring unit status check, temperature checks, accelerometer failure tests, etc., although not all tests are done every 100 ms.

Alternative Program Organization

These program interrupts have a hardware basis as well as a software basis. To avoid changing hardware, the software is changed so that a program interrupt posts a flag indicating the need to do a certain task.

The one interrupt allowed is used as a timer interrupt having two purposes--to turn RCS jets off and to initiate the 20 ms basic cycle, as is shown in Figure 3.

Down telemetry is the first part of this 20 ms cycle--every 20 ms two new words are transferred to the output channels.

The digital autopilot 60 ms cycle job follows immediately after the telemetry operations because this maintains the cycle of 60 ms best. Autopilot calculations are begun on the first, fourth, seventh, etc. 20 ms cycle and one-third of the autopilot is done each 20 ms cycle.

Sextant marks (40 ms service rate) come next. Here 40 ms means that the flag should be checked at least every 40 ms. Actually once the mark is made it is usually several seconds before a second mark occurs, so servicing the mark can be permitted to take more than 40 ms. But the servicing job must be divided into many parts each of which fits into the 20 ms cycle.

The other jobs fit into the 20 ms cycle in much the same way--each must be divided into parts which can fit into the 20 ms cycle, and the sum of the parts must in no case exceed 20 ms. For the general situation in which less than 20 ms is required, the idle period shown is included, and the exact 20 ms cycle insured thereby.

In this organization the executive has two levels. The high level one switches from 20 ms to 60 to 40 ms, etc., jobs. At the lower level, the pieces of the 100 ms (and 60 and 40 ms) tasks are connected together into a meaningful sequence. This overhead is estimated to cost about 1% of machine time: the idle overhead is estimated at about 10% (based on the LM Abort Guidance System experience).

Thus under this organization there is a forced idle period, which means that the AGC must be somewhat slower than it is at present. Under the present scheme the AGC is idle only when there is nothing to do. Because of this, it is likely that the slow cycle for guidance would increase to somewhat more than two seconds, estimates being three seconds or less. This should be acceptable, however.

Variations

Variations on this scheme are easily conceived and two are now presented.

One is to add the 20 ms down telemetry to the timer interrupt (giving it three functions instead of two) and set the basic cycle to, say, 90 or 100 ms. The 60 ms job would become a 45 or 50 ms job appearing twice in the 100 ms cycle, while the 40 ms would be either 30, 45, or 50 ms as would be most appropriate. This reduces the overhead somewhat, particularly the idle portion.

Another is to increase the storage capacity of the telemetry so it can hold 10 words (100 ms worth). Then telemetry would not be an interrupt and the 100 ms cycle scheme above could be used.

Conclusions

It is possible to reorganize the AGC computer program to reduce the number of programmable interrupts to only one--a timer interrupt used to turn RCS jets off to within 1 ms granularity and to control a basic 20 ms cycle of operations.

No hardware changes are required for this approach: present counter interrupts should be kept and used, and present program interrupts used to set flags indicating that servicing is required.

Many variations on this basic approach are possible.

2014-WGH-bjh

W. G. Heffron

29 COUNTER INTERRUPTS - NOT PROGRAMMABLE, REQUIRE 1 MACHINE CYCLE TIME (12 μ SEC.), NOT UNDER COMPUTER PROGRAM CONTROL, CANNOT BE INHIBITED. EXAMPLES: PIPA COUNTS, CDU PULSES

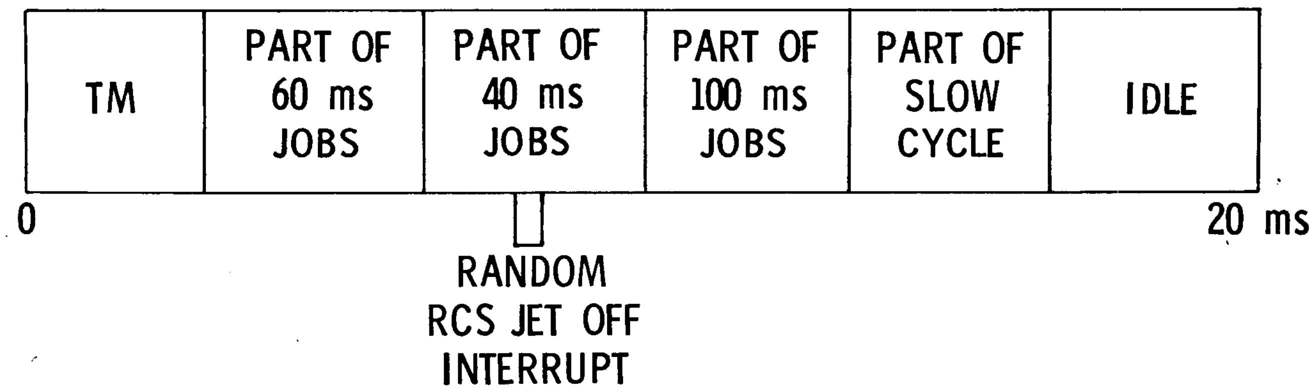
11 PROGRAM INTERRUPTS - 4 CLOCKS, 2 DSKY, 1 UPLINK, 1 DOWNLINK, 1 RADAR, 1 HAND CONTROLLER, 1 RESTART. ALL BUT RESTART CAN BE INHIBITED, BUT ONLY AS A GROUP. SERVICING AN INTERRUPT REQUIRES A MINIMUM OF 4 MACHINE CYCLES (46.8 μ S)

FIGURE 1 - INTERRUPT STRUCTURE OF AGC

- RCS JET OFF COMMANDS AT GRANULARITY OF 1 ms
- DOWN TELEMETRY NEEDS NEW DATA EVERY 20 ms
- SEXTANT MARKS NEED CHECKING ABOUT EVERY 40 ms
- DSKY, HAND CONTROLS NEED CHECKING ABOUT EVERY 100 ms
- DAP CYCLE TIME ABOUT 60 ms
- HOUSEKEEPING CYCLE IS 100 ms
- GUIDANCE, NAVIGATION CYCLE ABOUT 2 SEC. IN POWERED FLIGHT, SLOWER IN COASTING FLIGHT
- UPLINK GIVES NEW DATA EVERY 100 ms

FIGURE 2 - TIMING DEMANDS

- ENDLESS PROGRAM CYCLE REPEATS EVERY 20 ms ON TIMER INTERRUPT
- TIMER ALSO USED FOR RCS JET OFF INTERRUPT



- ABILITY TO STORE TELEMETRY DATA FOR NEXT 100 ms VERY DESIRABLE HARDWARE CHANGE

FIGURE 3 - PROGRAM ORGANIZATION

ATTACHMENT 4

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: AGC Reprogramming Study--Utility and Service Programs - Case 320

DATE: March 29, 1968

FROM: J. J. Rocchio

ABSTRACT

A brief analysis of the utility and service portion of the Apollo Command Module and Lunar Module guidance computer programs was conducted in support of Bellcomm's study of the feasibility of achieving a substantial reduction in the size and complexity of the AGC software. The current allocation for utility and service programs in both the CM and LM computers is on the order of 12,700 words. The first cut analysis indicates a potential saving of 5000 words from this total.

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: AGC Reprogramming Study--Utility and Service Programs - Case 320

DATE: March 29, 1968

FROM: J. J. Rocchio

MEMORANDUM FOR FILE

1. INTRODUCTION

As part of Bellcomm's study of the feasibility of reducing the size and complexity of the programs for the Apollo command module and lunar module guidance computers, a brief investigation of the utility and service routines was made. Since these programs are either identical or very closely related in both computers (except for the routines associated with the LM radars, which have no counterpart in the CMC) they can be discussed without differentiating between the CM and LM.

The baseline for the study was an MIT/IL document providing budgeted and actual memory word allocations for CM program Colossus Rev. 135. These are shown in Table 1, along with the estimated allocations for the proposed reduced program, for all routines categorized under the heading Utility and Service Programs. Individual routines in this category allocated less than 100 words of memory are lumped under a miscellaneous heading. As the requirements on the utility and service programs are essentially mission independent, it was assumed that these allocations were representative of the lunar landing mission programs.

2. BASIC ASSUMPTIONS

The objective of the programming study was to examine the feasibility of reducing the requirements on the guidance computer program by simplifying computer operations and sacrificing nonessential accuracy or minor savings in required propellants. Analysis of the tradeoffs associated with the required guidance computer functions by the study group led to the following assumptions pertinent to potential reductions in the size of the utility and service routines:

a. Extensive use of single precision arithmetic operations, in place of currently used double precision, was feasible.

b. Computer operations in response to a hardware-generated restart would be greatly simplified, and the requirements for "restart proofing" of programs would be eliminated.

c. The number of nouns defined for DSKY use would be reduced and the requirements on input and display scaling would be relaxed.

d. A time-multiplexed executive system was feasible and would result in a strict time cadence of programs in each mission phase. This would replace the current job list, priority-based executive.

3. PROGRAM SIZE ESTIMATES

Each of the major programs grouped under the category of Utility and Service Routines was analyzed from a functional viewpoint. In some instances, as for example, with the restart programs, the assumptions outlined above allowed a complete elimination of a function, and the corresponding size estimate could be reduced to zero. In the other cases the proposed size reductions were based either on the elimination of some set of subfunctions performed by a given routine or a general reduction in the overall requirements of the mission programs.

3.1 Major Reductions

3.1.2 Pinball and Display Interface

The Pinball program incorporates the logic, control, and data manipulation capabilities required for the exchange of information between the astronaut and the guidance computer. A major contribution to the size of this program is the number of nouns defined for a given mission. Each defined noun requires entries in one or more tables. In addition, many nouns require special scaling routines for input and display. The 30% reduction in the Pinball allocation is based primarily on reducing the number of nouns and the associated scaling routines.

In addition, the set of programs under the label Display Interface are related to DSKY operations. These routines provide for various interlocks that prevent overlapping usage of the DSKY by conflicting mission programs. The almost complete elimination of this allocation was based on the proposed change in executive structure.

3.1.3 Interpreter

The AGC interpreter was also analyzed in some detail. The basis for the almost 50% reduction in memory allocation was primarily a result of the assumption of increased use of single precision arithmetic operations. On the basis of the time available for this study it was not possible to determine in fact whether an interpreter was required. However, even if there was no longer a need for it, these 1200 words allocated were felt to be a reasonable estimate of the size of those subroutines (such as the trigonometric functions) which would still be required.

4. CONCLUSION

The total reduction in the memory allocation budget for utility and service programs amounts to about 5000 words, as shown in Table 1. It must be emphasized that estimates of program requirements are notoriously difficult. Furthermore, the time available for this study allowed only a first cut at analyzing the change in requirements for utility and service routines that would result from the simplifications in G&N mission operations and in the associated computer programs. Thus, the estimate that the Utility and Service Functions in a reprogrammed AGC would require about 7700 words must be interpreted with caution.

J. J. Rocchio

1031-JJR-sel J. J. Rocchio

UTILITY AND SERVICE ROUTINES

Program	Budget	(Colossus Rev.135) Actual	Proposed
Pinball and Noun Tables	3000	2875	2000
Interpreter + Subroutines	2262	2198	1200
T4 Rupt	875	789	800
Display Interface	700	678	100
Systems Test	650	633	___
IMU Mode Switching	580	573	580
Extended Verbs	550	526	500
Down Link + Lists	500	371	350
Fresh Start & Restart	420	396	270
Restart Routines & Tables	400	374	___
Extent Mark	353	358	250
Executive	328	332	500
Self Check	310	314	310
Program Select	300	268	___
IMU Compensation	250	246	100
Wait List	240	240	240
RTB Op. Codes	200	147	100
Phase Table Maintenance	183	183	___
Miscellaneous	578	586	360
TOTALS	12,679	12,087	7660
SAVINGS	5019	4327	

TABLE I

BELLCOMM, INC.

Subject: AGC Reprogramming Study--Utility and Service Programs - Case 320

From: J. J. Rocchio

Distribution List

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ATTACHMENT 5

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Simplified Software for the Apollo Guidance Computer - CSM Powered Flight Programs
Case 310

DATE: March 11, 1968

FROM: D. A. Corey

ABSTRACT

A substantially simplified software budget is discussed for the powered flight related portions of the Apollo Guidance Computer programs. Maximum dependence on ground computation facilities has been assumed and on board capability is retained only where absolutely required for mission execution and safety. It is felt that this budget, which requires about half of the current MIT non erasible budget, would have a negligible impact on the probability of mission success.

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Simplified Software for the Apollo Guidance Computer - CSM Powered Flight Programs
Case 310

DATE: March 11, 1968

FROM: D. A. Corey

MEMORANDUM FOR FILE

INTRODUCTION

This memorandum describes the basic philosophy and implementation of a simplified software package for the Apollo Guidance Computer insofar as the powered flight portions of the Apollo Lunar Missions are concerned. Some of the computer programs and routines are also used in other phases of the mission; these requirements are reflected in the figures presented.

The basic philosophy of the simplifications relates to maximum dependence on ground assistance during all powered flight maneuvers. Generally speaking, there is no actual increase in the amount and type of ground computation requirements, although specific parameters to be computed and uplinked differ from current requirements in several cases.

It is this writer's opinion that the impact of the simplified powered flight programs on the probability of mission success is virtually negligible. There is some loss of backup capability, but under current mission planning this capability would only be used in the event of highly unlikely situations, for example, loss of all ground communications with both the CSM and the LM for an extended period of time.

Table II presents a list of the powered flight related programs and routines currently programmed for the AGC based on the list of routines for MIT's version called COLOSSUS Rev. 135. Three estimates of the number of fixed memory words required are presented: (a) MIT's budget, (b) the current count of the actual number of words used, and (c) the proposed budget for the simplified programs. The following paragraphs will explain the changes proposed for the routines which would have a word count different from MIT's budget.

DESCRIPTION OF PROGRAM AND ROUTINE CHANGES

P11 - EOI Monitor - The portion of the program related to computation of the launch vehicle attitude deviations from a prescribed pitch polynomial during the interval from 10 to 133 seconds after launch has been removed. The attitude as displayed by the "8 Ball" will remain and the DSKY displays will be unchanged.

P15 - TLI Monitor - No on-board guidance computations will be done. The magnitude of ΔV and the current attitude rate will be displayed on the DSKY.

P17 - TPI Search - This program has been eliminated. The ground or the LM will do the targeting.

P30 - External ΔV Prethrust - This program has been simplified considerably. The compensation of $\Delta\vec{V}$ for maneuver angle is removed. State vector extrapolation to ignition time less 30 seconds and to ignition time is done on the ground. ΔV is sent up in platform coordinates, eliminating the conversion from local vertical. The Tgo routine is not called.

P31 - General Lambert Maneuver - All powered flights are to be performed in the External ΔV mode. There are no Lambert maneuvers so this program is not required.

P34/P74 - TPI Prethrust - The TPI maneuver will be targeted from the ground or by the LGC and executed in the External ΔV mode. TPI Prethrust becomes the same as External ΔV prethrust.

P35/P75 - TPM Prethrust - This maneuver is targeted on the ground or by the LGC and executed in the External ΔV mode. No on-board targeting capability is retained.

P37 - Return to Earth - Only an extremely limited on-board targeting capability is retained, consisting basically of astronaut controlled iteration using the Kepler routine.

P38 - SOR Prethrust - The maneuver, if done at all, will be performed in the External ΔV mode with external targeting.

P39 - SOM Prethrust - See P38.

P49 - SPS Thrusting - Some words have been eliminated due to the simpler operation. IMU orientation will always be the same with respect to nominal thrust direction. Some restart protection would be eliminated. Items related to Lambert steering and the Lambert aim point are removed. The estimated reduction is 25% of the current estimates.

P47 - Thrust Monitor - This task will be done by the astronaut. Some of the functions may still be required (for example turning the ullage off automatically) but basic operations such as switch settings will be done before ignition. If steering computations are to be performed at all they will be done from the first call to the routines.

S40.1, S40.2,3 - Desired Thrust Direction - The preferred IMU alignment computations are removed. The computations of initial engine bell trim and the resulting desired gimbal angles are retained. Portions dealing with time of burn are simplified or eliminated. The routine really only needs to be told which propulsion system is to be used. It is estimated that 100 words can be saved.

S40.8 - Cross Product Steering - The equations can be simplified somewhat since C = 0 and CbΔt = 0 in all cases.

S40.9 - VG Calculation - This routine is used with Lambert steering and can be eliminated.

S40.13 - Time of Burn Calculation - This function is done on the ground, eliminating the need for this routine.

Time of Free Fall Routines - These are basically eliminated since they are rather redundant. The Conic routines do substantially the same job. A small budget has been retained to manipulate the Conic routines when certain parameters are required for display purposes.

Conic Routines - The Lambert routine is not required.

Periaps Routine - This function will be done by calling subroutine APSIDES directly.

Latitude Longitude Altitude Routine - If the astronaut needs these quantities, he can ask the ground.

Initial Velocity Routine - This routine is the entry point for Lambert operations, which are not used. Consequently, the routine is not required.

TPI Search Routine - Basic Targeting for CSM/LM Rescue will be done on the ground, with the LGC serving as a backup. A profile for LM rescue is available which places TPI at such a location that sufficient tracking time and computing time are available such that the CSM can receive TPI targeting from the ground. TPI is then performed in the External ΔV mode.

Return to Earth Routines - See P37 Return to Earth.

IMPLICATIONS OF NO LAMBERT STEERING

The impact of External ΔV mode powered flights on mission fuel requirements and on maneuver accuracy appears to be negligible. In the first place, current planning already utilizes the External ΔV mode for everything but the first part of LOI (and maybe TEI). Strong consideration is being given to using it everywhere. In the second place, studies have shown that very little penalty is incurred by using the External ΔV mode in place of Lambert Steering. Reference 1 compares the two steering methods for insertion into an elliptical orbit (first part of LOI). For a 10.15 degree plane change into a 95 x 80 mile orbit, the External ΔV mode required 10 fps more fuel nominally (3274 vs. 3264). A one degree plane change into a 200 x 80 mile orbit required .3 fps more fuel in the External ΔV mode.

Generally, error analysis studies performed at MSC have shown that for direct insertion into a circular lunar parking orbit, the External ΔV mode produces significantly larger errors than does the Lambert steering. The errors are acceptable at the 3σ level, however. The difference between the two methods is much smaller for insertion into a 170 x 60 ellipse, which is the current plan. Reference 2 presents the results of a 550 run Monte Carlo Error Analysis comparing the two methods. The trajectory involved a plane change of 10° and a final nominal orbit of 60 x 170 miles. Table II is extracted from Reference 2 and presents a summary of the results of that error analysis. The following data is extracted from that table, and assumes that the statistics were gaussian and that + 3σ = 98%.

	EXTERNAL ΔV	LAMBERT
	3σ Errors	3σ Errors
Apolune Altitude (N.Mi)	5.3	5.3
Perilune Altitude (N.Mi)	.60	.67
Inclination (deg)	.25	.25
Longitude of ASC. Node (deg)	.25	.26
Burn Time (Sec)	3.51	3.41

The differences in the errors are obviously negligible.

SUMMARY

The simplified powered flight non-erasable word budget presented in this memorandum is principally based on the philosophy that everything that can be done on the ground is done on the ground. On-board backup capability has been retained only where absolutely necessary for mission operations and safety. The on-board capabilities to compute and display various parameters to the astronaut have largely been retained in order to assist them in making decisions where either time or safety is critical.

The proposed complete reliance on the External ΔV mode for powered flight maneuvers actually differs very little from current mission plans, although more parameters would be computed on the ground in the simplified plan.

Close coordination between the AGC programming and the RTCC programming is, of course, especially important in the case of the simplified on-board system. This is a matter of degree, however, since close coordination is also required under current plans. The program impact on the RTCC is judged to be relatively small. Most of the computations required by the simplified system are already made for the current system. Some changes are required in particular parameters to be uplinked.

Substantial non-erasable memory savings have also been obtained by removing most of the on-board maneuver targeting capabilities. It is felt, however, that they would only be required in the most remote of circumstances and the actual impact on the probability of mission success and safety is negligible.

Non-erasable core requirements are completely meaningful only in terms of totals for the entire computer since many routines are used for more than one function. However, the estimated core savings for the powered flight related programs is in the neighborhood of 50% of the current MIT requirements.

D. A. Corey

2011-DAC-vh D. A. Corey

Attachments
References
Table I
Table II

REFERENCES

1. Comparison of the Sensitivity of Lambert and External ΔV Guidances to Dispersions in the LOI Burns of Missions F and G. U. S. Government Memorandum, Robert F. Wiley, NASA/MSC (not yet a published memorandum).

2. A Comparison Analysis Between the External ΔV and Lambert Guidance for the Lunar Orbit Insertion (LOI) Maneuver. U. S. Government Memorandum 68-FM73-81, February 21, 1968, Richard M. Moore, Jr., and Aldo J. Bordano NASA/MSC.

TABLE I

AGC WORD COUNTS OF PROGRAMS AND ROUTINES DEALING WITH CSM POWERED FLIGHTS

		WORD COUNT		
ROUTINE IDENTIFICATION	TITLE	MIT BUDGET	MIT CURRENT ESTIMATE	SIMPLIFIED BUDGET
PROGRAMS AND MISC. ROUTINES				
P11	EOI MONITOR	380	376	325
P15	TLI MONITOR	250	-*	150
P17	TPI SEARCH	150	-	0
P30	EXTERNAL ΔV PRETHRUST	250	244	50
P31	GENERAL LAMBERT MANEUVER	120	-	0
P34/P74	TPI PRETHRUST	450	138	0
P35/P75	TPM PRETHRUST	100	502	0
P37	RETURN TO EARTH	250	240	50
P38	SOR PRETHRUST	50	-	0
P39	SOM PRETHRUST	50	-	0

*Where no current estimate is available, MIT Budget figures are used in the totals.

TABLE I Contd.

ROUTINE IDENTIFICATION	TITLE	WORD COUNT: MIT BUDGET	WORD COUNT: MIT CURRENT ESTIMATE	WORD COUNT: SIMPLIFIED BUDGET
PROGRAMS AND MISC. ROUTINES				
P40	SPS THRUSTING	560	563	422
P41	RCS THRUSTING	150	128	150
P47	THRUST MONITOR	100	72	0
P51/P53	IMU ORIENT & BACKUP	265	259	265
P52/P54	IMU REALIGN & BACKUP	260	240	260
R30	ORBIT PARAMETER DISPLAY	115	290	290
R31-R34	RENDEZVOUS PARAMETER DISPLAY	100	158	100
POWERED GUIDANCE ROUTINES				
	SERVICER	500	401	500
S40.1, S40.2,3	DESIRED THRUST DIRECTION	350	338	238
S40.8	CROSS PRODUCT STEERING	140	128	100
S40.9	VG CALCULATION	100	99	0
S40.13	TIME OF BURN CALCULATION	80	68	0
S41.1	INITIAL GC	20	17	20

TABLE I Contd.

ROUTINE IDENTIFICATION	TITLE	WORD COUNT MIT BUDGET	MIT CURRENT ESTIMATE	SIMPLIFIED BUDGET
BASIC MATH ROUTINES				
	POWERED FLIGHT SUBROUTINES	156	156	156
	TIME OF FREE FALL	300	266	50
	CONIC SUBROUTINES	1050	1082	800
	PERIAPO	100	78	0
	LATITUDE LONGITUDE ALTITUDE	170	159	0
	INITIAL VELOCITY	175	175	0
	INFLIGHT ALIGNMENT ROUTINES	287	225	225
TARGETING ROUTINES				
	TPI SEARCH	320	304	0
	RETURN TO EARTH	1200	888	0
UTILITY AND SERVICE ROUTINES				
	IMU MODE SWITCHING	580	573	580
	IMU COMPENSATION	250	246	100
	IMU STATUS CHECK	30	17	17
	TOTALS	9408	9050	4848

TABLE II

(Extracted from Reference 2)

LOI DISPERSION ANALYSIS SUMMARY

Guidance for LOI	Resulting Apocynthion (n. mi.)	Resulting Pericynthion (n. mi.)	Resulting Inclination (deg)	Resulting Longitude of Ascending Node (deg)	Burn Time (sec)
External ΔV					
Nominal	170.00	59.98	157.99	178.74	381.48
Mean	170.07	59.99	157.98	178.73	381.57
Largest Sample	178.52	60.97	158.34	179.11	388.03
98%	175.65	60.62	158.22	178.97	385.04
2%	164.85	59.41	157.72	178.48	378.03
Smallest Sample	162.53	59.16	157.58	178.35	377.18
Lambert					
Nominal	169.55	60.21	157.99	178.74	381.73
Mean	169.68	60.21	157.98	178.72	381.76
Largest Sample	176.87	69.19	158.35	179.03	386.80
98%	175.05	60.89	158.25	178.98	385.17
2%	164.55	59.56	157.73	178.47	378.25
Smallest Sample	162.40	59.04	157.54	178.28	376.62

BELLCOMM, INC.

Subject: Simplified Software for the Apollo Guidance Computer - CSM Powered Flight Programs

From: D. A. Corey

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ATTACHMENT 6

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Possible Simplification of the Apollo Entry Guidance - Case 310

DATE: March 25, 1968

FROM: I. Bogner

MEMORANDUM FOR FILE

This memorandum describes possible simplifications to the primary entry guidance. They are a means of reducing computer memory requirements.

The present entry guidance is designed to provide for a safe spacecraft entry which permits reaching the target site anywhere within a 1000. N.M. footprint from a nominal 1500. N.M. range to a 2500. N.M. range. To accomplish this task the guidance passes through five modes or phases: three open-loop attitude-hold modes and two closed-loop attitude-controlled modes. In all cases the spacecraft is held in an aerodynamically trim condition. The controlling variable is the roll angle which orients the lift vector to null the down-range and cross-range errors. The primary guidance modes may be described with reference to Figure 1.

1. Initial Roll

   The lift vector is held lift up until 0.05 g is sensed. Following a calculation to determine whether the entry is shallow or steep, the lift is oriented full up or full down to assure a safe entry.

2. At an altitude rate of -700. fps trajectory planning takes place. If an executable plan is not feasible due to a predicted overshoot, the craft is guided to attempt to maintain a constant drag.

3. Assuming an acceptable plan takes form, the first guided phase involves guiding the spacecraft along an internally generated reference trajectory to an exit or skip condition.

4. Ballistic Phase

   The skip portion starts when the sensed deceleration decreases below approximately 0.2 g and ends when it builds up to that value. In the interim the vehicle is maintained in attitude hold.

5. Final Phase

In the final phase the vehicle is guided along an internally stored reference trajectory to the target. Here lift orientation is processed through an acceleration limiting program which prevents lift down if excessive g's are anticipated.

In all cases the roll command is processed through a lateral logic computation which reflects the lift vector about the vertical when lateral range error exceeds lateral range capability.

The proposed simplification to the above described entry guidance has as its basis, work concerned with entry backup modes, performed in the Lunar Mission Analysis Branch, MPAD, at the Manned Spacecraft Center*.

There are two key modes to the proposed scheme:

1 Lift is held full up until 0.05 g is sensed. The spacecraft is then flown at a constant g. The g-level is determined by the initial flight-path angle and the initial range-to-go.

2. When the inertial velocity reduces to some subcircular value, e.g., 23,500. fps, the spacecraft is flown along an internally stored reference trajectory to the target .

Roll reversals, and hence the lateral logic in the program, are eliminated by placing the target landing site off the initial projected ground track. Thus, in holding less than full lift up or down as necessary while maintaining the g level the vehicle pulls always to the right or to the left depending on the direction of target offset.

---

*MSC Internal Note NO. 68-FM-20, January 22, 1968. Status Report on the Lunar Mission Entry Monitoring and Backup Mode Development. J. C. Harpold and J. C. Adams.

Figure 2 represents the results of data for Apollo-type entries, in which it was attempted to fly constant g using the constant-drag guidance equations in the present entry guidance. The abcissa represents the range traversed from the start of entry (altitude = 400,000. ft.) to simulation termination (23,500. ft), while the ordinate is the initial flight-path angle, γ.

Data in the figure is based upon a vehicle with an L/D of 0.38. The figure indicates a range capability of 880. N.M. and 670. N.M. at the corridor boundaries.

Nine additional points, covering the region of interest, are plotted in which it was attempted to reach a particular target. Each point was obtained by first flying a constant g. At a velocity of 23,500. fps the spacecraft was flown along the internally stored reference trajectory. The target was off to the side thus inhibiting the roll-reversal logic in the program. Misses in the nine cases simulated were found to be less than 30. N.M.

Figure 3 illustrates roll, acceleration and altitude histories for one of the nine points.

A gross review of the programming requirements indicates that by eliminating from the existing guidance Huntest, Upcontrol, the Ballistic Phase and Lateral Logic, the remaining phases provide the simplified guidance at a saving of approximately 50. percent of the original memory storage. It may also be noted that, in addition to saving core space, the abridged guidance appears to be compatible with entry backup schemes contemplated for use in lunar-return entries.

I. Bogner

2014-IB-ek I. Bogner

Attachments
Figures 1, 2 & 3

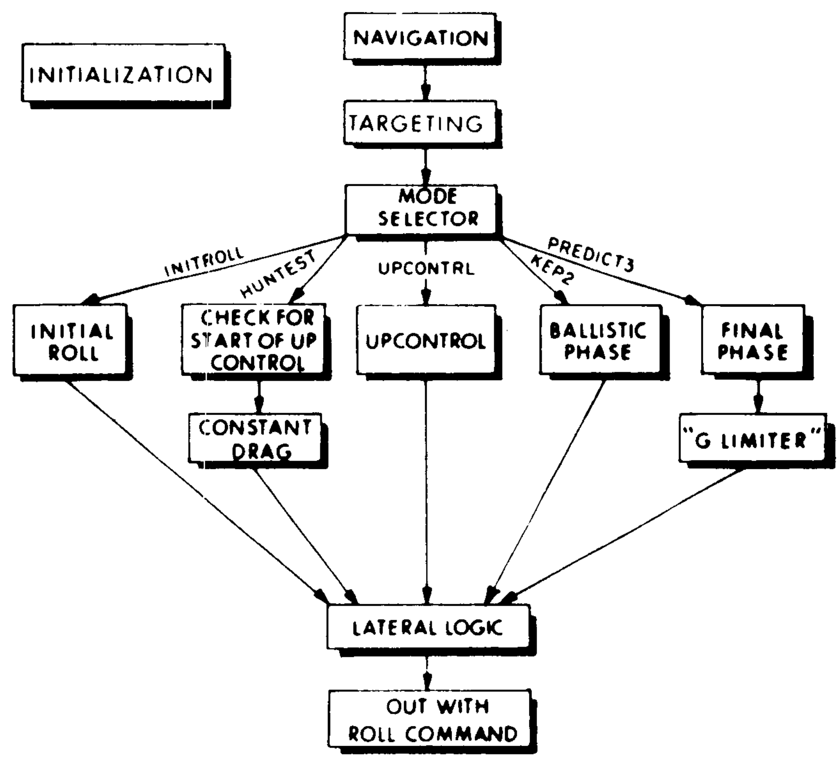


FIGURE 1 - ENTRY COMPUTATION

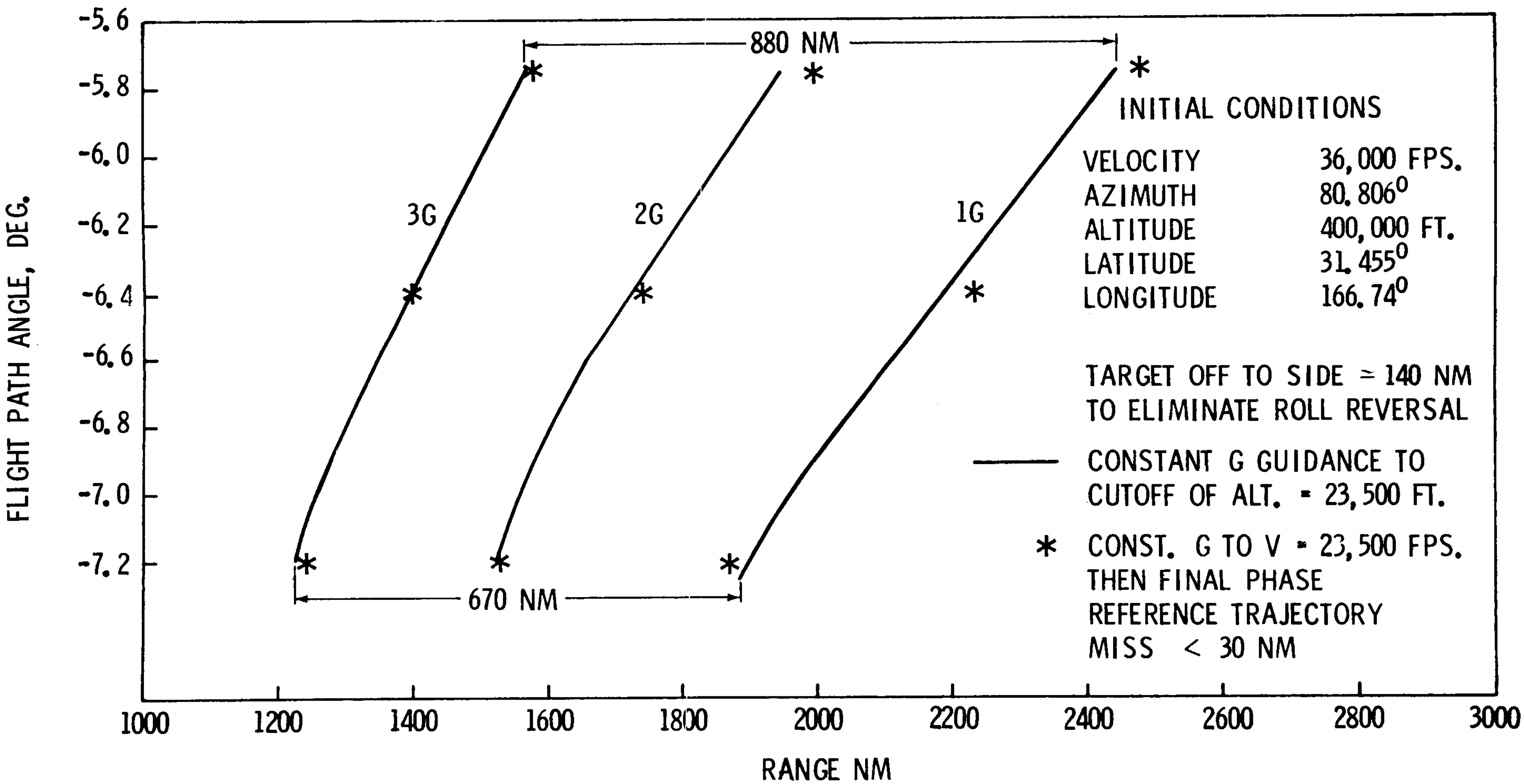
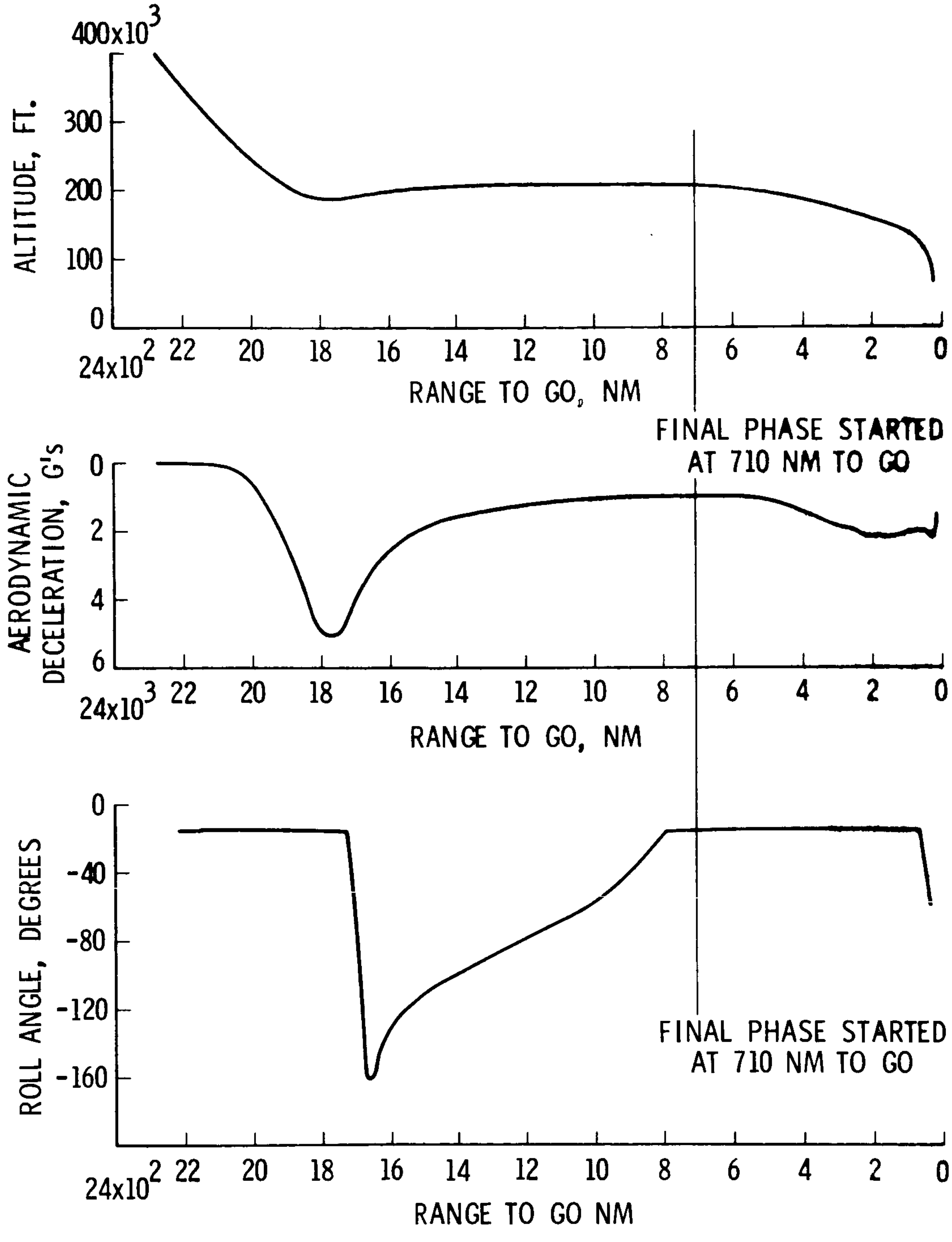


FIGURE 2 - FLIGHT PATH ANGLE vs. RANGE TRAVERSED FOR CONSTANT G ENTRIES



G LEVEL 1 G   INITIAL LAT. 31.45^0 N   RANGE-TO-GO 2271.5 NM

$\gamma = -6.4^0$ INITIAL LONG. 166.74^0 FINAL MISS 8. NM

V = 36,000 FPS.   AZIMUTH 80.806^0

NO LATERAL LOGIC IN GUIDANCE. TARGET OFF TO SIDE

FIGURE 3 - CONSTANT G - FINAL PHASE REFERENCE TRAJECTORY ENTRY GUIDANCE

BELLCOMM, INC.

SUBJECT: Possible Simplification of the Apollo Entry Guidance
Case 310

FROM: I. Bogner

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ATTACHMENT 7

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Proposed Simplification of the CSM-Digital Autopilots - Case 310

DATE: March 27, 1968

FROM: A. Heiber
F. La Piana

ABSTRACT

This memorandum describes proposed simplifications of the CSM digital autopilots (DAP). The objective is to determine the feasibility of using half the fixed and erasable memory of the AGC.

It appears possible that the memory requirement for the CSM-TVC DAP maybe reduced by 50%; the CSM-RCS by 40%; and the CM Entry DAP by 80%.

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Proposed Simplification of the CSM-Digital Autopilots - Case 310

DATE: March 27, 1968

FROM: A. Heiber
F. La Piana

MEMORANDUM FOR FILE

This memorandum describes proposed simplifications of the CSM DAP. The objective is to determine the feasibility of using half the fixed and erasable memory of the AGC.

The following recommendations were made for the CSM-TVC DAP:

1. Eliminate the thrust misalignment corrector (TMC) loop.

2. Implement a sixth order filter for both the CSM and CSM/LM.

3. Implement the filter in cascaded nodal form, single precision.

4. Use constant gain for CSM; gain change for CSM/LM.

The higher gain sixth order filter has recently been proposed by MIT/IL. We would take advantage of the higher gain to eliminate the TMC loop.

The autopilots in the Sundisk program have been compared to the proposed designs in a simulation that was built for the Bellcomm Powered Flight Simulator.

The CSM and CSM/LM vehicle configurations were simulated for each autopilot with a 1/2 degree step input and a transient due to a 1/2 degree mistrim between the thrust vector and the c.g.

In the case of the CSM, the response to the step input is shown in figures 1 and 2. The rise time, overshoot and convergence to final value are superior for the proposed design. There was no c.g. offset for these runs. The step input occurs at 3.83 seconds.

For the CSM, the response to an initial 1/2 degree offset of the c.g. is shown in figures 3 and 4. In order to understand the comparison it is necessary to understand the effect of the TMC loop. The TMC loop acts as a low frequency integrator on the output of the digital filter. It picks up the D.C. component of the digital filter output and feeds it back to the filter output. The D.C. component is equal to the c.g. offset. As a result the TMC loop picks up the value of the c.g. offset and the output of the digital filter can go to zero in the steady state. This implies a zero input error signal to the digital filter. Therefore when the TMC loop is present the vehicle terminal attitude should be the commanded value, zero in the present case.

When the TMC loop is not present, as in the proposed design, the vehicle must hold an attitude offset to compensate for the c.g. displacement. The attitude offset will be equal to the c.g. angular displacement divided by the steady state gain of the filter. The effect of the offset will be taken out by the guidance commands when the guidance loop is closed.

Therefore the significant information in figures 3 and 4 are the rise time and overshoot and not the terminal value.

Figures 5 and 6, and 7 and 8 are a repeat of the above for the CSM/LM vehicle configuration.

These runs demonstrate the feasibility and, in fact, the superiority of the simplified TVC DAP's.

For the CSM-RCS DAP we recommend a simple, digital version of the pulse ratio modulator (PRM). This type of autopilot has been designed and tested at Bellcomm by E. A. Nussbaumer. The PRM is relatively insensitive to off-nominal vehicle parameters.

For the CM Entry DAP we recommend that the external atmospheric DAP use the same autopilot provided for the CSM-RCS. For the intra-atmospheric DAP we would use the roll portion of the CSM-RCS DAP and the same pitch and yaw rate dampers as MIT/IL.

Conclusion

The feasibility of the simplified CSM-TVC DAP has been demonstrated. It is estimated that the simplified DAP would use 50% of the memory required for the present design.

The proposed CSM-RCS DAP is a proven approach and would use approximately 60% of the present memory requirement.

The greatest percentage savings are possible with the entry DAP. Only the pitch and yaw rate dampers must be provided and these are trivial. Therefore it is estimated that 80% of the memory requirement for entry DAP can be saved.

2012-$\frac{\text{AH}}{\text{FL}}$-ek

A. Heiber

F. La Piana

Attachments
Figures 1 thru 8

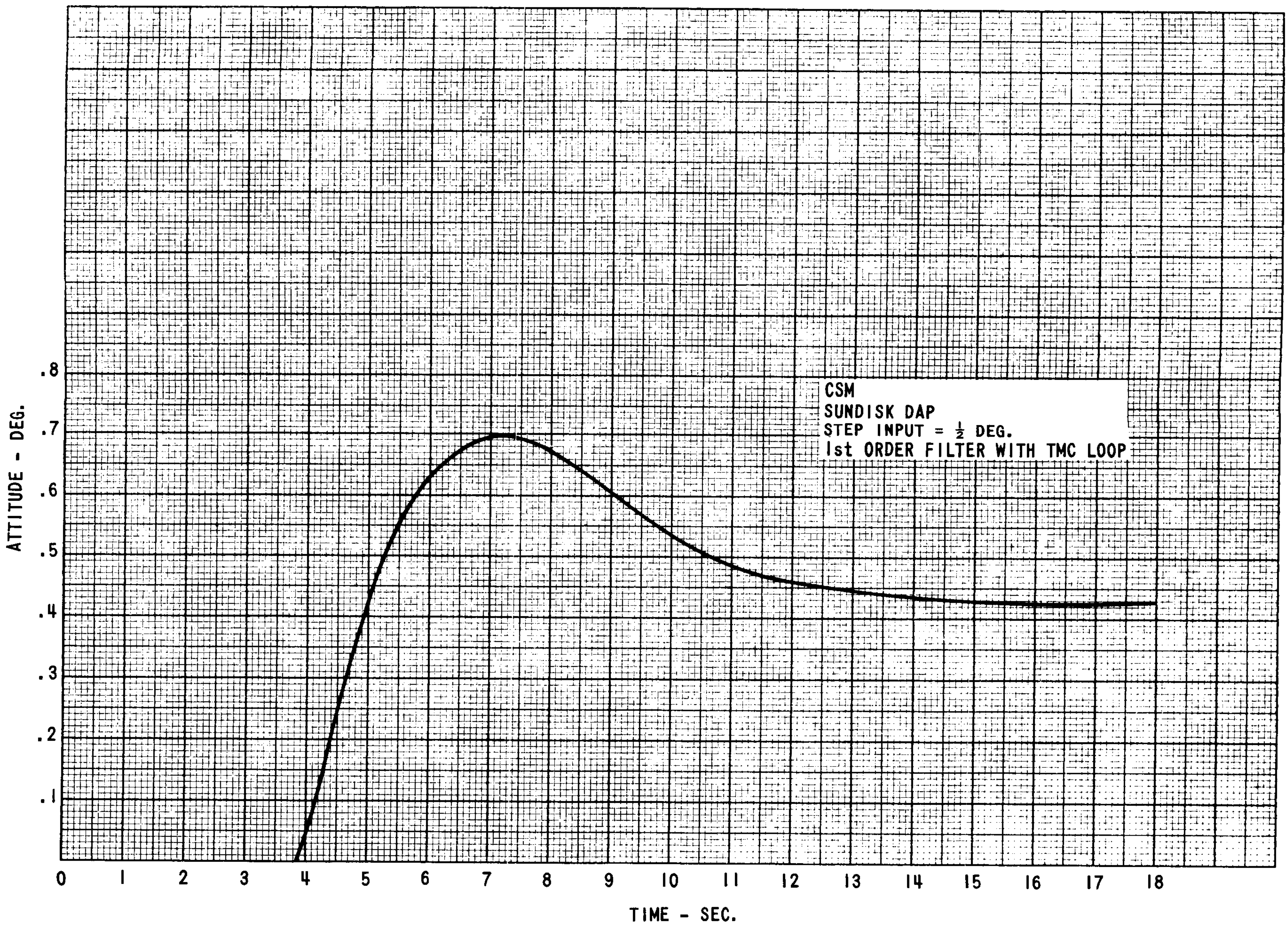

CSM
SUNDISK DAP
STEP INPUT = ½ DEG.
1st ORDER FILTER WITH TMC LOOP
ATTITUDE - DEG.
.8
.7
.6
.5
.4
.3
.2
.1
0
1
2
3
4
5
6
7
8
9
10
11
12
13
14
15
16
17
18
TIME - SEC.

FIGURE 1

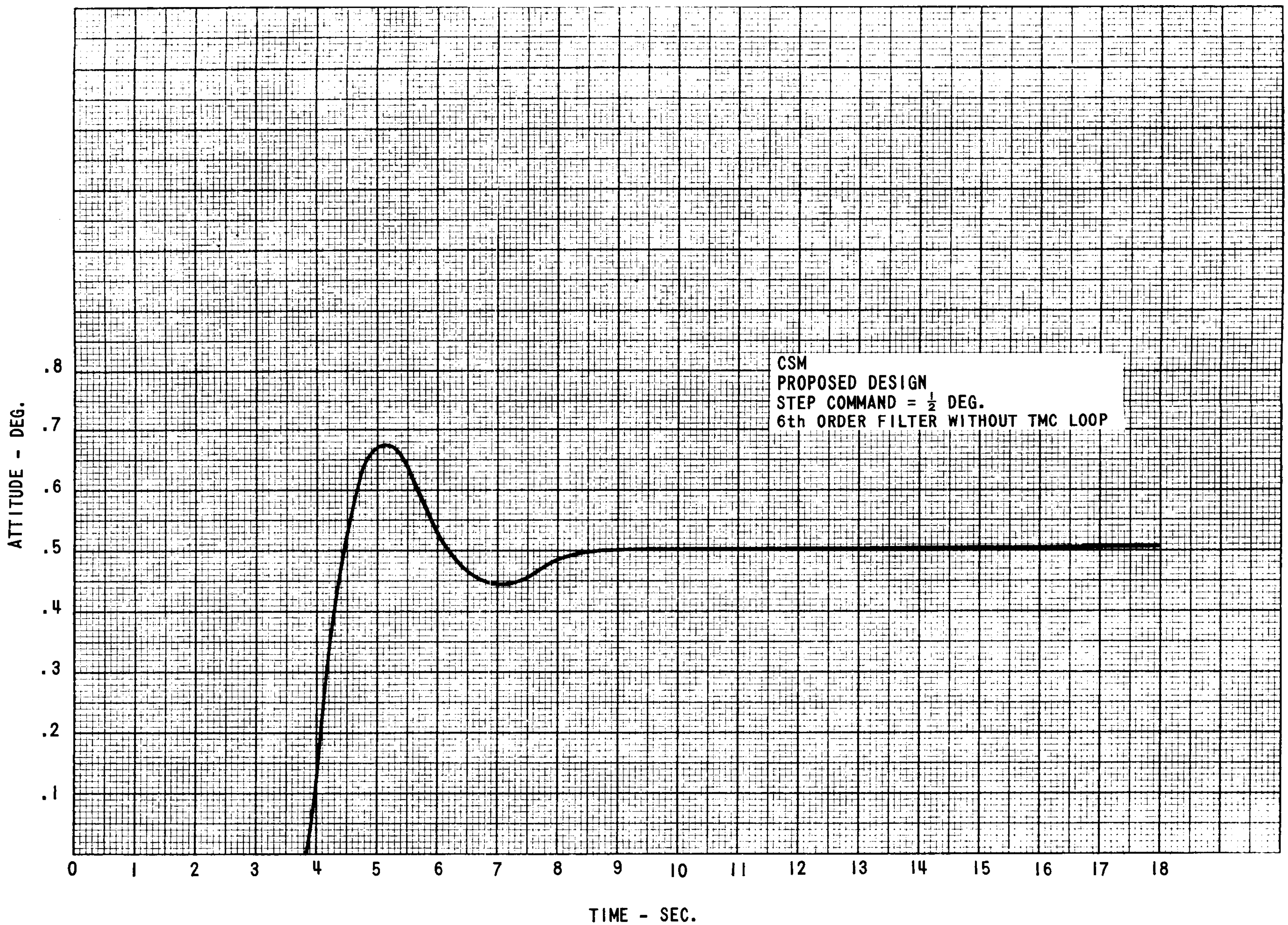

CSM
PROPOSED DESIGN
STEP COMMAND = ½ DEG.
6th ORDER FILTER WITHOUT TMC LOOP
ATTITUDE - DEG.
.8
.7
.6
.5
.4
.3
.2
.1
0
1
2
3
4
5
6
7
8
9
10
11
12
13
14
15
16
17
18
TIME - SEC.

FIGURE 2

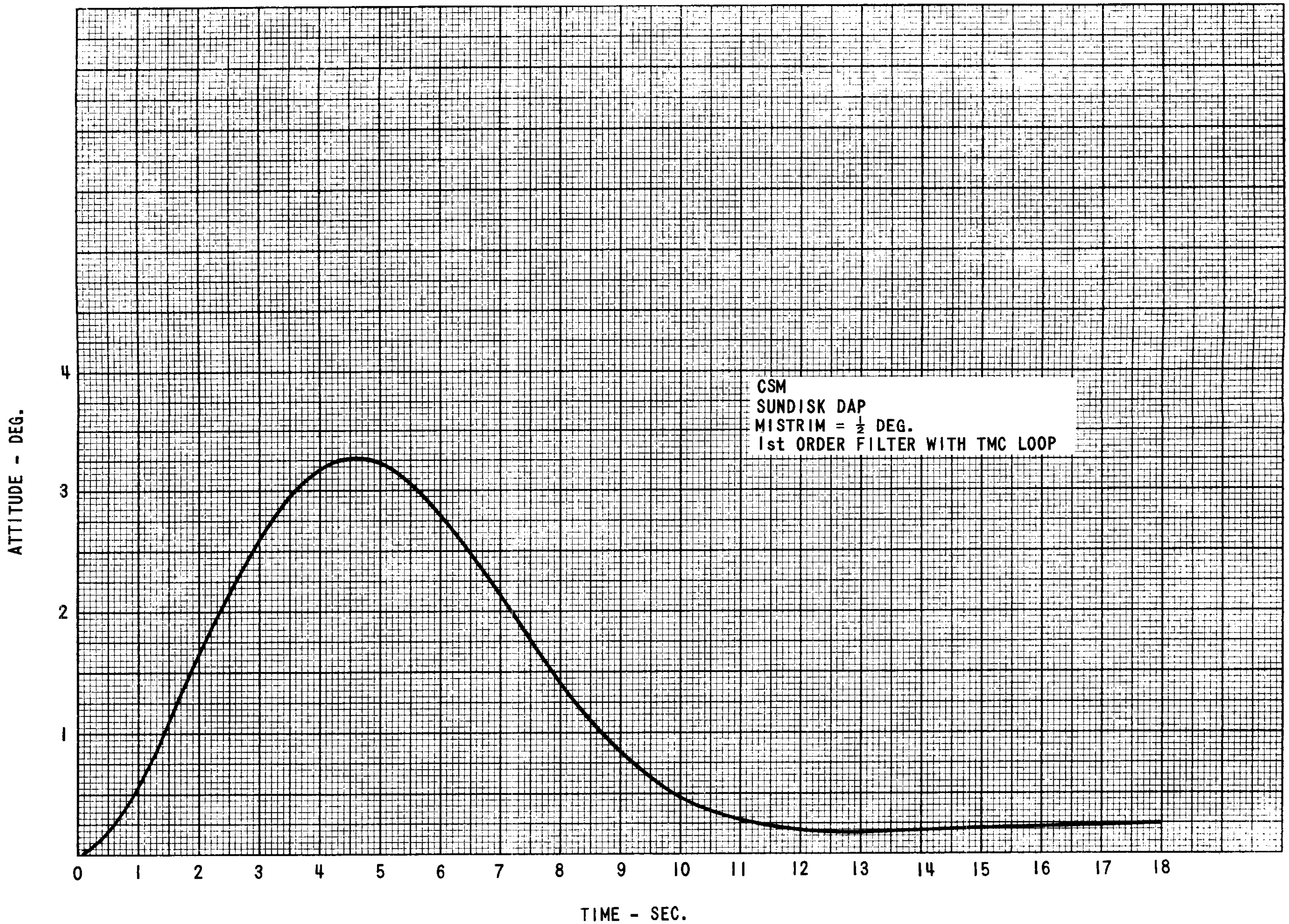

CSM
SUNDISK DAP
MISTRIM = ½ DEG.
1st ORDER FILTER WITH TMC LOOP
ATTITUDE - DEG.
1
2
3
4
0
1
2
3
4
5
6
7
8
9
10
11
12
13
14
15
16
17
18
TIME - SEC.

FIGURE 3

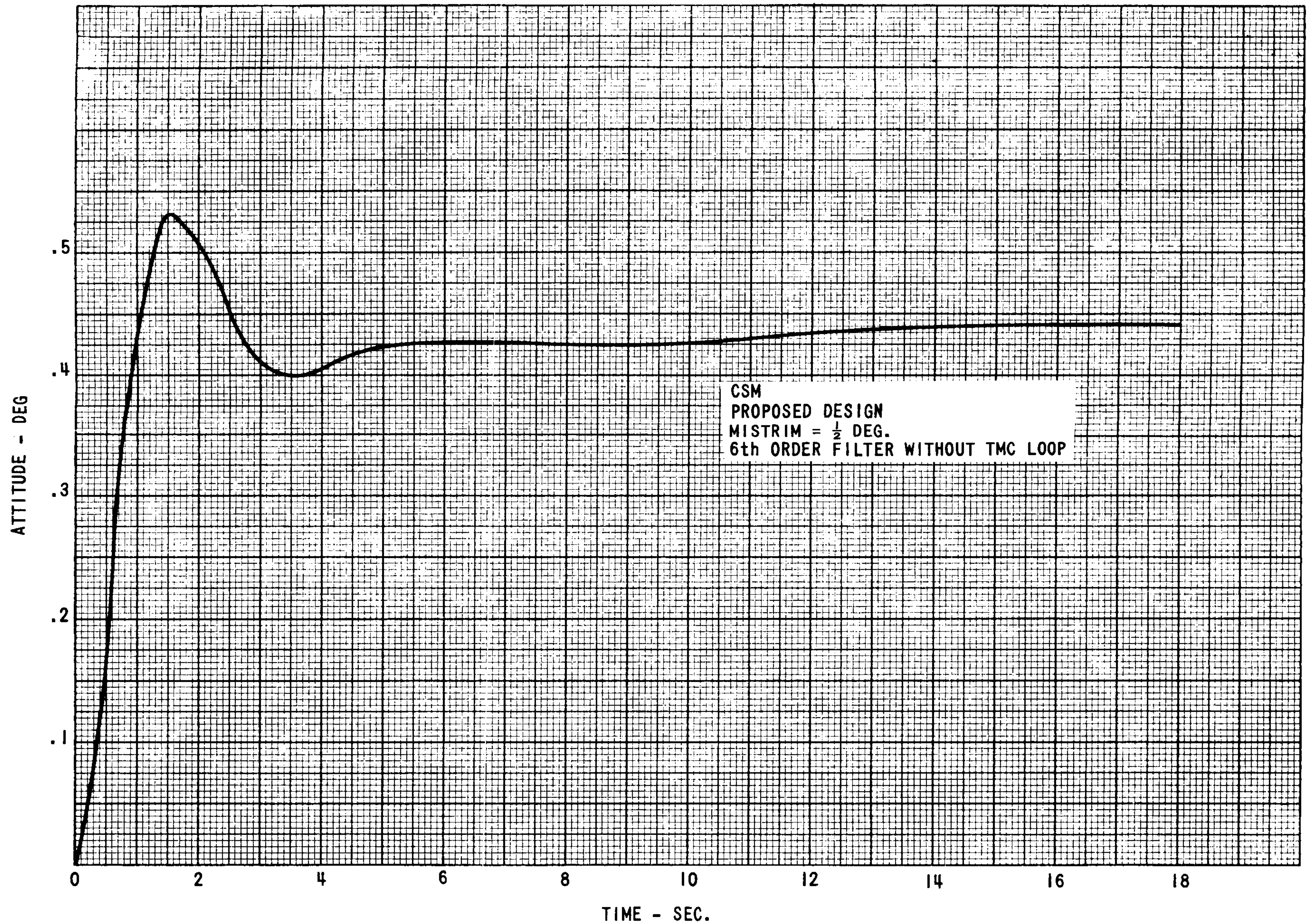

CSM
PROPOSED DESIGN
MISTRIM = ½ DEG.
6th ORDER FILTER WITHOUT TMC LOOP
ATTITUDE - DEG
.1
.2
.3
.4
.5
0
2
4
6
8
10
12
14
16
18
TIME - SEC.

FIGURE 4

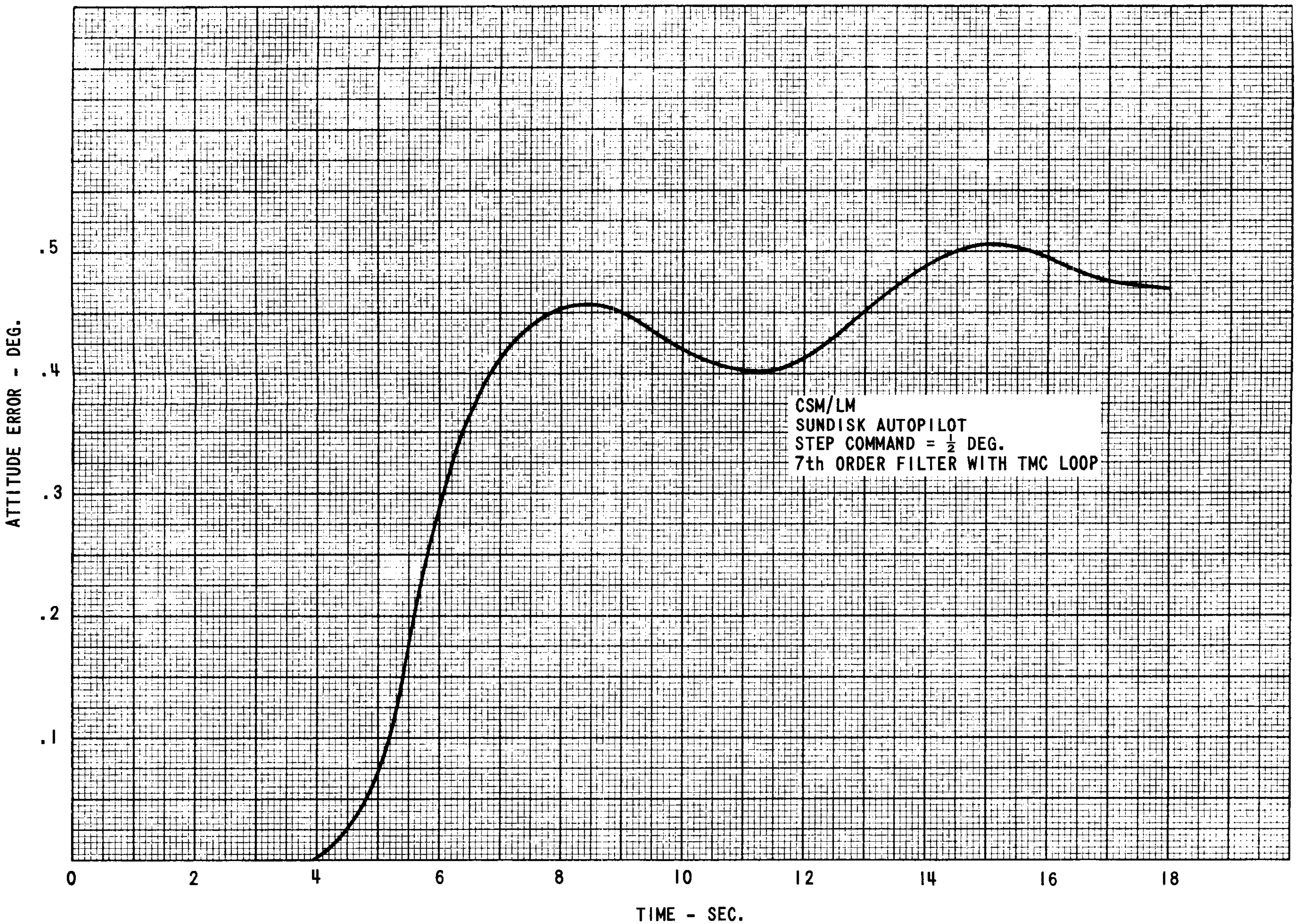

CSM/LM
SUNDISK AUTOPILOT
STEP COMMAND = 1/2 DEG.
7th ORDER FILTER WITH TMC LOOP
ATTITUDE ERROR - DEG.
.5
.4
.3
.2
.1
0
2
4
6
8
10
12
14
16
18
TIME - SEC.

FIGURE 5

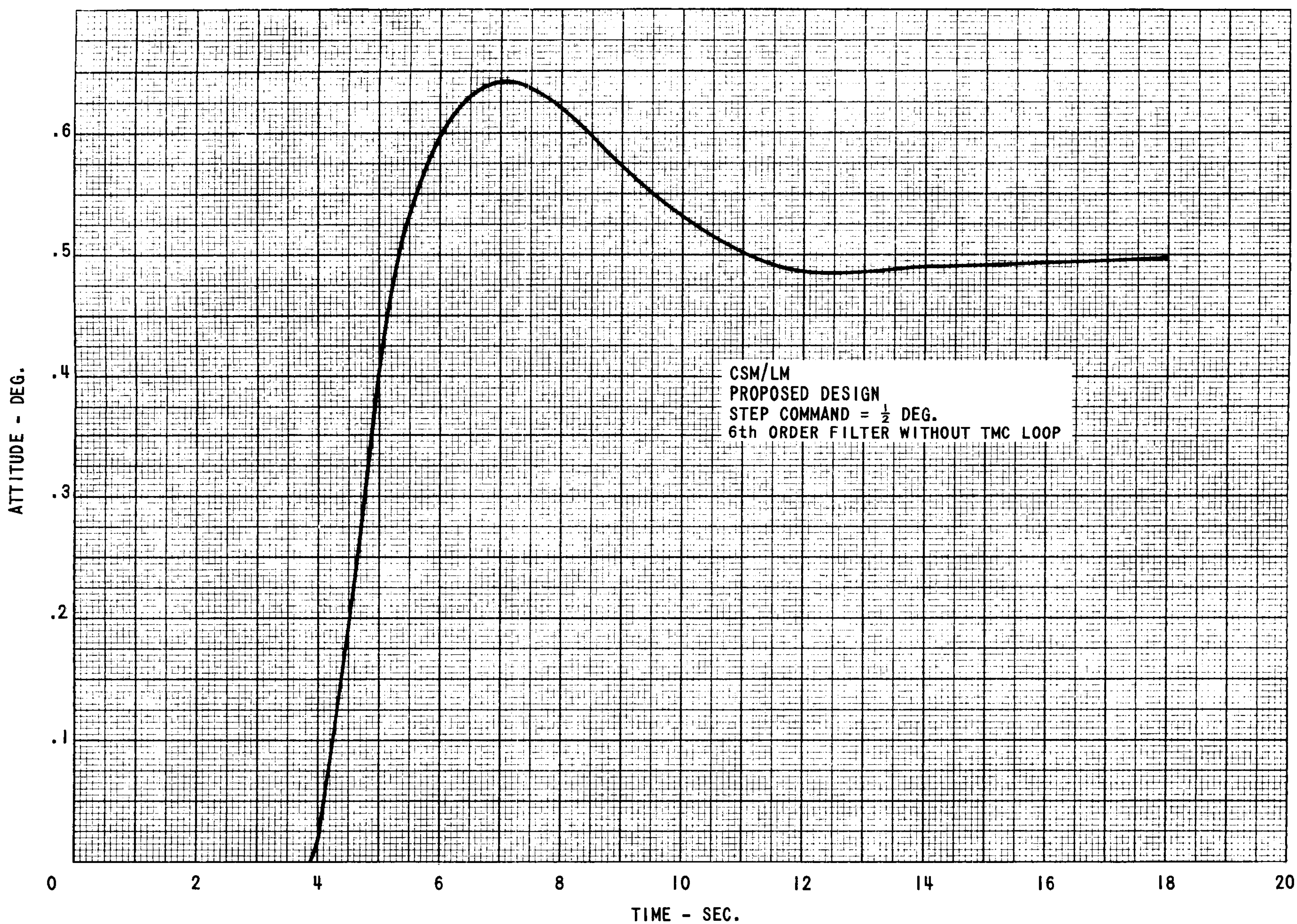

CSM/LM
PROPOSED DESIGN
STEP COMMAND = ½ DEG.
6th ORDER FILTER WITHOUT TMC LOOP
ATTITUDE - DEG.
.6
.5
.4
.3
.2
.1
0
2
4
6
8
10
12
14
16
18
20
TIME - SEC.

FIGURE 6

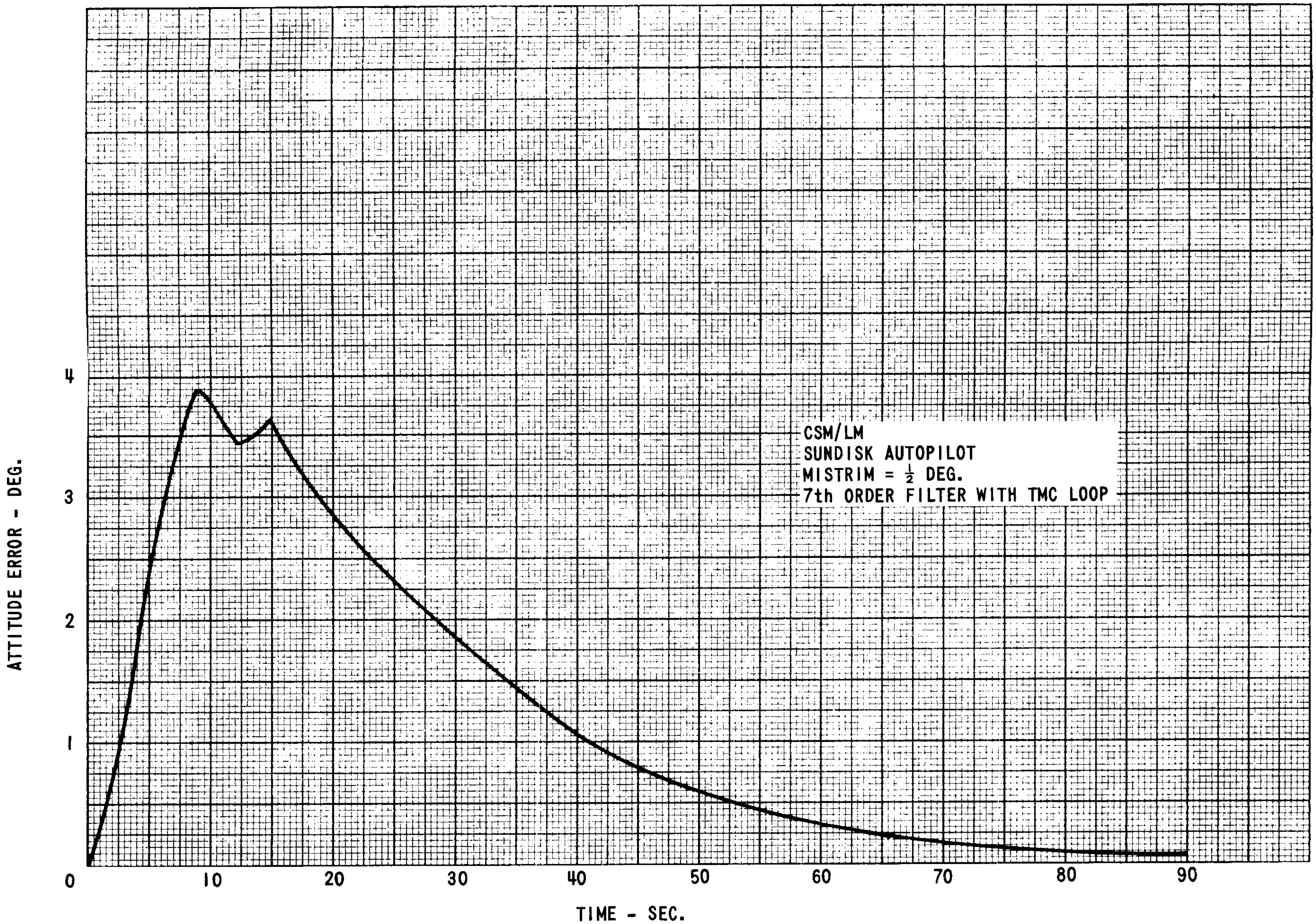
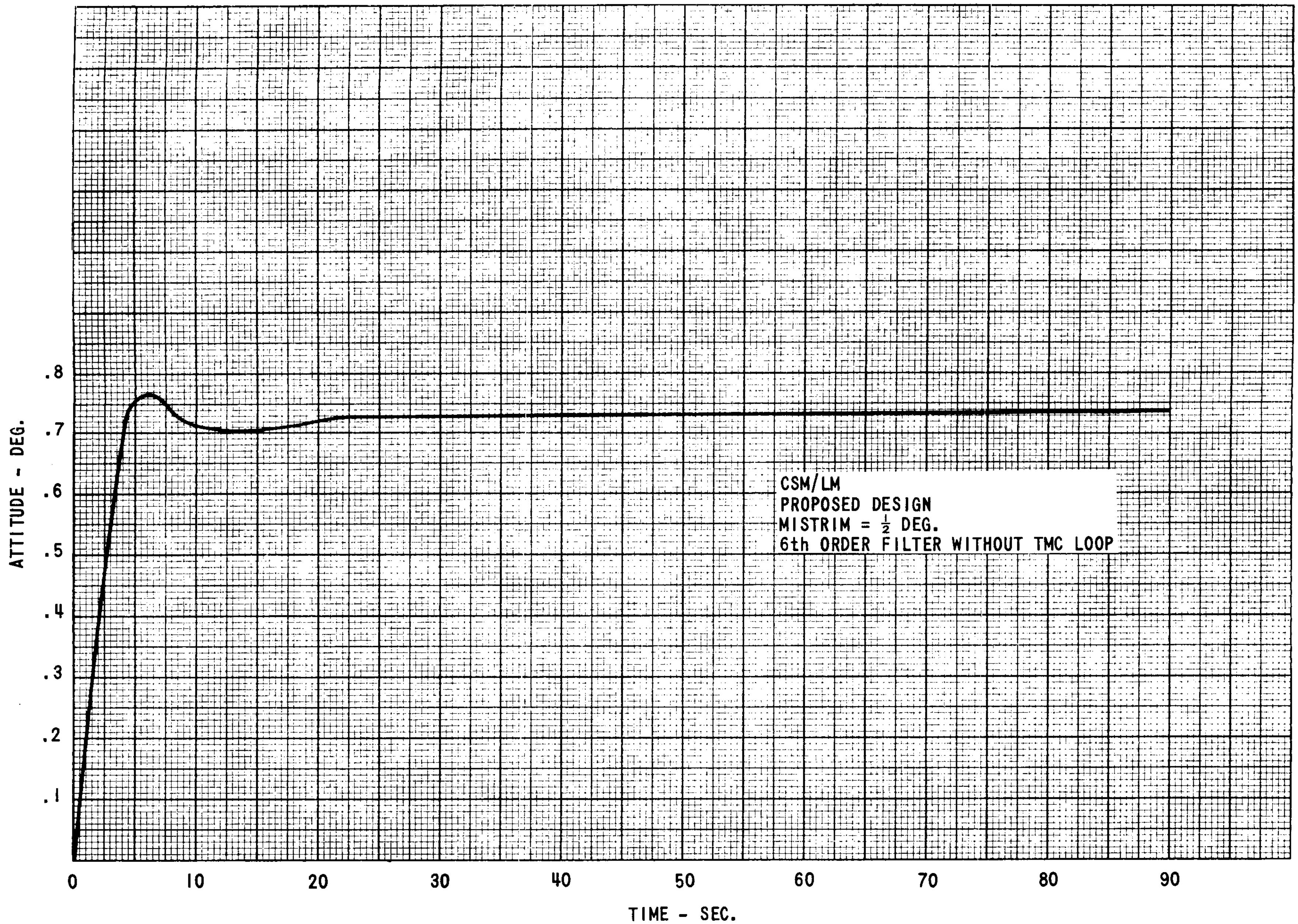

CSM/LM
SUNDISK AUTOPILOT
MISTRIM = ½ DEG.
7th ORDER FILTER WITH TMC LOOP
ATTITUDE ERROR - DEG.
0
1
2
3
4
10
20
30
40
50
60
70
80
90
TIME - SEC.

FIGURE 7



.8
.7
.6
.5
.4
.3
.2
.1
ATTITUDE - DEG.
0
10
20
30
40
50
60
70
80
90
TIME - SEC.
CSM/LM
PROPOSED DESIGN
MISTRIM = $\frac{1}{2}$ DEG.
6th ORDER FILTER WITHOUT TMC LOOP

FIGURE 8

BELLCOMM, INC.

Subject: Proposed Simplification of the CSM-Digital Autopilots
Case 310

From: A. Heiber
F. La Piana

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ATTACHMENT 8

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Simplified LM Digital Autopilot
Case 310

DATE: March 29, 1968

FROM: E. A. Nussbaumer

ABSTRACT

This memorandum discusses a simplified LM Digital Autopilot (DAP) and compares it to the present MIT Autopilot. The simplified DAP requires less computer storage and less computational time. It performs as well except for a slight RCS fuel penalty for nominal operation. For high noise environment, highly off-nominal conditions, and undetected failures the simplified DAP is superior.

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Simplified LM Digital Autopilot
Case 310

DATE: March 29, 1968

FROM: E. A. Nussbaumer

MEMORANDUM FOR FILE

Introduction

As part of the Bellcomm task to study the feasibility of programming the LM guidance computer with one half the present memory, a simplified LM digital autopilot design is discussed. This design, designated the LAD-DAP (LM ascent-descent digital autopilot) is the result of several years work in evaluation of LM autopilot designs as part of Bellcomm's guidance and control validations. Both the MIT/DAP and the LAD/DAP have been simulated at Bellcomm.

In designing the LAD-DAP, all emphasis was placed upon simplicity rather than upon achieving a fuel optimum system. However, simulation results have shown that the penalty in fuel consumption is acceptable for nominal conditions. For off-nominal and failure conditions, the LAD-DAP performed considerably better than the MIT-DAP and a fuel saving was obtained.

Philosophy

1. RCS Autopilot:

The current implementation of the LM autopilot uses a programmed dynamic model of the LM vehicle and a phase-plane algorithm to command jet firings and engine gimbaling. Parameter used in the dynamic model include moments of inertia, torque output of the RCS jets, location of the center of gravity, and thrust level and lever arm of the DPS engine. Those parameters vary during the mission and have to be stored or periodically updated. Parameter storage, prediction algorithm and recursive filter technique require considerable computational time, and memory. In addition, the performance degrades when undetected changes in the model occur, such as degraded performance, jet or gimbal actuator failures.

The LAD-DAP bases its decisions for firing jets and stepping engine gimbal solely on the present and past state of the vehicle. The approach is similar to an analog pulse ratio modulator. However, the digital technique provides a definite advantage over the analog implementation, gain changes can be easily implemented and a more adaptive technique can be applied by evaluating the present against the past state.

2. DPS Trim Gimbal Law:

During powered descent the present DPS trim gimbal law tries to null in minimum time attitude error, vehicle rate and off-set acceleration. The employed time-optimal bang-bang control law requires to solve a set of complicated equations. The procedure is costly in computation time and memory requirement. The question to be asked here: Does the fuel saving really justify the high price in complexity? It was found that the time-optimal controller works well as long as the state of the vehicle is closely known. However, the trim gimbal actuator was never designed to have a controlling function. The actuator is very slow (.2°/sec.) its function was meant to track the slowly moving center of gravity.

The LAD-DAP does not try to control the vehicle with the trim gimbal actuator; it merely regulates the thrust vector through the center of gravity. RCS firings are necessary to control the vehicle attitude, however, the fuel expenditure was small. In any case, during the visibility phase, when radar data is being assimilated, considerable RCS activity is occurring. During that phase, any attempt to steer the spacecraft with the DPS trim gimbal is of marginal value.

<u>Computer Memory and Real Time Requirement</u>

The LAD-DAP function have been programmed in machine language by Mr. J. M. Nervik of the Systems Programming Group to obtain estimates of the amount of computer time and memory requirement.

The autopilot functions are organized into three computation loops. The inner loop, at 5 (descent) or 10 (ascent) iterations per second contains the control functions of the yaw, pitch and roll channels. The middle loop at 5 iterations per second contains the interrogation of the hand controller and display requirements. The outer loop at 1 iteration per second contains the evaluation of the trigonometric functions of the IMU gimbal angles. In addition to these three loops computer interrupts and computations are needed to turn on and off reaction jet between computational intervals.

A summary of permanent memory is given in the following table:

Inner Loop	1196 words
Middle Loop	116
Outer Loop	151
Jet Interrupts	39
Parameter Changing	102
TOTAL	1604 words

About 125 words of temporary memory are required.

136 ms/sec of computer time is required to execute the atuopilot programs during LM ascent. For LM descent 83 ms/sec have been estimated.

Performance

No reduction in performance could be found for the RCS autopilot. However, for nominal conditions the LAD-DAP uses more jet firings, to achieve the same maneuver, but converged more rapidly into a minimum impulse limit cycle. The increased pulsing causes a slight fuel penalty for maneuvering but this is partially compensated by the more rapid convergence into a limit cycle.

For slightly off-nominal conditions both autopilots showed equivalent maneuver performance.

For a high-noise environment, for highly off-nominal conditions, and for undetected failures, the LAD-DAP performed considerably better in regard to fuel, pulsing, and attitude control quality.

During powered descent flight the DPS trim gimbal showed good regulating capability. A capability reduction was found for the DPS engine start. Under a high-noise environment the regulator produced a slower recovery from a large initial c.g. offset (5°) than the time-optimal MIT-DAP.

E. A. Nussbaumer

2012-EAN-ek E. A. Nussbaumer

BELLCOMM, INC.

Subject: Simplified LM Digital Autopilot - Case 310

From: E. A. Nussbaumer

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ATTACHMENT 9

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: LM Descent Guidance for a Feasibility Study to Simplify Apollo Guidance
Case 310

DATE: March 27, 1968

FROM: G. L. Bush

ABSTRACT

This memorandum suggests how LM descent powered flight guidance can be simplified in the interest of saving computer storage locations. An estimate of the possible word savings is made. Conclusions are reached as to the impact of the simplicity upon ΔV budget, accuracy, and guidance capability.

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: LM Descent Guidance for a Feasibility Study to Simplify Apollo Guidance Case 310

DATE: March 27, 1968

FROM: G. L. Bush

MEMORANDUM FOR FILE

Introduction

As part of a Bellcomm study of the feasibility of reducing Apollo guidance computer (CMC, LGC) memory capacity by a factor of one-half, simplified lunar descent guidance and navigation was examined. All aspects of the Apollo lunar landing flight were examined to find simpler methods in terms of required computer memory to perform guidance, navigation, and control functions. This memorandum describes a simplified way to perform LM descent guidance and navigation.

It was decided that the basic LM descent trajectory design should not be changed. A braking phase is desirable to reduce the LM's horizontal velocity efficiently and safely. An approach phase is desirable to allow the crew to visually assess their landing site. Ignition altitude and velocity and high-gate altitude should remain essentially as they are in present LM descent plans.

Those aspects of the guidance plan which could be simplified are discussed next.

Simplification

One way to greatly simplify LM descent guidance would be to fly the vehicle open loop. A pitch polynomial for the braking phase can be found which will properly control the vehicle's thrust to fly the LM to a desired position and velocity at high-gate. Another pitch polynomial for the approach phase can be found to control the LM's trajectory to lo-gate. The problem with flying the vehicle open loop, however, is that trajectory errors and system uncertainties cannot be simply reduced in flight. Errors at ignition, thrust uncertainties, and thrust pointing errors all add and continue to grow if the vehicle is flown open loop. The vehicle's state (position and velocity) could not be used automatically to produce guidance commands, and it is difficult to conceive of a plan whereby the crew could maintain trajectory accuracy by manually adjusting vehicle thrust when off-nominal performance was recognized.

It is possible to simplify LM descent guidance while maintaining closed-loop guidance. The guidance is closed loop in that it computes periodically, during the flight, thrust acceleration commands as a function of desired values of state as well as the vehicle's present state. Off-nominal performance is measured through the estimation of the present state and periodically (every two seconds) new thrust commands are computed to fly the vehicle on a trajectory to the desired state. The guidance system can be simplified in the following ways.

1. Compute Ignition Time and Initial Attitude on Ground

   The computation of T_{IG} and ignition attitude are computed both on-board the LM and by RTTC on the ground. Since there exists a communication link between the ground and the LM, this computation can be eliminated from the LGC (LM Guidance Computer).

2. Provide Limited Throttling in High-Thrust Portion of Braking Phase

   Providing limited throttling in the high thrust portion of the braking phase could overcome problems associated with engine thrust uncertainties. In particular, a ΔV savings would be realized probably sufficient to offset ΔV penalties of manual operation and loss of velocity updating as suggested in 7 and 8.

3. Eliminate Automatic Landing Site Redesignation (LPD)

   In the interest of saving computer storage locations, the LPD is an obvious victim. The manual final landing phase would serve to avoid small rocks and craters. A simple yaw control during the visibility phase would allow the astronauts to avoid large hinderances in their landing path by flying sufficiently laterally to avoid them.

4. Use Linear Guidance

   Present plans(1) use a quadratic guidance law which controls not only end-of-phase position and velocity, but also acceleration which is required to insure good trajectories with LPD operation. In the interest of simplicity use of the linear guidance law originally suggested by MIT to control terminal position and velocity can be used since the LPD is being eliminated.

5. Eliminate Guidance Coordinate Frame

Elimination of automatic landing site redesignation obviates the guidance coordinate frame. Present plans use this frame as a rotating coordinate system, with its x-axis pointing continuously through the landing site; if the landing site is changed through astronaut use of the LPD, the coordinate frame is changed. Guidance commands are computed in this coordinate frame and are transformed to inertial (platform) coordinates for actual thrusting. It is suggested that this scheme be eliminated and that guidance commands be computed in the inertial coordinate frame.

6. Compute Time-to-Go On the Basis of Elapsed Time

Present LM guidance computes phase time-to-go as the solution of a cubic equation of which terminal down-range derivative of acceleration is one parameter. Computing time-to-go in this fashion provides the advantage of controlling vehicle body rates at the end of the phase, and it is essential in the automatic redesignation scheme used with the LPD. Since it has been suggested that the LPD be eliminated and simplicity is the goal, an alternate method of computing T_{GO} is to precompute a final phase time t_f, and compute $T_{go} = (t-t_f)$.

7. Update Only the Altitude Component of State Vector

The updating equations for altitude and velocity are very costly in terms of required computer storage locations. Studies have shown[(2)] that the LM can achieve high-gate with sufficient accuracy by updating only altitude information from the landing radar. It is suggested that velocity updating be eliminated and that the LM's pilot reduce velocity errors manually in the final landing phase. Expected velocity errors which would result are shown in Figure 1.

8. Use Manual Final Landing Phase

Present plans target the visibility phase to an altitude approximately 100 feet above the landing site. Vertical descent is then possible through the crew's option of an automatic descent mode, a rate-of-descent mode, or manual control. It is suggested that the visibility phase be targeted to a lo-gate point uprange from the landing site (range-to-go = 1200 ft., alt. = 500 ft.) whereupon the astronaut would manually control engine throttle and vehicle attitude to land the vehicle.

Estimate of Word Savings

Estimates of the word savings possible were made assuming that the above suggestions were implemented. Estimates were based on a Sunburst program listing and Sundance budgets, so they should be considered as only gross estimates. Table I shows the estimate of savings that could be made from LM descent guidance programs. The estimate of savings possible is 640 words. In addition, it is estimated that 150 words could be saved from radar subroutines if landing radar velocity updates were eliminated.

Conclusions

It is anticipated that there would be little ΔV penality to the LM descent ΔV budget as a result of the simplification if limited throttling in the high-thrust portion of the braking phase were implemented. If throttling were not implemented a small ΔV penalty would result because of the proposed longer manual operation of the LM in final landing.

It is believed that no reduction in visibility during the approach phase, nor any increase in the landing ellipse size would result as a result of the simplification. An increase in the final landing velocity error would result, but the crew could null those errors manually in the final landing phase.

This proposed simplification would eliminate the automatic landing site redesignation capability.

The suggestions for simplificaiton presented here have not been thoroughly verified.

2012-GLB-ek

G. L. Bush

Attachments
- Figure 1
- Table I
- References

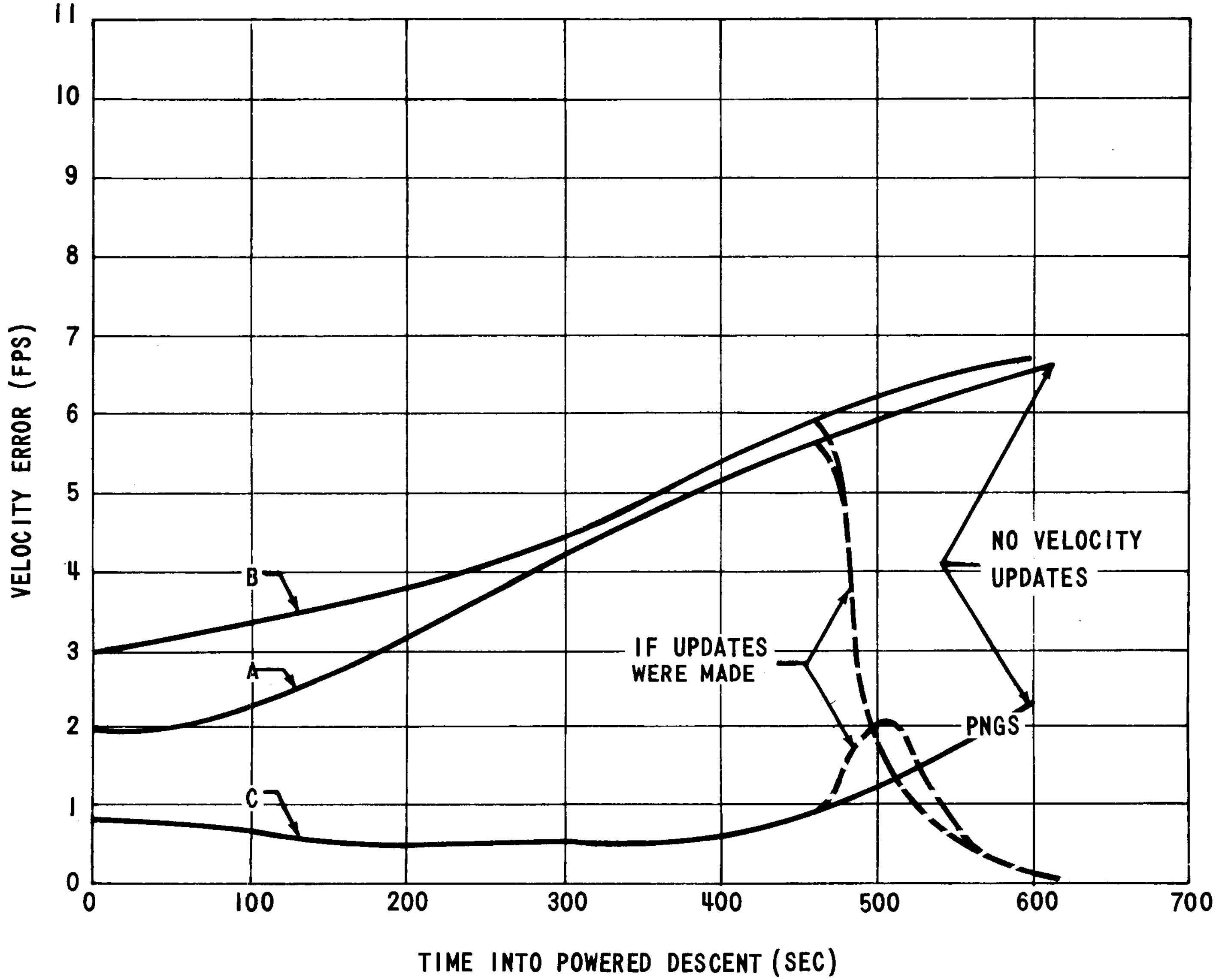

FIGURE 1 - A - RMS RADIAL VELOCITY ERROR DURING POWERED DESCENT
B - RMS NORMAL VELOCITY ERROR DURING POWERED DESCENT
C - RMS TANGENTIAL VELOCITY ERROR DURING POWERED DESCENT

TABLE I

Estimate of Savings Computer Words Possible by Using

Simplified LM Descent Guidance

Program Function	Word Budget (Sundance)	Estimate of Savings Possible
1. Control		
Landing Braking (P63 + R11, R13)	200	20
Landing Visibility (P64)	150	20
Landing Automatic (P65)	100	100
Landing ROD (P66)	100	100
Landing (Manual) (P67)	100	0
2. Ignition Computations and Initilization	72	45
3. Overhead	173	16
4. Attitude Control Computations	193	20
5. Guidance		
Quadratic	130	130
Linear	41	0
T_{go} Calculation	146	129
6. Estimate State Vector		
IMU Calculations	90	0
Update Altitude Component	20	0
Update Velocity Components	60	60
TOTAL	1575	640

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BELLCOMM, INC.

Subject: LM Descent Guidance for a Feasibility Study to Simplify Apollo Guidance - Case 310

From: G. L. Bush

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BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: An Open Loop Crew-Monitored LM Descent - Case 310

DATE: March 26, 1968

FROM: F. Heap

ABSTRACT

As part of a study to determine the feasibility of reducing the Apollo guidance computer memory to half its present size, a simple open loop LM descent scheme was postulated and examined. The scheme is considered minimal in terms of capability and safety; it would require much increased crew and RTCC participation. It has utility, however, in providing an example against which the relative superiority and memory cost of a simplified explicit guidance closed loop system could be measured.

A trajectory was designed to follow the scheme; final approach and landing constraints could be met, considering 3σ altitude errors and reasonable flight path errors at Hi Gate. The ΔV cost above the current budget could be about 190 ft/second.

Many factors which might qualify the results were not considered. No estimate of the computer word count was made.

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: An Open Loop Crew-Monitored LM Descent - Case 310

DATE: March 26, 1968

FROM: F. Heap

MEMORANDUM FOR FILE

INTRODUCTION

As part of the study to determine the feasibility of reducing the Apollo guidance computer memory to half its present size (Reference 1), a simple open loop LM descent scheme was postulated and examined. The scheme is considered to be minimal in terms of capability and safety; it would require much increased crew and RTCC participation. Because of these factors, it may be found wanting on further investigation. It has utility, however, in providing an example against which the relative superiority of a simplified explicit guidance closed loop system can be measured.

BASIC SCHEME

Under the simplified open loop scheme, the LM is inserted into the Hohmann descent transfer orbit by firing the descent engine to apply the necessary ΔV in a fixed vehicle attitude. The ΔV, attitude, and time of ignition are computed by the RTCC, based on the lunar parking orbit elements and the position of the landing site.

The ignition point for powered descent is at a fixed range (central angle) from the landing site. There is a nominal ignition time based on a pre-targeted reference trajectory. The RTCC evaluates the transfer orbit, based on tracking and insertion telemetry, and updates the ignition time and the powered descent parameters based on a real-time retargeting, if necessary.

The braking phase is at near-full throttle thrust to Hi Gate with constant pitch rate from an initial pitch attitude. Minor ($\pm 3\%$) throttle capability is required to adjust the thrust to the nominal, in face of thrust errors. (In the current descent, the engine operates only at full throttle or below 58% thrust. Recent investigations by MSC have explored the feasibility of adjusting the throttle to eliminate the expected ± 2-1/2% thrust error.) This throttle capability requires a simple regulator function in the computer.

Neglecting for the moment downrange and crossrange errors, which will be addressed later, the major uncertainty at

ignition is altitude, which propagates to an altitude error and second order flight path error at Hi Gate. The criterion for Hi Gate is a fixed total velocity magnitude. The crew monitors the landing radar output and estimates the altitude and flight path errors. The altitude error at Hi Gate is currently estimated to be 5500 feet (3σ) based on previous MSC studies where landing radar updates were suppressed. For this study, the flight path angle error has been assumed to be not more than $\pm 1°$. During the last minute of the braking phase, the computer uses a simple algorithm to determine time to go to Hi Gate and to estimate the altitude and flight path angle at Hi Gate, using inertial data corrected by the crew's estimate of the difference between inertial data and radar data. To eliminate smoothing and data handling requirements in the computer, actual radar data is not used.

At Hi Gate the computer determines and commands the final approach phase pitch angle and thrust acceleration. The final approach phase has a constant flight path angle, with constant pitch attitude, constant thrust acceleration and essentially constant look angle. Figure 1a shows the profile of the final approach phase schematically. Because altitude errors are not eliminated by radar updates, altitude errors propagate into downrange errors as shown, increasing the landing ellipse downrange dimension from the currently estimated downrange dispersion of 25,000 feet to 37,500 feet. Crossrange error is not affected. The phase has nominal values of the control constants, pitch angle and thrust acceleration, to give a look angle (angle from the negative x body axis to the line of sight to the landing site) of say 40 degrees (site 15 degrees above the window bottom). The effect of errors in altitude and flight path angle at Hi Gate is to modify the control constants. The nominal trajectory is chosen so that, even with maximum errors, thrust remains less than maximum throttleable thrust and look angle is more than 25 degrees.

Lo Gate is chosen to be at altitude 500 feet, with horizontal and vertical velocities within the crew requirement range. The landing phase is flown manually.

FINAL APPROACH PHASE DYNAMICS

Using a flat moon approximation, the equations of motion of the LM in the vertical plane are

$$\dot{V} = -\frac{T}{M}\sin(\theta-\gamma) - g\sin\gamma \tag{1}$$

$$V\dot{\gamma} = \frac{T}{M}\cos(\theta-\gamma) - g\cos\gamma \tag{2}$$

where the symbols are as illustrated in Figure 1b.

For a constant flight path, γ = constant, $\dot{\gamma} = 0$, so Equation (2) yields

$$\frac{T}{M} = \frac{g \cos\gamma}{\cos(\theta-\gamma)} \tag{3}$$

$\theta-\gamma$ is the complement of the look angle, so constant θ gives constant look angle and thrust acceleration.

Substituting into (1),

$$\dot{V} = g\,(-\cos\gamma \tan(\theta-\gamma) - \sin\gamma). \tag{4}$$

For constant pitch angle and flight path angle, $\dot{V}$ is also constant. Denoting S as distance along the flight path and h as altitude, $\dot{V}$ can be written as

$$\dot{V} = \frac{VdV}{dS} = \frac{VdV}{dh} \sin\gamma \tag{5}$$

then

$$g(-\cos\gamma \tan(\theta-\gamma) - \sin\gamma)\,dh = VdV \sin\gamma \tag{6}$$

Integrating from h_o, V_o (Hi Gate) to h_D, V_D (Lo Gate) gives

$$g(-\cos\gamma \tan(\theta-\gamma) - \sin\gamma)\,(h_o - h_D) = \tfrac{1}{2}(V_o^2 - V_D^2)\sin\gamma \tag{7}$$

$$\theta-\gamma = \arctan\left\{-\tan\gamma(1 + (V_o^2 - V_D^2)/2g(h_o-h_D))\right\} \tag{8}$$

Equations (8) and (3) yield the controls θ and T/M. given the initial Hi Gate conditions (V_o, h_o, γ) and the desired Lo Gate conditions (V_D, h_D). For the computer, they can be written as

$$\theta = \gamma + \arctan\left\{-\tan\gamma(1 + k_1/(h_o-k_2))\right\} \tag{9}$$

$$T/M = k_3 \cos\gamma/\cos(\theta-\gamma) \tag{10}$$

where the k's are fixed computer constants.

TRAJECTORY SIMULATION

The validity of the postulated scheme and its capability to perform a landing within certain mandatory constraints were tested by conducting two simulations. The constraints are:

1. The line of sight to the landing site should be above the window bottom during the final approach phase. This means that the look angle, $90 - (\theta - \gamma)$, should be greater than 25 degrees approximately, as the bottom of the window is 25 degrees from the negative T/M axis.

2. The descent engine thrust should be less than 58% (6090 lbs.) during the final approach phase. This means that T/M should be less than 10.5 ft/sec^2 approximately (assuming LM weight of 18,500 lbs. in the final approach phase).

3. The Lo Gate conditions should be within the range to allow for a switch to manual control for the landing phase. For a Lo Gate altitude of 500 feet, the vertical velocity should be less than 17 ft/sec., and the horizontal velocity should be less than 60 ft/sec. (Reference 2)

For the first simulation, the flight path angle (γ) was chosen at -15 degrees, giving a Lo Gate velocity (V_D) of 65.5 ft/second (equal to 17/sin 15°). The nominal look angle was chosen to be 40 degrees ($\theta - \gamma = 50°$). Hi Gate altitude (h_o) was chosen to be 9500 feet, similar to that in the current reference trajectory (Reference 3). Hi Gate velocity (V_o) was found, by inverting Equation 8, to be 577 ft/sec. Equations (8) and (3) were then used to determine the variation in look angle and thrust acceleration required for 3σ off-nominal altitudes ($\pm 1°$) at Hi Gate. The results showed that for an off-nominal descent in which the Hi Gate altitude was 3σ low at 4000 feet, the resulting line of sight to the landing site in the final approach phase was below the window bottom (look angle <25 degrees) and the thrust acceleration required a thrust in excess of 6090 lbs. A BCMASP (Bellcomm Apollo Simulation Program) targeting run indicated that, for a 3σ high altitude error at Hi Gate, the ΔV cost was about 300 feet per second greater than that for a no-error trajectory.

For the second trajectory, the Hi Gate altitude was increased to 12,500 feet (intuitively) to correct the off-nominal look angle and thrust acceleration excesses. At the same time the flight path angle was changed (intuitively) to -20 degrees to maintain a tolerable increase in ΔV cost for the 3σ high altitude case. Lo Gate velocity was 46.8 ft/sec. (= 17/sin 20°);

Hi Gate velocity was 540 ft/sec. Nominal look angle was kept at 40 degrees. The results are shown in Figure 2. The look angle remains above the window bottom for all off-nominal altitudes and flight path angles. The thrust acceleration only exceeds the 6090 lb. thrust level at the lowest extreme of the expected altitude range. (It is probable that if necessary this constraint could be eased slightly.) The maximum ΔV penalty is 295 ft/sec. above the no-error case.

Figure 3 is a ΔV summary, comparing the ΔV requirements for this descent with those in the standard ΔV budget (Reference 4). The total descent budget for the latter is 7180 ft/sec. Because of the arbitrary way in which the conditions for the postulated descent were chosen, it is possible that the 190 ft/sec. ΔV penalty could be lowered by a judicious choice of Hi Gate altitude and flight path angle.

LIMITATIONS

This study was performed without considering many factors which could qualify the results. Among these are digital autopilot accurary, RCS usage to achieve the constant pitch rate, ΔV required to correct for residual altitude and velocity errors at Lo Gate, crew capability to monitor the descent and assess the performance, increased landing site dimension, aborts from the descent, and safety. No estimate of the computer word count was made.

SUMMARY

A simple open loop LM descent scheme was postulated and examined. A trajectory was designed to follow the scheme; final approach and landing phase constraints could be met, even if 3σ altitude errors and 1° flight path angle errors at Hi Gate were considered. ΔV cost above the current budget could be about 190 ft/sec. The scheme would require much increased crew and RTCC participation. Many factors which might qualify the results were not considered.

2013-FH-wcs

F. Heap

Attachments
References
Figures 1, 2 & 3

REFERENCES

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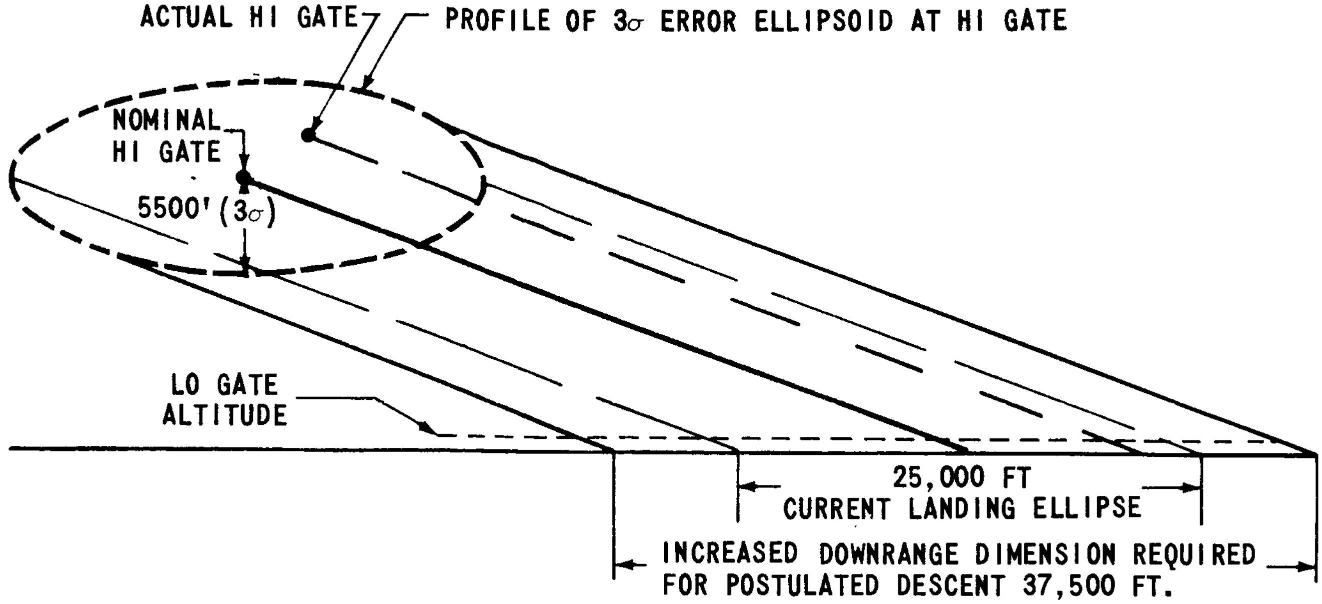
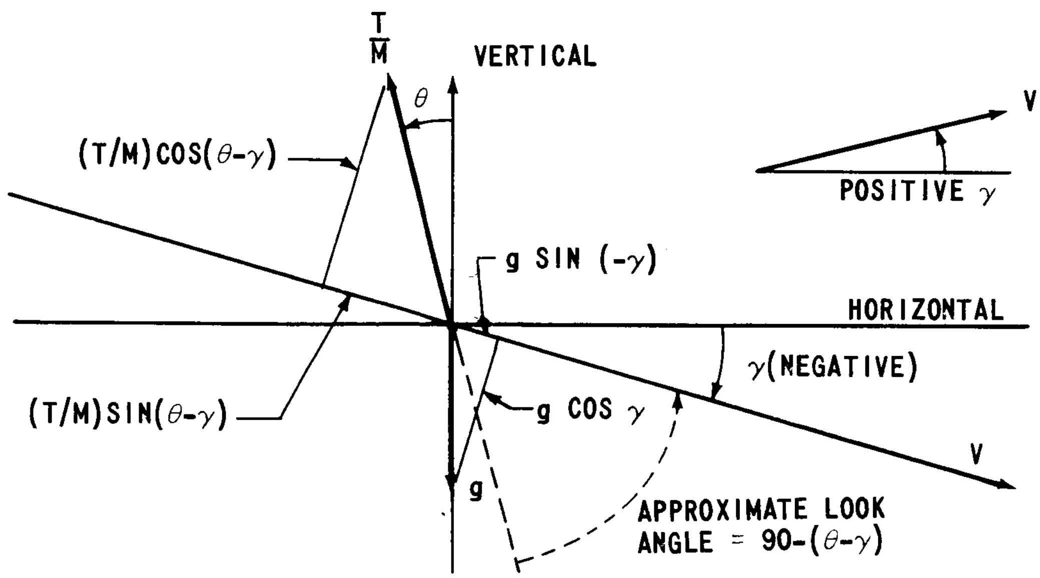
3. Proposed LM Powered-Descent Trajectory for the Apollo Lunar Landing Mission. W. M. Bolt and F. V. Bennett. MSC Internal Note 67-FM-117.

4. Revised LM Descent and Ascent ΔV Budgets for the Lunar Landing Mission. F. V. Bennett, MSC Memorandum 67-FM9-15.



a) LM FINAL APPROACH PHASE



ALONG THE FLIGHT PATH:

$$\dot{V} = -(T/M)\sin(\theta-\gamma) - g\sin\gamma$$

NORMAL TO THE FLIGHT PATH:

$$V\dot{\gamma} = (T/M)\cos(\theta-\gamma) - g\cos\gamma$$

b) LM DYNAMICS (FLAT MOON)

FIGURE 1

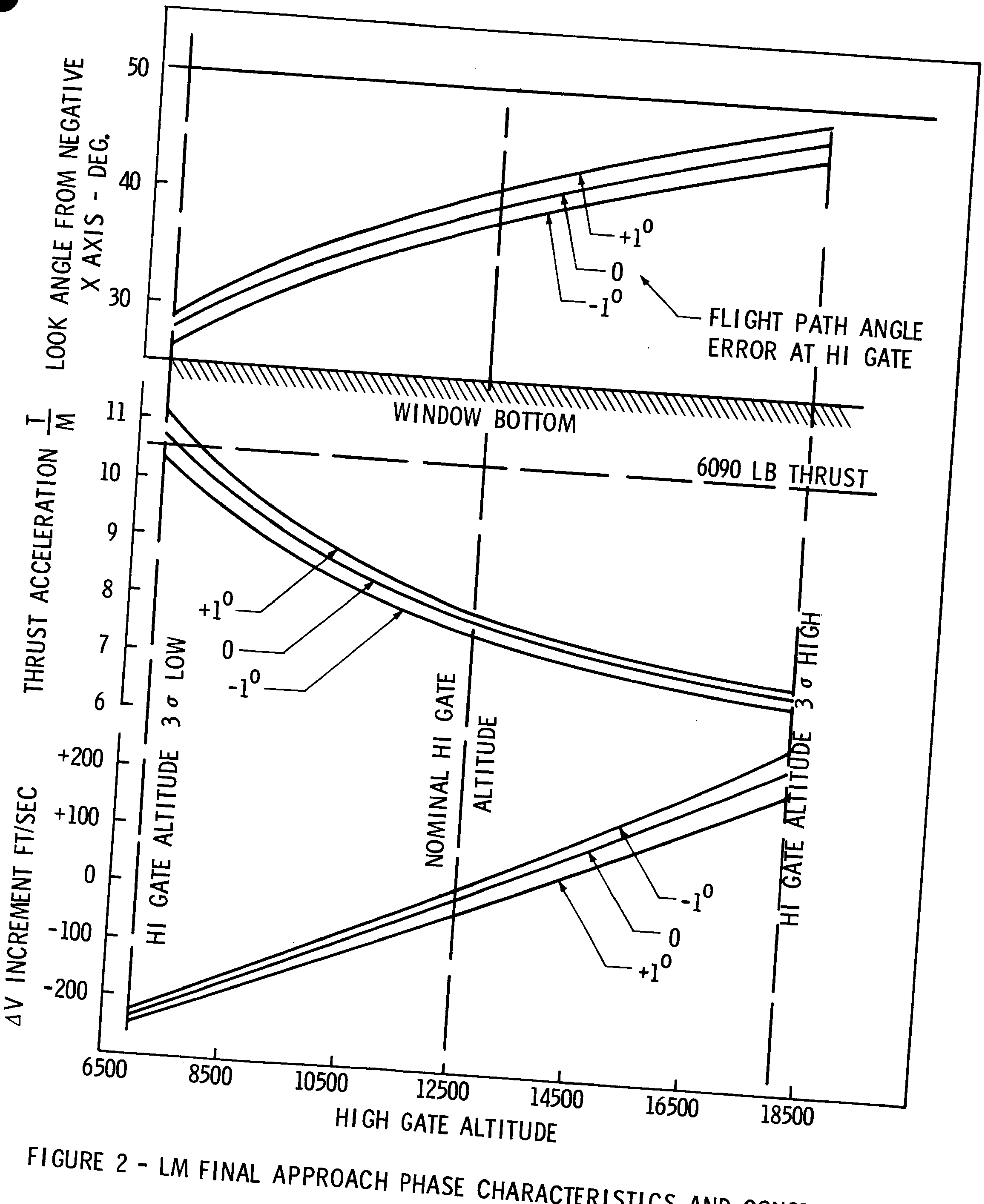


FIGURE 2 - LM FINAL APPROACH PHASE CHARACTERISTICS AND CONSTRAINTS

	PRESENT* DESCENT	POSTULATED DESCENT	ADDED ΔV
BRAKING	5345	5245	-100
FINAL APPROACH	866	915	+ 49
DPS THRUST VARIATION	40	0	- 7**
NAV. AND TERRAIN UNCERTAINTY	40	0	- 7**
LPD OPERATION	60	20	- 40
HI GATE ALTITUDE 3σ HIGH	--	295	+295***
		TOTAL	+190

*MSC MEMO 67-F M 9-15

**RSS'D WITH OTHER DISPERSIONS IN CURRENT BUDGET

***CONSERVATIVE, NOT INCLUDED AS RSS

FIGURE 3 - LM DESCENT ΔV SUMMARY

BELLCOMM, INC.

Subject: An Open Loop Crew-Monitored LM Descent - Case 310

From: F. Heap

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BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Feasibility of A Simplified LM Ascent and Rendezvous Scheme - Case 310

DATE: March 26, 1968

FROM: D. R. Anselmo
D. J. Toms

ABSTRACT

This paper presents a simplified ascent and rendezvous scheme designed to minimize LGC flight software requirements. The scheme includes simplified launch-to-insertion guidance and a concentric flight plan re-shaped to allow more complete ground support.

The simplified launch to insertion guidance has been simulated and preliminary performance and dispersion data is presented. This data shows that acceptable insertion conditions are attained with no appreciable ΔV penalty. The flight profile is discussed maneuver by maneuver and the necessary equations for on-board targeting are developed.

A list of programs for nominal and backup use is provided. Based on estimates of the software required to implement this ascent and rendezvous scheme, it appears that the LGC requirements for these portions of the profile can be reduced by about seventy-five percent.

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Feasibility of A Simplified LM Ascent and Rendezvous Scheme - Case 310

DATE: March 26, 1968

FROM: D. R. Anselmo
D. J. Toms

MEMORANDUM FOR FILE

INTRODUCTION

This study was performed as part of a Bellcomm effort to determine the feasibility of developing simplified LGC flight software while meeting the minimum requirements for carrying out a lunar landing mission. The study's objective was to determine the feasibility of carrying out a lunar mission using half the fixed and erasable memory of the LGC. The portion of the mission examined here begins at LM launch and extends through rendezvous.

The scheme examined in this report achieves considerable LM software simplification by introducing two modifications to the basic concentric flight plan:

1. Simplified launch-to-insertion guidance is employed.
2. The CSI maneuver is delayed until after a circularization maneuver is performed.

The first modification reduces computer word requirements by eliminating the present powered ascent guidance equations and using an open loop pitch program in conjunction with the cross product steering programs which are contained in the LGC under the existing system. It is shown that this simplified guidance scheme produces acceptable dispersions and no appreciable ΔV penalty.

Performing the CSI maneuver on a circular orbit has the effect of permitting all on-board calculations to be done with very simple equations and routines. In addition, the delayed CSI occurs on the front side of the moon and hence this maneuver (indeed, all maneuvers through TPI) can be targeted from the ground. Much of the on-board computational capability, therefore, has to be carried only for backup purposes. Delaying CSI also has the effect of extending the timeline thereby permitting more flexible mission planning and reducing the work load for the crew. The total time for launch to terminal braking is approximately 4.5 hours.

The body of this report is divided into three sections: Profile, Profile Calculations, and Performance and Dispersions. The Profile section is a general description of the flight profile including a discussion of lighting and tracking constraints. The Profile Calculations section describes each maneuver and presents the appropriate equations. The Performance and Dispersions section contains the results of simulated runs showing dispersions at insertion, ΔV costs and a discussion of other dispersions in the profile. The report also has appendices which define the LGC requirements for the profile studied and the simplified guidance equations for launch-to-insertion.

PROFILE

The LM ascent profile is shown in Figure 1 and covers the period from LM liftoff to terminal braking. For a nominal mission the earliest possible and latest possible times for liftoff are supplied from the ground. After liftoff the LM rises vertically for a fixed time (a time of 12 seconds was used for the simulated runs discussed in this report). At the end of the vertical rise phase the LM pitches at a high rate for a specified time. Thereafter the LM pitches at a low rate until a specified time at which the cross product steering program is switched on. The remainder of powered ascent is guided closed-loop to the proper cutoff conditions. Cutoff occurs at approximately 400 seconds after launch and results in an orbit with an apolune of 40 nautical miles (this altitude for apolune was chosen in order to insure that the CSM would always be within rendezvous radar range long before the first circularization maneuver). The CDH1 maneuver establishes a circular orbit thereby creating conditions very favorable for the subsequent CSI maneuver. Since CSI is performed on a circular orbit, the direction of the CSI burn is always along the velocity vector as well as along the local horizontal so the burn is the most efficient possible in terms of ΔV. The circular orbit also allows CSI to be delayed until lighting and tracking conditions are the most favorable.

Specific equations for on-board calculation of the CDH and CSI maneuvers are given in the Profile Calculations section of this report. These equations are developed to illustrate the simplicity of the calculations involved; it is felt by the authors that these calculations could be performed by the use of suitable charts prepared pre-flight.

In this report CSI and TPI are placed at the same longitude (over the landing site) since the most favorable lighting and tracking conditions for TPI are also very favorable for CSI. The placement of TPI was guided by the following identified constraints.

1. A requirement for 15 minutes of LM to CSM visibility prior to TPI with the line of sight 30° clear of the sun line.[1*]

2. Optical tracking requirements as follows:[2]

   (a) Continuous CSM (optical) tracking of the LM from approximately 35 minutes before TPI until 15 minutes before TPI.

   (b) Continuous CSM (optical) tracking of the LM from 5 minutes after TPI through the last terminal midcourse correction, approximately 30 minutes after TPI.

   (c) The LM must not be separated from the CSM by more than 10 n.mi. when the undocked vehicles enter the region of a lighted lunar background at the end of the terminal transfer.

3. The time between MSFN tracking acquisition and execution of a maneuver should be about 20 minutes. This allows approximately 10 minutes for tracking, 2 to 3 minutes for up link of data and approximately 7 minutes on-board for verification and pre-thrust maneuvering.[3]

The first of these constraints can be satisfied by placing TPI no sooner than 32 minutes before the CSM enters darkness. The optical constraints can be satisfied by placing TPI between 5 and 35 minutes prior to CSM entrance into darkness. The last constraint can be satisfied by placing the CSM's TPI position between 50° east and 50° west of the earth-moon line.

If the landing sites between 42°E longitude and 42°W longitude are considered all of the above constraints can be satisfied by placing TPI over the landing site. With CSI also placed over the landing site the launch window and the nominal times for all maneuvers are the same for all landing sites.

PROFILE CALCULATIONS

The powered ascent from launch to insertion.

The present LM guidance equations used from LM lunar launch to orbit insertion are based on an explicit guidance scheme which controls insertion altitude and the insertion velocity vector.

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*While this constraint originated with optical tracker considerations, it is assumed that astronaut verification of CSM elevation at TPI will also require a line of sight clear of the sun.

The scheme presented in this report consists of an open loop powered ascent up to a fixed time prior to nominal orbit insertion followed by closed loop guidance for the remainder of powered ascent. A vertical rise for a specified time, a pitch over at a high pitch rate for a specified time, followed by a low pitch rate constitutes the open loop portion of the profile. The five constants involved (vertical rise time, high pitch rate, high pitch rate time, slow pitch rate, and guidance switch-over time) are based on a nominal pre-targeted trajectory. The guidance mode employed is the well known $\overline{V}_G x \dot{\overline{V}}_G = 0$ (cross product steering) explicit scheme.

The cross product equations used involve the derivative of the required velocity equation ($\dot{\overline{VR}}$). This derivative can be either analytically evaluated or approximated by first differences. The need for this derivative can be eliminated if the $\overline{V}_G x \overline{A}_t = 0$ law is employed rather than $\overline{V}_G x \dot{\overline{V}}_G = 0$ law. The simpler alternative ($\overline{V}_G x \overline{A}_t = 0$) was studied and was found to be problematical in that a large discontinuity in the commanded thrust direction occurs at guidance switch-over. In addition, neglecting the $\dot{\overline{V}}_r$ and $\overline{g}$ accelerations would result in lower insertion altitudes. Calculation of the derivative of $\overline{V}_r$ results in an improved closed-loop system at the cost of the computer memory required to make the calculation.

The specific on-board equations required for this powered ascent mode are developed in Appendix II.

The first constant delta height maneuver (CDH1)

CDH1 is a circularization maneuver performed at apolune of the insertion orbit. For a nominal insertion this maneuver occurs approximately 56 minutes after SI. The required time and $\Delta\overline{V}$ for this maneuver can be computed on-board using the following calculations:

$$a = \frac{\mu R}{2\mu - RV^2} \quad (1)$$

$$t_{CDH1} = t_{SI} + \pi \sqrt{\mu/a^3} \quad (2)$$

t_{CDH1} = time at which CDH1 should occur.

t_{SI} = time at which SI occurred.

$\bar{R}$, $\bar{V}$ = any position and velocity attained after SI (this can be an updated state vector obtained from the rendezvous radar).

$$\Delta\bar{V} = \left(\sqrt{\mu/RA} - VA\right)\hat{U} \qquad (3)$$

RA, VA = radius and velocity at apolune (obtained with KEPLER routine)

$\hat{U}$ = unit local horizontal vector at flight azimuth.*

For central and eastern landing sites it is expected that ground tracking and communication time will be sufficient to permit the required time and $\Delta\bar{V}$ to be supplied from the ground. For far western landing sites (42°W) the time between SI and loss of MSFN is approximately 10 minutes. Since this may prove to be insufficient time for a ground based computation the maneuver may have to be computed on-board in this case. Tabular data may, however, be sufficient for this calculation.

The Concentric Sequence Initiation (CSI)

With this profile the CSI maneuver is the initiation of a Hohmann transfer. This maneuver occurs on the front side of the moon and is placed so that sufficient time is available for ground tracking, ground computation, and data up-linking both before and after the maneuver. In this study CSI is placed so as to occur over the landing site. As mentioned earlier, placing CSI over the landing site satisfies lighting and tracking constraints for all landing sites and for all anticipated lighting conditions; it does not, however, maximize the duration of the launch window.

---

*The ^ appearing over a quantity is used to designate a unit vector.

If loss of ground tracking or ground communication occurs, the time for this maneuver and the required $\Delta\overline{V}$ can be computed on-board using the following method.

First we consider the calculation of CSI time after the first CDH maneuver has been executed. Referring to Figure 2, the angle ϕ between the LM present position and the position at TPI is calculated. First a unit vector in the direction of the CM's TPI position is projected into the LM's flight plane.

$$PR\hat{T}PI = \overline{NRTPI} \times \overline{H}$$

where $NR\hat{T}PI = \overline{H} \times \overline{RTPI}$

$\overline{H}$ is the angular nomenturn vector of the LM orbit i.e., $(\overline{R}x\overline{V})$.

$\overline{RTPI}$ is the position vector of the CM at TPI.

Then ϕ is given by

$$\phi = [1-(S)1]\,[1-(D)1]\pi + S \cos^{-1}[\hat{R}\cdot PR\hat{T}PI] \qquad (4)$$

where: $S = \text{Sign}\,[\overline{H}\cdot(\overline{R}xPR\hat{T}PI)]$

$D = \text{Sign}\,[\overline{R}\cdot P\hat{R}TPI]$.

The CM's lead angle at TPI is given by

$$\Delta\phi = \frac{\Delta h \cot EL}{R} \qquad (5)$$

where: Δh = differential height between LM and CM orbits at TPI

EL = elevation of CM above LM local horizon at TPI

R = LM orbital radius.

The total coast angle from present position to the LM position at TPI is then

$$A = 2\pi + \phi - \Delta\phi. \tag{6}$$

The time for CSI execution is

$$t_{CSI} = T\left[1 + \frac{W_2}{W_1 - W_2}\right] + \frac{A - \pi - W_2(t_{TPI} - t_{HOH})}{W_1 - W_2} \tag{7}$$

where

W_1 = orbital rate at CDH1 altitude = $\sqrt{\frac{\mu}{RP^3}}$

W_2 = orbital rate at CDH2 altitude = $\sqrt{\frac{\mu}{RA^3}}$

T = present time

$t_{HOH} = \pi\sqrt{\frac{(RP + RA)^3}{8\mu}}$

t_{TPI} = time of TPI supplied from ground or chart

RP = perilune of transfer from CSI to CDH2

RA = apolune of transfer from CSI to CDH2.

The $\Delta\overline{V}$ at CSI can be computed using

$$\Delta\overline{V} = \left[\sqrt{\frac{\mu\ RA}{(RA + RP)\ RP}}\ \hat{U}\right] - \overline{V} \tag{8}$$

where

$\overline{V}$ = LM velocity at time of CSI (obtained with Kepler routine).

The second constant delta height maneuver (CDH2)

This maneuver is the same as CDH1 except that for all nominal missions the computations are done on the ground.

The terminal phase initiation maneuver (TPI)

A nominal time for TPI is given to the LM before launch. This nominal time is used to target the CSI maneuver. Because of dispersions in the CSI and CDH2 maneuver the actual time of TPI will differ somewhat from the nominal time. Ordinarily, the actual time of TPI and the $\Delta\overline{V}$ for the maneuver are computed on the ground. In the case of ground tracking or communication failure the actual time of TPI can be obtained on-board as follows:

Propagate both vehicles to the nominal time of TPI using the KEPLER routine.

Calculate the central angle between the LM position and its desired TPI position.

$$A' = A - 2\pi \tag{9}$$

where A is calculated using Equation (6).

Obtain a new estimate for t_{TPI}:

$$t_{TPI} = t_{TPI} + A' (W_1 - W_2) \tag{10}$$

where W_1 = orbital angular rate of the LM

W_2 = orbital angular rate of the CSM.

By repeating the above process using the new estimate as an input the time of TPI can be obtained to any desired degree of accuracy.

The $\Delta\overline{V}$ for the maneuver is obtained as follows:

Propagate the LM to the time of TPI using the KEPLER routine (position $\overline{R1}$).

Propagate the CSM to the time of rendezvous (position $\overline{R2}$) using the KEPLER routine (the time between TPI and rendezvous is set prior to the mission and is independent of Δh).

Obtain the desired LM velocity vector at TPI ($\overline{VTPI}$) using the TIME-THETA routine iteratively to determine P in conjunction with the following equation.[4]

$$\overline{VTPI} = \frac{\sqrt{\mu P}}{R1\ R2\ \text{Sin}\theta} \left[\overline{R2} - \left[1 - \frac{R2}{P} (1-\cos\theta) \right] \overline{R1} \right] \tag{11}$$

where: θ = the angle between R1 and R2

P = semi-latus rectum of transfer ellipse

Then $\Delta\overline{V} = \overline{VTPI} - \overline{V}$. (12)

Midcourse Corrections

With TPI over the landing site the time between TPI and loss of MSFN is approximately 23 minutes for a 42°W site and much longer for central and eastern sites. This should be ample time for ground computation of a midcourse correction. If for some reason a midcourse correction is desired and the ground cannot supply it (for example, a second mid-course correction) the computation can be done on-board using the technique used for determining $\overline{VTPI}$.

PERFORMANCE AND DISPERSIONS

Powered Ascent

A preliminary determination of the validity of powered ascent using an open loop pitch profile followed by cross product steering was performed by testing the behavior of these equations for both nominal and perturbed cases. For this preliminary study, perturbations were considered singly, no correlation effects were considered. The results show that it is possible to achieve cut off conditions which meet the standard insertion target conditions at apolune very precisely. The guidance scheme does not control altitude at standard insertion. For

this reason the open loop portion of the ascent profile must be designed such that a safe insertion altitude is assured in the presence of worst case initial conditions (worst case vehicle performance characteristics, and worst case error sources). The simulations performed indicate that safe insertion conditions will be obtained for a wide set of perturbations. The resulting errors introduced at the target point (apolune) can be absorbed in later maneuvers.

Two different guidance switch-over times were examined, one 200 seconds after liftoff (approximately halfway through the burn), and one 386 seconds after liftoff (approximately 14 seconds prior to nominal cutoff). In both cases safe insertion conditions resulted.

Switch-over at the earlier time produces more accurate target conditions but generally results in lower insertion altitude. Late switch-over results in a wider dispersion in insertion altitude and slightly less accuracy in insertion velocity direction.

The results of these simulations are given in detail in Table I. Results obtained for open loop flight to cutoff are also given. The cutoff in this case was determined by the attainment of a preset inertial velocity magnitude. One open loop flight (3σ low thrust) resulted in unsafe insertion conditions.

The 200 second switch-over resulted in a maximum dispersion in apolune time of arrival of approximately $\pm$ 20 seconds, the particular error source in this case being 3σ high and low thrust. The flight plan considered can handle this dispersion within the planned range of differential altitudes. The dispersion in apolune arrival time can be equated to dispersions in lift-off time, hence a 40 second range in apolune arrival time results in a reduction of 40 seconds in the available lunar launch window.

The ΔV costs associated with this powered ascent scheme for the nominal and perturbed cases are given in Table I. The present budget allows 6030 feet per second for launch into the nominal 10 n.mi. by 30 n.mi. orbit with an additional 30 fps budgeted for flexibility and dispersions. The trajectories in Table I employ a nominal 8.25 n.mi. by 40 n.mi. insertion. The standard open loop (unperturbed) target run to insertion required 6010 feet per second. Employing a 200 second switch-over time required 5998 feet per second and resulted in a 6.3 n.mi. by 40 n.mi. orbit. Guidance switch-over at 386.5 seconds does not produce a significantly different ΔV cost than the 200 second switch-over.

Minimum Δh

In order to obtain the longest possible launch window it is desirable to use the widest possible range of Δh. Maximum Δh is restricted, of course, to be less than or equal to the differential height at CDH1 unless a retrograde maneuver is permitted at CSI. Minimum Δh is restricted by the amount of dispersion that can be tolerated in the time of TPI. The dispersion in TPI time grows rapidly as Δh is reduced below 10 nautical miles. This is shown in Figure 3 which is a graph of the following equation.

$$\sigma_t = \frac{\sigma_A}{W_1 - W_2}$$

where σ_t = dispersion in TPI time

σ_A = dispersion in CSM lead angle at time of nominal TPI

W_1 = orbital angular rate of the LM

W_2 = orbital angular rate of the CSM

In this report a minimum Δh of 10 n.mi. and a maximum Δh of 20 n.mi. was used for all calculations. This results in a launch window duration of 80 seconds. As mentioned earlier the dispersion in the time of CDH1 can be as great as $\pm$20 seconds (for 3σ high and low thrust). This reduces the launch window to an effective length of 40 seconds.

CONCLUSIONS

It is concluded that considerable LGC software simplification can be achieved by employing one or both of the modifications presented here, namely the simplification of powered ascent guidance and the simplification of on-board rendezvous calculations made possible by greater reliance on ground support. The introduction of a delayed CSI provides additional MSFN coverage to support the use of these rendezvous calculations. In addition it may be possible to perform

the CSI and CDH calculations described by means of graphical or tabulated data. It should be emphasized that the two modifications are independent; each can be used alone and each leads to a significant reduction in software.

It is shown that this software simplification can be achieved with nearly the same performance requirements while meeting presently identified constraints. Further validation of dispersions, safety, and mission success probabilities would have to be performed before this scheme could be considered for actual use.

Based on the specific equations and present LGC programs retained for the implementation of this LM ascent profile it is estimated (Appendix I) that a seventy-five percent reduction in LGC software requirements associated with LM ascent and rendezvous could be achieved.

D. R. Anselmo

D. J. Toms

2013-$\frac{\text{DRA}}{\text{DJT}}$-srb

Attachments:
References
Appendix I
Appendix II
Table I
Figures 1, 2, 3

REFERENCES

1. "Introduction of the Second Design Reference Mission", M. V. Jenkins, August 24, 1965.

2. Merritt, A. C., "Revisions to the Lunar Landing Mission Rendezvous", Bellcomm Memorandum for File, February 29, 1968.

3. Personal communication with J. D. Alexander, Mission Planning and Analysis Division, MSC, February 20, 1968.

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APPENDIX I

PROGRAMS

The flight software required for this ascent and rendezvous profile can be conveniently divided between nominal and backup requirements. The following list summarizes the nominal LGC requirements in terms of present programs and specific equations which have been developed in the text.*

1. Pre-launch - Ground calculations.
2. Powered ascent.
   (a) Open loop Equations AII-1, AII-2, AII-3.
   (b) Guided flight Equations AII-4, AII-12.
   (c) Present cross product steering routine.
3. CDH maneuver
   (a) Present External ΔV program.
   (b) Present cross product steering routine.
   (c) Present conic Kepler routine.
   (d) Equations (1), (2), and (3).
4. CSI maneuver
   (a) Present External ΔV program.
   (b) Present cross product steering routine.
   (c) Ground calculations.
5. TPI maneuver
   (a) Present External ΔV program.
   (b) Present cross product steering routine.
   (c) Ground calculations.

---

*Supporting routines such as Middle gimbal and Servicer have not been considered in this list but are included in the LGC word estimates given at the end of this appendix.

6. Midcourse maneuvers

   (a) Present External ΔV program.

   (b) Present conic Time-theta and Kepler routines.

   (c) Present cross product steering routine.

   (d) Equation (11) and (12).

Suggested backup programs for the LGC which are additions to the nominal programs:

1. CSI maneuver - Equations (4), (5), (6), (7) and (8).

2. TPI maneuver - Equations (4), (5), (6), (9) and (10).

Present programs and routines which are employed for LM powered ascent and rendezvous maneuvers are given in the following lists. The MIT budget and the estimated word requirements are given.

PROGRAMS	PRESENT BUDGET	NEW ESTIMATE
AGS Initialization	100	100
Rendezvous Out of Plane Display	120	0
Preferred Tracking Attitude	50	50
External ΔV	150	50
Predicted Time of Launch	200	0
General Lambert	120	0
APS Abort	100	100
Thrust monitor	70	0
CSI Pre-thrust	70	0
CDH Pre-thrust	100	0
TPI Pre-thrust	460	0
TPM Pre-thrust	95	0
APS thrust	30	30

Basic Routines		
Conic	1050	800
Orbit Integration	1400	0
Latitude Longitude Altitude	170	0
Planetary Initial Orientation	280	0
Initial Velocity	75	0
Rendezvous Parameter	90	0
Middle Gimbal	75	0
Target Routines		
Predicted Launch Time (CGP)	650	0
CSI Initiation	440	130
Constant Delta Altitude	300	50
TPI Initiation	0	120
Predicted Launch Time (D.T.)	100	0
TPI Search	200	0
Powered Flight Routines		
Servicer	850	500
Cross Product Steering	75	60
VG calculation	100	0
Time of Burn Calculation	105	0
Ascent Guidance	700	100
Totals	8325	2090

It should be noted that while these estimates include on-board calculations of both the CDH and CSI maneuvers the possibility of using on-board charts for these maneuvers has not been excluded.

APPENDIX II

Powered Flight Equations:

The open loop portion of ascent is performed in three stages. The required acceleration vector direction is given by the following for the three open loop flight phases,

$$\hat{A}_t = \hat{R}_L \qquad \text{(vertical rise) } 0<T<T_V \qquad \text{AII-1}$$

$$\hat{A}_t = \hat{R}_L \cos W_h T - \hat{U}_L \sin W_h T$$

$$\text{(pitch over) } T_V<T<T_H \qquad \text{AII-2}$$

$$\hat{A}_t = \hat{R}_L \cos (W_L T + W_h T_H) - \hat{U}_L \sin (W_L T + W_h T_H)$$

$$\text{(low pitch rate) } T_H<T<T_{SO} \qquad \text{AII-3}$$

where

$\hat{R}_L$ = unit radius vector at liftoff

W_h = high pitch rate

W_L = low pitch rate

T_V = duration of vertical rise

T_H = T_V + duration of high pitch rate

T = elapsed time from liftoff

$\hat{U}_L$ = unit $(\hat{H} x \hat{R}_L)$, where $\hat{H}$ is unit angular momentum vector of desired orbit

T_{SO} = time of switch-over to closed loop guidance.

The guided portion of powered ascent is performed with the cross product steering equations presently employed in the on-board software. An additional calculation required is

the development of the required instantaneous velocity ($\overline{V}_R$). The derivative of $\overline{V}_R$ with respect to time must also be computed. This can be done numerically by first differences or analytically. The $\overline{V}_R$ equation and the analytic $\dot{\overline{V}}_R$ equations are developed in what follows.

$$\overline{V}_R = \left[\frac{2\mu \ RA}{(RA + R) \ R} \right]^{1/2} \hat{U} \qquad \text{AII-4}$$

where

RA = apolune target radius

R = magnitude of present radius vector

μ = lunar gravitational constant

$\hat{U}$ = unit local horizontal ($\overline{H} x \overline{R}$)

The necessary equation for $\dot{\overline{V}}_R$ is now developed

$$\overline{V}_R = V_R \ e^{-j\omega t}$$

$$\text{where } \omega = \frac{V_{Tangential}}{R} = \frac{\overline{V} \cdot \hat{U}}{R} \qquad \text{AII-5}$$

$\overline{V}$ = actual velocity

Note that $e^{j\omega t}$ is the instantaneous local horizontal if we define

$$e^{-j\omega t_o} = \hat{U}_o. \qquad \text{AII-6}$$

Taking the derivative of $\overline{V}_R$ we obtain

$$\frac{d\overline{V}_R}{dt} = - V_R \, [j\omega e^{-j\omega t}] + \frac{dV_R}{dt} \, [e^{-j\omega t}] \qquad \text{AII-7}$$

where

$$\frac{dV_R}{dt} = [\frac{RA + 2R}{2R \, (R+RA)}] \, V_R \frac{dR}{dt} = K \frac{dR}{dt} \qquad \text{AII-8}$$

Hence

$$\frac{d\overline{V}_R}{dt} = K \frac{dR}{dt} \, [e^{-j\omega t}] - V_R \, [\frac{\overline{V} \cdot \hat{U}}{R}] \, je^{-j\omega t} \qquad \text{AII-9}$$

if we observe that:

$$\frac{dR}{dt} = \frac{\overline{V} \cdot \overline{R}}{R} \qquad \text{AII-10}$$

and

$$e^{-j\omega t} = \hat{U}$$

then

$$je^{-j\omega t} = \hat{R} \qquad \text{AII-11}$$

The required derivative is then given by

$$\frac{d\overline{V}_R}{dt} = K(\overline{V} \cdot \hat{R})\hat{U} - V_R(\frac{\overline{V} \cdot \hat{U}}{R}) \, \hat{R} \qquad \text{AII-12}$$

TABLE I
POWERED ASCENT SIMULATIONS

OPEN LOOP POWERED ASCENT

PERTURBED PARAMETER	BURN DURATION	ΔV	FLIGHT PATH ANGLE	ALTITUDE INSERTION	ALTITUDE PERICENTER
STANDARD RUN ISP 306.3 SEC WGT 10,209 LBS APS THRUST 3,500 LBS LOW PITCH RATE -.1263 DEG/SEC DURATION H.P. RATE 6.6786 SEC	403.3 (SEC)	6010.4 (FPS)	-.0152 (DEG)	49,764.8 (FT)	49,752.5 (FT)
ISP 303.3 (3σ LOW)	402.3	6011.4	.0554	50,551.6	50,388.5
ISP 309.3 (3σ HIGH)	404.2	6009.4	-.0846	48,985.3	48,600.3
THRUST 3395 (3σ LOW)	413.6	5982.8	-1.1914	30,385.6	-36,015.5
THRUST 3605 (3σ HIGH)	393.3	6034.9	1.1057	67,269.7	20,903.0
WGT 10109	399.9	6018.9	.3643	55,794.7	49,302.0
WGT 10309	406.6	6001.8	-.3936	43,644.4	35,187.5
DURATION OF H.P. RATE -.05	403.9	6023.3	.3407	55,852.7	50,155.0
DURATION OF H.P. RATE +.05	402.6	5997.6	-.3707	43,700.0	36,164.0
LOW PITCH RATE (-1%)	403.5	6016.1	.2444	52,657.7	49,597.4
LOW PITCH RATE (+1%)	403.0	6004.8	-.2691	46,938.7	33,026.7

OPEN LOOP PLUS CROSS PRODUCT POWERED ASCENT

GUIDANCE SWITCH-OVER TIME	ALTITUDE AT INTT	ALTITUDE AT INS.	ALTITUDE PERICENTER	FLIGHT P. ANGLE	BURN DURATION	ΔV
200 (SEC)	31,703.4 (FT)	38,149.8 (FT)	38,149.7	.0022 (DEG)	402.6 (SEC)	5,997.5 (FPS)
386.5	49,706.6	49,769.2	49,769.2	.00036	403.2	6009.7
200	31,798.3	38,433.1	38,432.8	.0022	401.6	5997.3
386.5	50,397.2	50,498.5	50,351.1	-.0032	402.2	6010.2
200	31,610.5	37,870.8	37,870.8	.0022	403.7	5997.7
386.5	49,033.0	49,052.6	48,904.0	-.0035	404.2	6009.3
200	27,408.8	28,632.8	28,632.5	.0019	414.4	5997.2
386.5	33,157.2	31,970.7	31,900.4	.0377	414.8	6006.1
200	36,021.7	47,112.2	47,111.9	.0022	391.7	5999.9
386.5	66,536.0	66,939.8	65,474.5	-.158	394.9	6069.0
200	33,138.6	41,193.1	41,192.8	.0025	398.9	5998.1
386.5	55,280.9	55,534.2	55,494.6	-.0266	399.8	6017.0
200	30,297.3	35,103.4	35,103.3	.0020	406.4	5997.1
386.5	44,265.8	44,036.9	44,020.5	.0177	406.8	6006.7
200	32,895.5	40,864.1	40,863.9	.0024	402.9	6001.9
386.5	55,207.0	55,531.0	55,511.6	-.0187	403.7	6020.3
200	30,506.9	35,430.0	35,429.8	.0020	402.4	5993.2
386.5	44,187.0	44,002.0	44,977.1	.0218	402.9	6002.6
200	31,951.2	38,874.3	38,874.1	.0023	402.7	5998.6
386.5	52,192.3	52,430.5	52,421.4	-.0129	403.5	6014.4
200	31,460.2	37,439.0	31,438.9	.0022	402.6	5996.4
386.5	47,266.5	47,160.6	47,152.1	.0126	403.1	6006.5

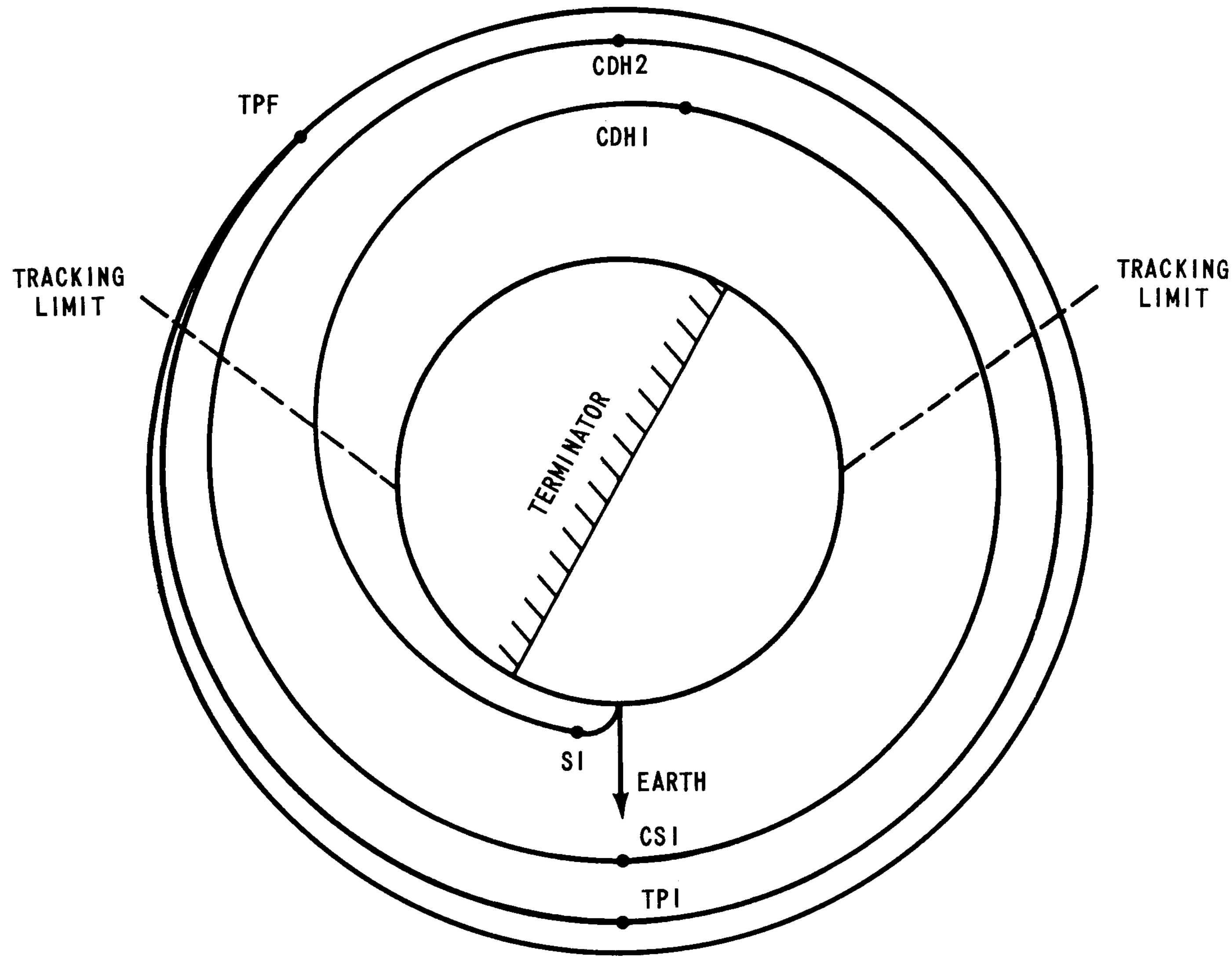

FIGURE 1 - CONCENTRIC FLIGHT PLAN WITH DELAYED CSI

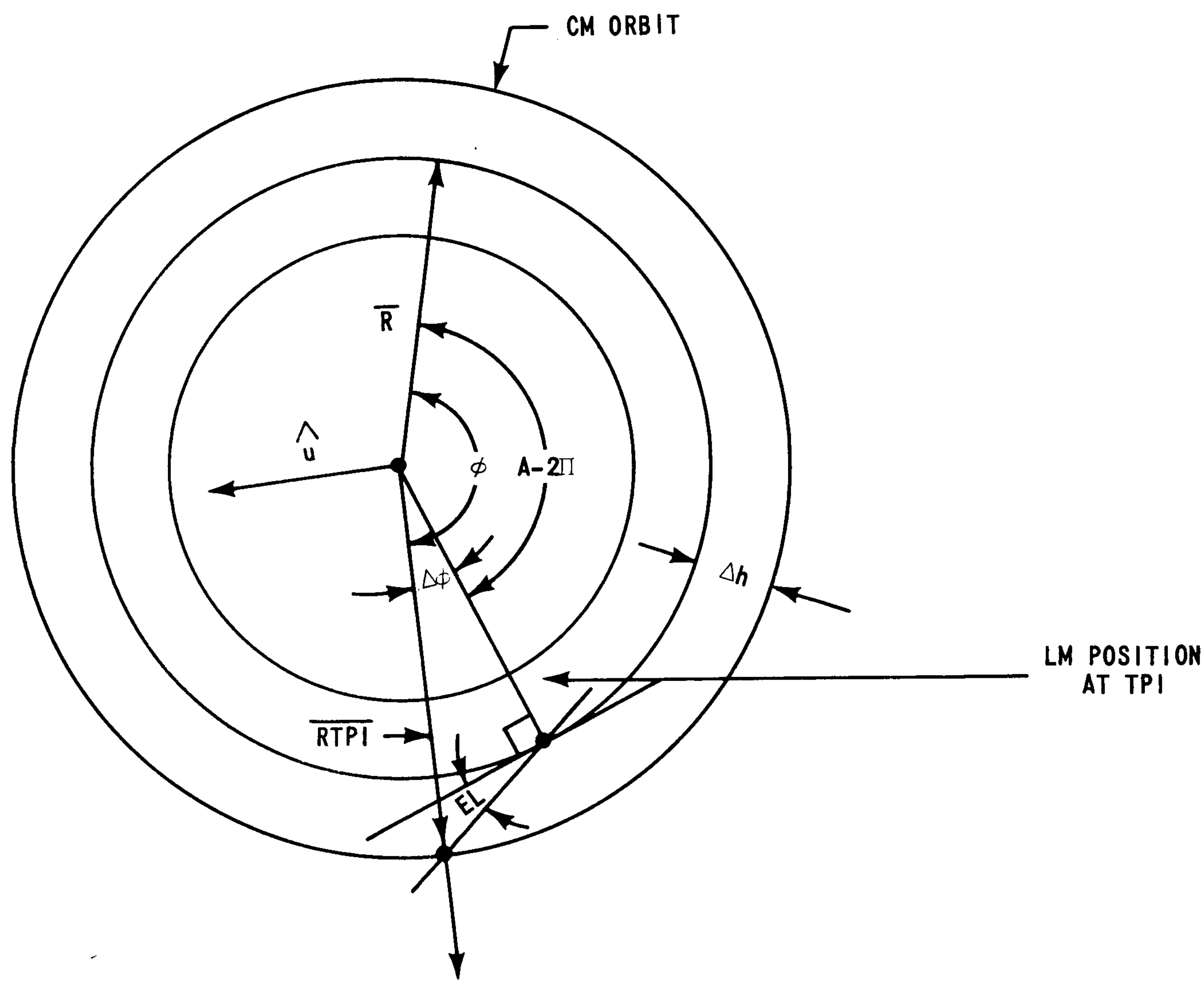

FIGURE 2 - TPI GEOMETRY

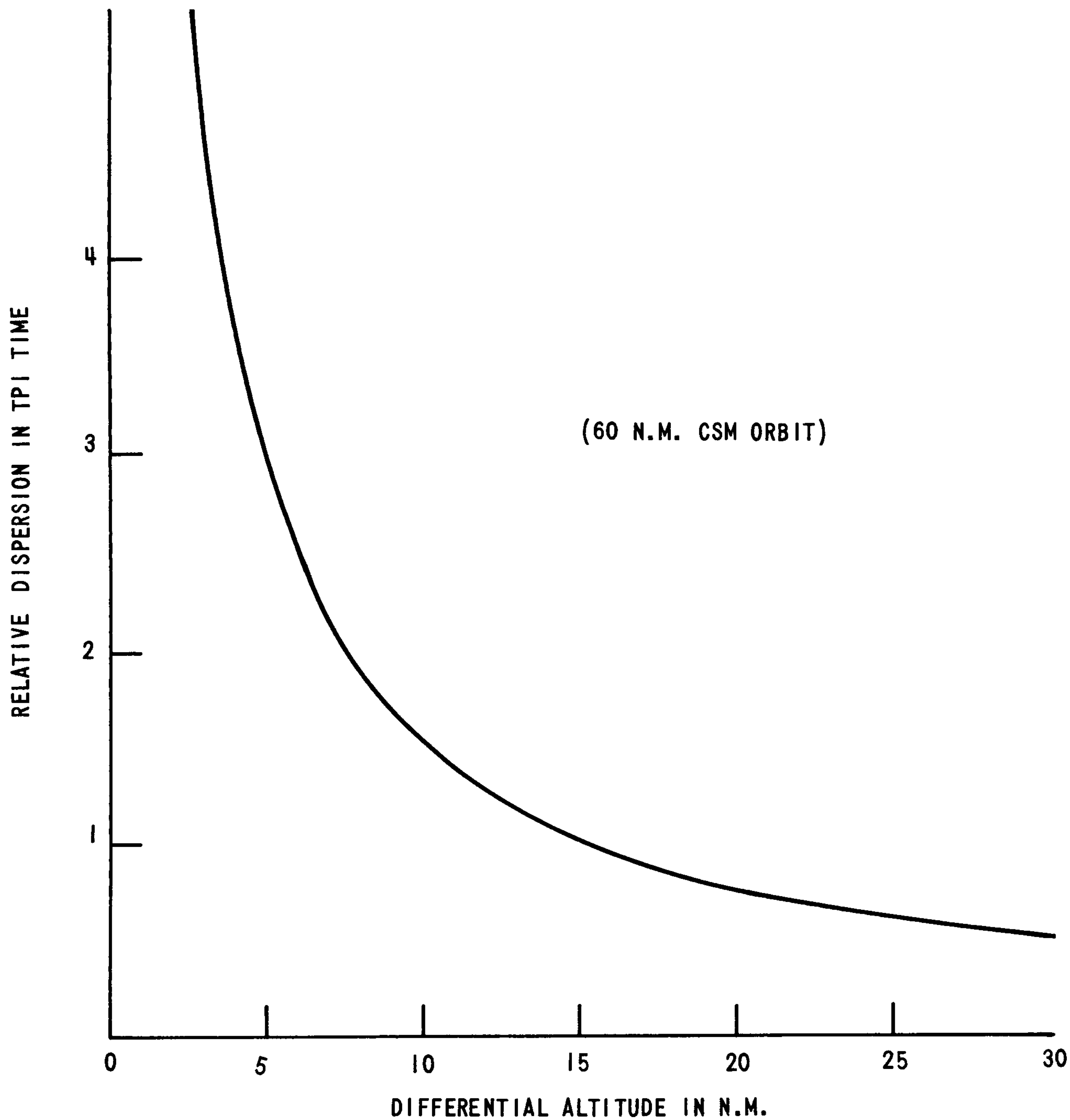

FIGURE 3 - RELATIVE DISPERSION IN TPI TIME AS A FUNCTION OF Δh

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SUBJECT: A LM Rescue Strategy Requiring No On-Board Targeting Case 310

DATE: March 28, 1968

FROM: S. L. Levie, Jr.

ABSTRACT

A four-burn strategy for LM rescue is proposed in which provisions are made for all necessary real-time targeting to be done on the earth. Rescue rendezvous is constrained to occur in the dark and hard docking in the light. The strategy has been applied to rescuing the LM from a 10 nm by 40 nm ascent ellipse.

In this application the total rescue delta velocity is well within the present budget of 791 fps, and the braking delta velocity can be kept below about 140 fps. The elapsed time between the strategy's first and last burns is less than four hours. Perilunes on all CSM orbits are above 50,000 ft. It is anticipated that these figures will hold for more general applications of the strategy.

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: A LM Rescue Strategy Requiring No On-Board Targeting
Case 310

DATE: March 28, 1968

FROM: S. L. Levie, Jr.

MEMORANDUM FOR FILE

1.0 INTRODUCTION

This memorandum presents a simple, effective strategy for CSM rescue of the LM. Developed as part of an effort to reduce substantially the computational requirements on the CSM and LM guidance computers, the strategy requires no on-board targeting capability for either vehicle. Time for earth tracking and earth targeting is provided for those maneuvers which must be targeted in real-time. The remaining maneuvers can either be targeted prior to a mission or uplinked to the CSM before it goes out of sight behind the moon.

The strategy's rationale will be illustrated by developing the case of rescuing the LM from a 10 nm by 40 nm Hohmann ascent trajectory[1]. Fuel-wise, this is one of the more difficult rescues the CSM might be called upon to make. The strategy has been applied only to this case; however, preliminary investigations indicate that it is applicable to other phases of LM ascent as well as to the LM's Hohmann descent trajectory. It is expected that the delta velocity requirements for such applications will not significantly exceed those developed in the 10 nm by 40 nm ascent trajectory application.

The targeting computations for rescue, assumed for this study to be done on the earth, can be done on-board with the programs currently planned for the Apollo Guidance Computer.

2.0 RESCUE STRATEGY

The rescue strategy utilizes four CSM burns. The first two constitute a Hohmann transfer to a circular phasing orbit. The altitude of this orbit depends on the expected range of CSM lead angles at initiation of the rescue. The CSM

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[1]These figures are consistent with those used in Reference 1, which examines LM ascent with simplified guidance and targeting logic.

coasts in its circular phasing orbit through approximately 360°. The third burn--the TPI[2] burn--occurs when the CSM is near the center of the lunar disc, as viewed from earth. The TPI burn is adjusted so that the fourth burn, or braking maneuver at rendezvous, occurs in the dark and requires a minimum velocity change.

It will be shown that the burns of the Hohmann transfer may be precomputed on the earth. Because TPI occurs near the center of the lunar disc, good earth tracking and earth targeting are made possible for both the third and the fourth burns of the rescue sequence.

2.1 Hohmann Transfer

It will be assumed that when the LM reaches apolune on its 10 nm by 40 nm ascent ellipse and at that point attempts a circularization burn, it is found to be without propulsive ability. For mathematical convenience, it is assumed that the CSM, which is in a 60 nm circular orbit and leading the LM by some angle θ, performs its rescue sequence initiation (RSI) maneuver at this instant. In reality there would be a time delay before the maneuver could be initiated, but the effect of reasonable delays on the phasing and on the delta velocity requirements is negligible. The RSI lead angle, θ, depends on when in its launch window the LM leaves the lunar surface.

In the rescue sequence proposed here, the RSI maneuver is the first burn of a Hohmann transfer of the CSM to a circular phasing orbit of altitude H. It will be shown that, given the landing site longitude and the expected range of θ resulting from the allowable launch window, that a value of H can be selected such that the delta velocity of the fourth burn is a minimum over the expected range of θ. This makes possible the computation of the parameters for the Hohmann transfer well in advance of a mission. Thus real-time computations are not required for the Hohmann transfer maneuvers.

For certain landing sites, additional refinements to the strategy for targeting the Hohmann burns are available. For eastern sites, sufficient tracking opportunity exists for

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[2]Terminal phase initiation.

reoptimizing the phasing orbit altitude for the actual value of θ resulting from LM launch. For western landing sites, sufficient tracking is available prior to the second Hohmann burn to allow computation of an update for the maneuver's parameters. This study does not take advantage of these possible refinements.

2.2 Terminal Phase

After the CSM has travelled through 360 + F degrees (a criterion for choosing F will be presented shortly) in its phasing orbit, it is to execute a maneuver computed to place it on a LM-intercept trajectory. This is the terminal phase initiation (TPI) maneuver. To be able to compute this maneuver and any subsequent midcourse correction with confidence, good earth tracking of the CSM before and immediately after TPI is required. This can be achieved by having TPI occur near the center of the lunar disc, as viewed from the earth. Accordingly, the TPI maneuver is located at zero degrees selenographic longitude.

With the TPI location defined, F can be computed, given the following: L, the landing site longitude; θ, the CSM lead angle at RSI; and A, the central angle for LM ascent to perilune on the 10 nm by 40 nm ellipse (A will be approximately 10°). Referring to Figure 1, if at RSI the CSM leads the LM by θ, then the CSM's transfer orbit perilune will be θ degrees west of the LM's perilune, and therefore θ+A degrees west of the landing site. Since the site is at longitude L, the CSM must therefore fly through F = L - (θ+A) degrees from transfer orbit perilune to zero degrees longitude[3].

If the phasing orbit altitude, H, is specified, the time from RSI to TPI can readily be computed as

$$\Delta t = \pi \sqrt{\frac{\left(R_M + \frac{60+H}{2}\right)^3}{\mu}} + 2\pi\left(\frac{360+F}{360}\right)\sqrt{\frac{\left(R_M+H\right)^3}{\mu}} \quad ,$$

[3]The assumption that the trajectories lie in the moon's equatorial plane has been made here. Although this condition will in practice never occur, it is a good approximation for Apollo flights and for presenting this strategy.

in which R_M is the lunar radius and μ is the lunar gravitational constant. (A criterion for choosing H will be stated shortly.) Knowledge of Δt is necessary for the computation of the LM's position at TPI.

2.3 Rescue Rendezvous

The final maneuver of the terminal phase (TPF) occurs when the CSM has flown from TPI through a central angle δ. The choice of δ will depend primarily on rendezvous requirements[4]. The principal requirement is that the maneuver should occur in the dark, at least sixty degrees before the LM would arrive in the sunlight. This allows rendezvous maneuvering to occur in darkness, followed by hard docking in sunlight.

Figures 2 and 3 depict the geometry involved in determining the value of δ for an assumed landing site on the equator at 40° East. Figure 2 shows that ε, the sun elevation at LM landing, is equal to the difference in longitude between the landing site and the dawn terminator. Current planning calls for the sun elevation at LM landing to be between 5° and 15°. Figure 2 assumes a 10° sun elevation. Figure 3 shows the landing site at LM launch, about 24 hours--or 13° of lunar rotation--after LM landing. The figure also shows the required location of TPF[5]. Since L = 40°, one sees from the figure that δ must be 103°.

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[4] Ideally, one would like to adjust δ such that the LM would be very near apolune at rendezvous, since its speed is smallest there. However, since all the lunar orbits in the Apollo program are nearly circular, any beneficial effects of rendezvousing near apolune will be slight. Therefore the location of the LM at rendezvous is of small importance, and the effect of δ on rescue delta velocity costs is small.

[5] The TPF location shown in Figure 3 does not include compensation for the moon's rotation during the interval from TPI to TPF. This rotation, which amounts to about 0.5°, may be ignored in selecting δ.

A similar calculation for a site at L = -40° indicates δ = 185°. This would place the LM very near apolune at rendezvous. However, to avoid possible computational difficulties when applying the strategy and using conic routines, an arbitrary upper limit of δ = 160° has been established. According to Footnote 4, this is certainly adequately close to 185°.

For sites west of -15° longitude, the required δ would always be greater than 160°. Therefore, for these sites a value of 160° will be used in lieu of the computed δ, to insure that no convergence problems will be encountered. For the sites east of -15° longitude, the computed value of δ will be used.

2.4 Phasing Orbit Altitude

In the general case, the CSM's phasing orbit altitude, H, as yet unspecified, can be used to adjust the phasing at TPI such that the terminal phase of the rescue uses as little CSM propellant as possible. In particular, H can be used as a parameter for minimizing the TPF fuel requirement. The best value for H will depend on the value which is selected for δ, and on the RSI lead angle, θ. As has been indicated, δ is a constant for a particular landing site. However, θ depends on the CSM's lead angle at LM injection, and this cannot be known prior to a mission. Nevertheless, it is possible to specify beforehand a value of H for which the TPF fuel costs will be small for all lead angles resulting from the expected LM launch window (this is shown in Section 3.0)[6]. Furthermore, for eastern landing sites there will be time after LM injection for earth tracking and real-time computation of the optimum H.

2.5 Time Estimate

For the range of δ indicated above, the total time interval from TPI to TPF is about half of a LM period. Since the interval from RSI to TPI is about one and a half periods, the total duration of the rescue sequence is about two LM periods, or four hours. Noting that after rescue rendezvous about two hours might be required for coasting into sunlight, docking, and transferring of the crew, and also that RSI occurs

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[6] But the TPF fuel cost will be a true minimum for only one value of the lead angle.

approximately one hour after LM launch, then about seven hours transpire between LM launch and crew transfer. This estimate lies within the LM ascent stage lifetime of nine or more hours.

3.0 NUMERICAL SOLUTION

The mathematics of the rescue strategy described above have been applied to the particular problem of initiating LM rescue when the LM arrives at apolune on its 10 nm by 40 nm elliptical trajectory. The CSM lead angle, θ, at this point will lie somewhere in the interval 12.9° to 17.3° for a landing site at 40° East, and in the interval 11.7° to 15.6° for a landing site at 40° West[7]. Computations were made for both of these sites.

For the eastern site, δ was taken as 103°, as computed earlier. H was chosen such that the minimum of the TPF delta velocity corresponded to $\theta = 15.1°$, the center of the expected interval. This value is H = 36.05 nm. In Figure 4 are plotted, as functions of θ, the total rescue delta velocity, the TPF delta velocity, and another important quantity: the terminal phase orbit's perilune altitude. (This figure should be kept above 50,000 ft, for CSM safety in case of difficulty.) It is to be observed that for all values of θ in the interval 12.9° to 17.3°:

1. Rescue delta velocity is below 380 fps.

2. TPF delta velocity is below 138 fps.

3. Terminal phase perilune is above 9.5 nm.

For the western site, δ was taken as 160°, as suggested earlier. H was chosen to place the minimum of the TPF fuel requirement at $\theta = 13.7$, the center of the expected interval. This is accomplished with H = 37.52 nm. Figure 5 plots as functions of θ the rescue delta velocity, the TPF delta velocity, and the terminal phase orbit's perilune altitude. For all values of θ in the interval 11.7° to 15.6° it is found that:

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[7]Reference 1 gives the range of CSM lead angles at LM injection for these sites. Knowing that the CSM loses 9.2° of lead as the LM travels from perilune to apolune allows the computation of the intervals used here.

1. Rescue delta velocity is below 275 fps.

2. TPF delta velocity is below 75 fps.

3. Terminal phase perilune is above 13 nm.

For practical purposes, the most important aspect of the results above is that for a given lunar landing site near the equator, δ and H can be preselected to give acceptable values of important quantities, for all expected lead angles. It is to be noted that if H can be selected in real-time, then it is possible to make the minimum TPF delta velocity correspond to the actual value of θ.

4.0 SUMMARY AND CONCLUSIONS

A simple strategy for LM rescue has been presented. The rationale of the strategy has been illustrated by applying it (with favorable results) to the difficult case of rescuing the LM from a 10 nm by 40 nm elliptical ascent trajectory.

The strategy uses four CSM burns. The first two place the CSM in a circular phasing orbit. The third, which occurs near the center of the lunar disc, places the CSM on a LM-intercept trajectory chosen such that the terminal, or braking, burn requires a minimum of delta velocity.

The strategy was designed with the intent of keeping real-time rescue computations simple and allowing full use of the RTCC. This last aim is achieved by providing good earth tracking of the CSM before and after the TPI burn. Contributing to the simplicity of the strategy is the fact that the constants affecting the first two burns may be computed prior to launching the mission.

The strategy offers rescue delta velocity costs (including braking phase) of less than about 400 fps. Even after allowing an additional 200 fps for a Hohmann transfer to a 60 nm orbit, from which transearth injection might be made, the delta velocity cost is well within the budget of 791 fps. The strategy requires a braking delta velocity of less than 140 fps, and it allows safe perilunes throughout the rescue. The rescue requires four hours at most.

Although the strategy was developed for rescuing the LM from an elliptical orbit, it can equally well be applied to rescuing the LM from a circular orbit. However, in that case the expedient of a direct, Hohmann transfer from the CSM orbit

to the LM, with phasing adjusted by delay of the first burn, provides a simpler alternative.

S. L. Levie, Jr.

2011-SLL-vh

S. L. Levie, Jr.

Attachments
Reference
Figures 1-5

REFERENCES

1. Anselmo, D. R., and Toms, D. J., "Feasibility of a Simplified LM Ascent and Rendezvous Scheme," Bellcomm Memorandum for File, March 20, 1968.

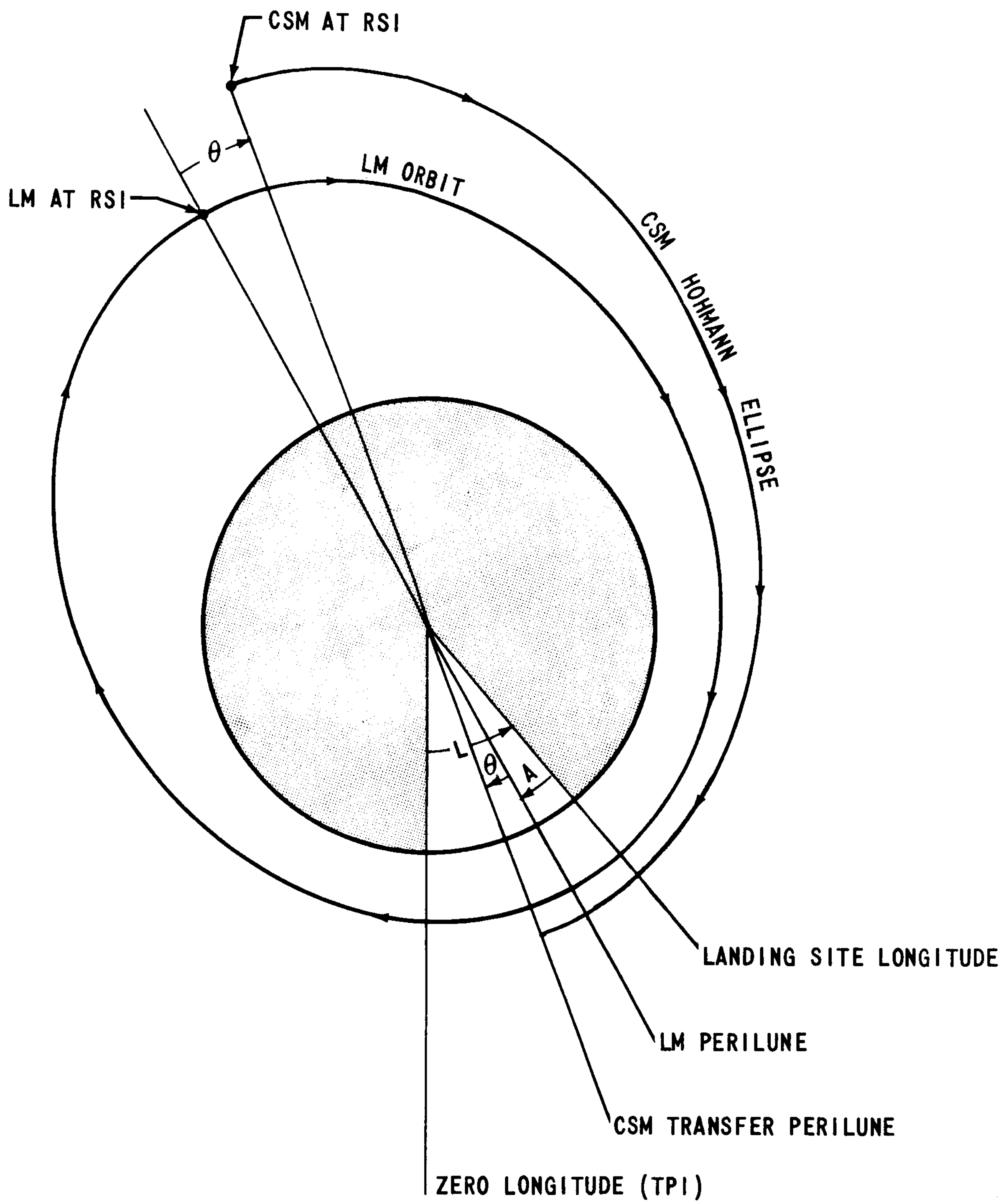


FIGURE 1 - PART OF THE LM RESCUE GEOMETRY. (VIEW LOOKING DOWN ON MOON'S NORTH POLE)

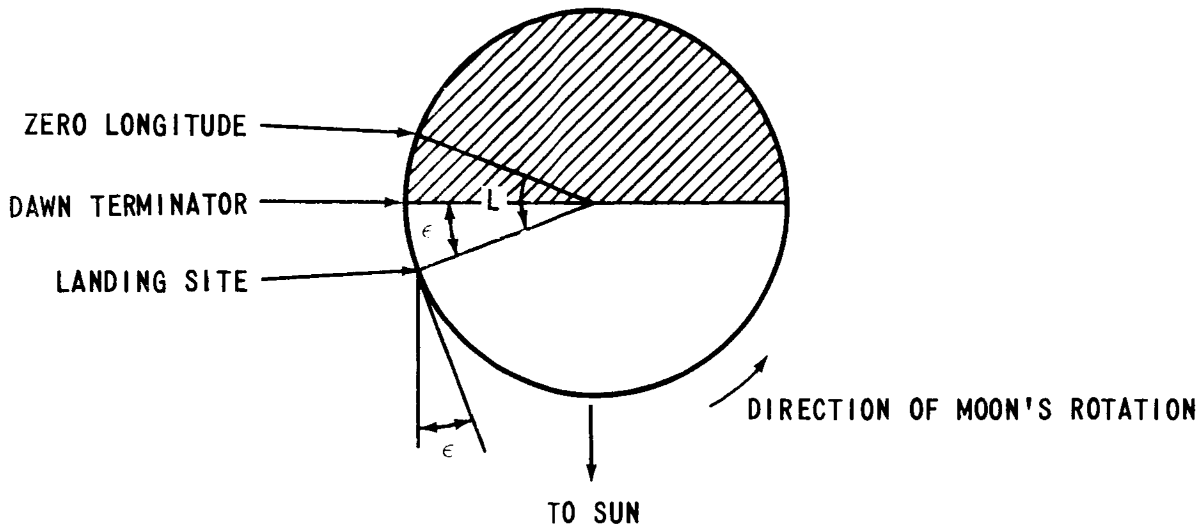


FIGURE 2 - RELATION BETWEEN THE LIGHTING CONSTRAINT AND THE LANDING SITE LOCATION ($\epsilon \sim 10°$)

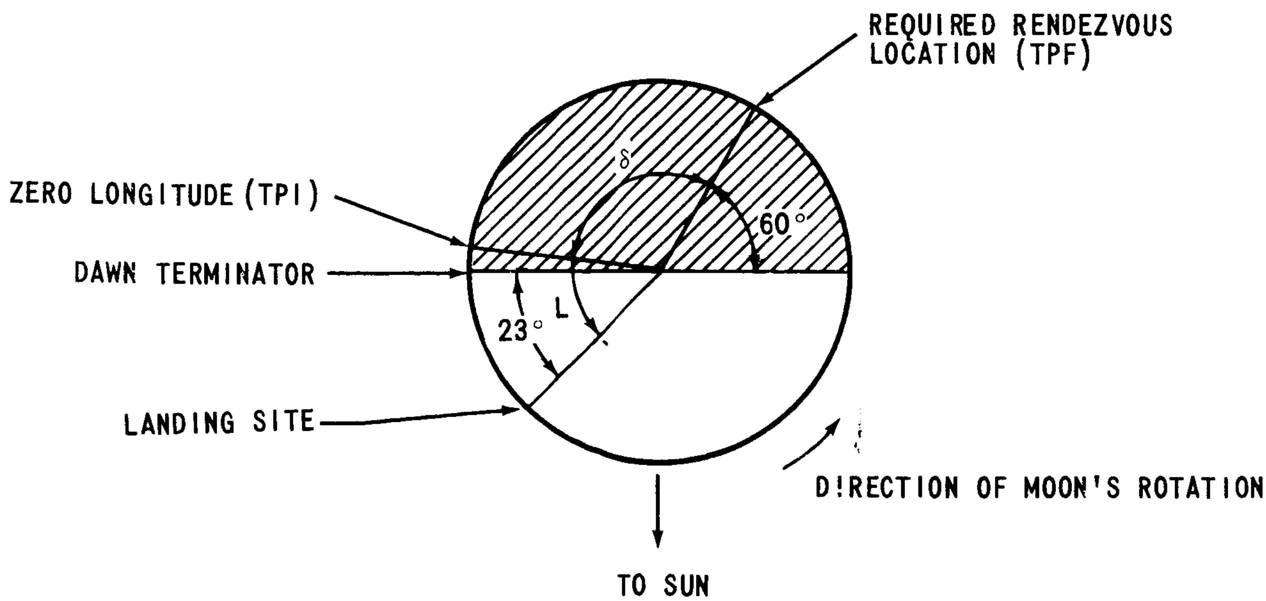
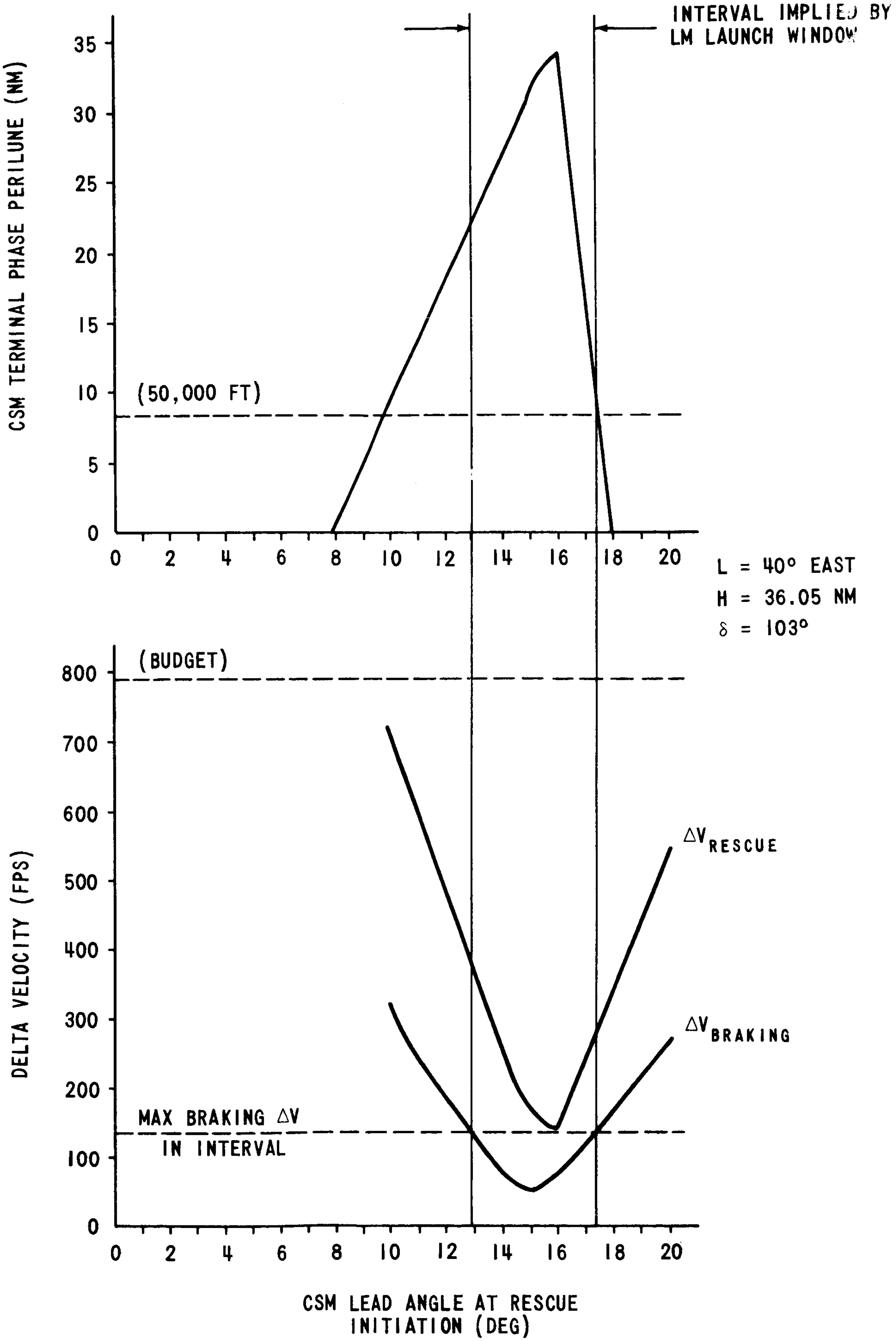


FIGURE 3 - RESCUE GEOMETRY AT LM LAUNCH. NOTICE THE (APPROXIMATE) LOCATIONS OF TPI AND TPF



INTERVAL IMPLIED BY
LM LAUNCH WINDOW
CSM TERMINAL PHASE PERILUNE (NM)
(50,000 FT)
35
30
25
20
15
10
5
0
0 2 4 6 8 10 12 14 16 18 20
L = 40° EAST
H = 36.05 NM
δ = 103°
DELTA VELOCITY (FPS)
(BUDGET)
800
700
600
500
400
300
200
100
0
ΔV RESCUE
ΔV BRAKING
MAX BRAKING ΔV
IN INTERVAL
0 2 4 6 8 10 12 14 16 18 20
CSM LEAD ANGLE AT RESCUE
INITIATION (DEG)

FIGURE 4

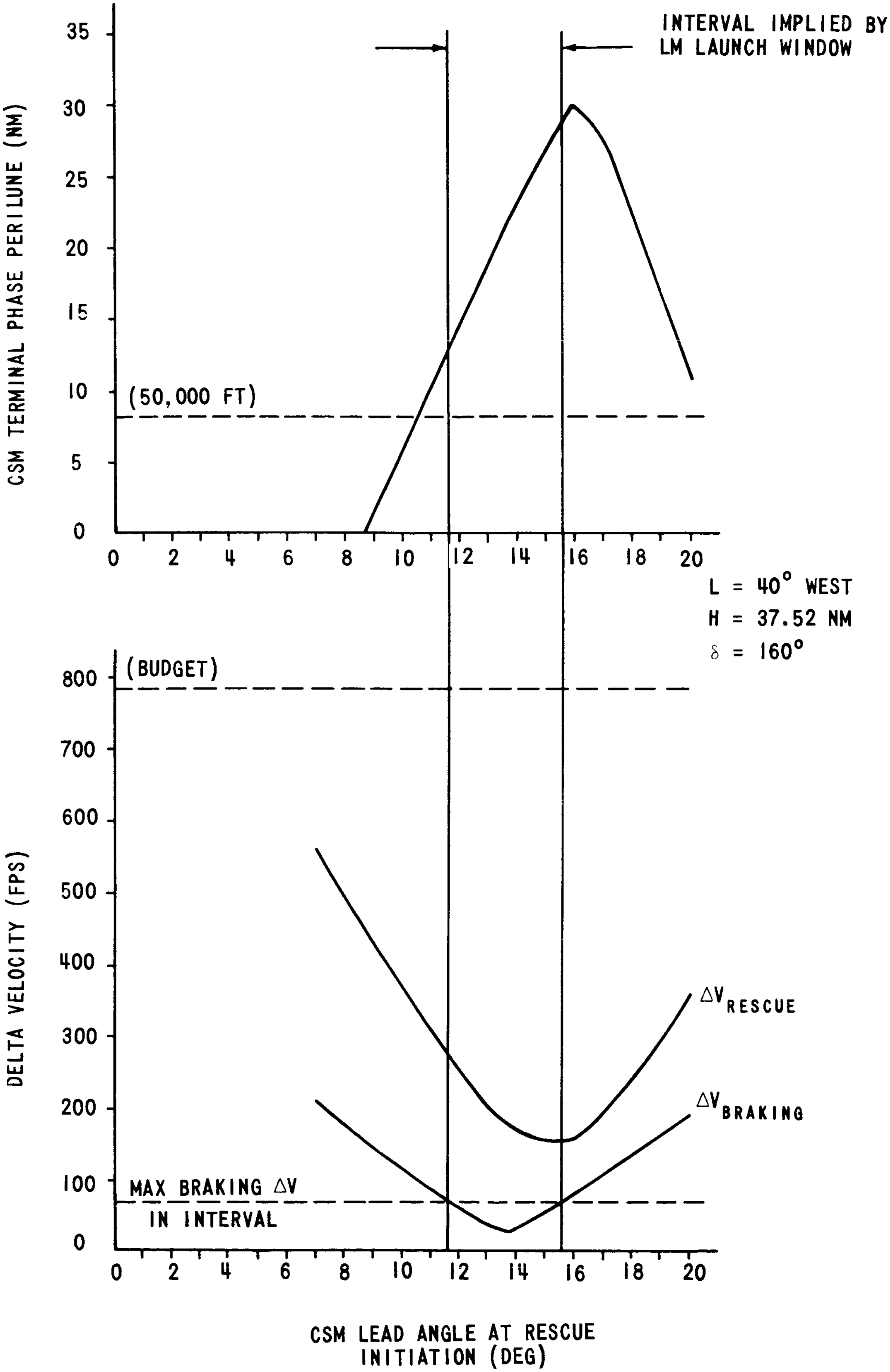

INTERVAL IMPLIED BY
LM LAUNCH WINDOW
CSM TERMINAL PHASE PERILUNE (NM)
35
30
25
20
15
10
5
0
(50,000 FT)
0 2 4 6 8 10 12 14 16 18 20
L = 40° WEST
H = 37.52 NM
δ = 160°
DELTA VELOCITY (FPS)
800
700
600
500
400
300
200
100
0
(BUDGET)
ΔV RESCUE
ΔV BRAKING
MAX BRAKING ΔV
IN INTERVAL
0 2 4 6 8 10 12 14 16 18 20
CSM LEAD ANGLE AT RESCUE
INITIATION (DEG)

FIGURE 5

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Simplified Navigation Technique Analysis - Case 310

DATE: April 3, 1968

FROM: W. O. Covington

ABSTRACT

This memorandum describes a simplified navigation technique which was one result of a recent Bellcomm study on the feasibility of reducing the complexity of the Apollo flight computer program. The simplified filter which would replace the Kalman Filter uses a set of preselected constant weighting factors to incorporate measurement information into the state vector estimate rather than the sequentially determined optimal weighting matrix of the Kalman Filter. Lunar rendezvous computer simulations of the simplified technique have indicated that acceptable state vector estimation errors result, for the limited set of conditions studied.

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Simplified Navigation Technique Analysis - Case 310

DATE: April 3, 1968

FROM: W. O. Covington

MEMORANDUM FOR FILE

DEVELOPMENT OF EQUATIONS

The equation for the simplified filter can be developed with the aid of the sketch of Figure 1 which shows actual and estimated positions of the CM and the LM. The estimated and actual relative position vectors of the LM with respect to the CM are given by RE and RM respectively. From this sketch the following vector equation can be written,

$$RE + (LMA-LME) = (CMA-CME) + RM$$

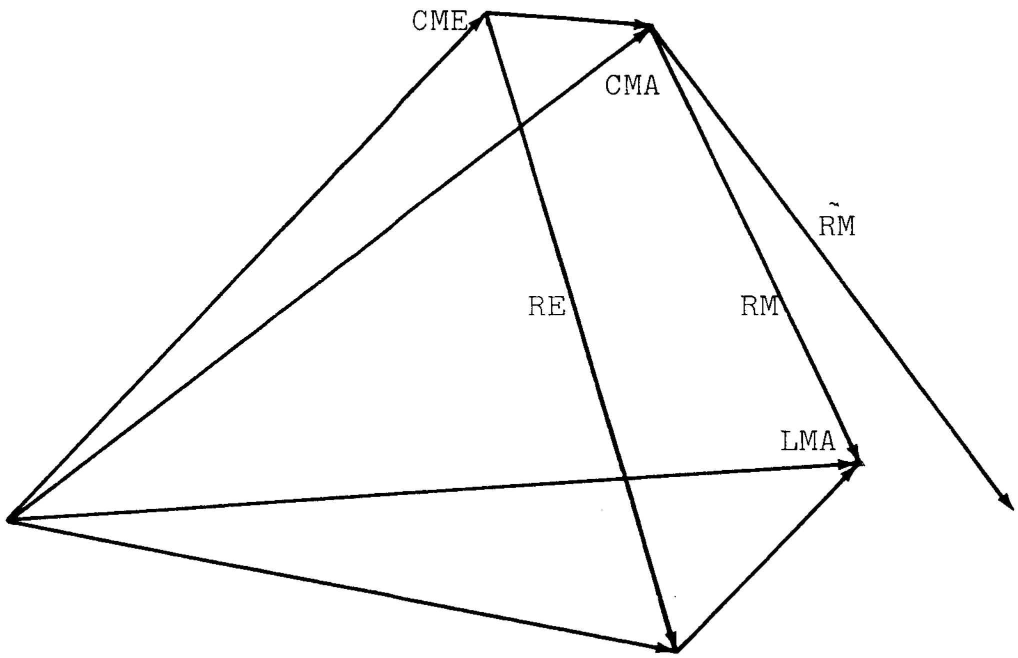


FIGURE 1 - GEOMETRY OF THE SIMPLIFIED FILTER

CMA = CM actual position
CME = CM estimated position based on prior measurements
LMA = LM actual position
LME = LM estimated position based on prior measurements
RM = relative vector between actual CM and LM positions
RE = relative vector between estimated CM and LM positions
$\tilde{RM}$ = measured relative state vector (components which are not measured are estimated)

Solving for the actual LM position

$$LMA = (CMA-CME) + LME + (RM-RE)$$

If the RM vector could be measured without error, this equation could be used to determine the position of the LM with an error corresponding to that associated with CM position vector.

Assume now that the CM position is known precisely (either as an assumption or from other data sources such as MSFN); the equation reduces to

$$LMA = LME + (RM - RE)$$

Since the RM vector cannot be measured precisely, only a portion of the measurement is used in the simplified navigation technique to update the position estimate. The sketch of Figure 2 indicates the way in which the LM state vector estimate is updated.

The equation by which new measurement data is incorporated into the position estimate is:

$$LME_{+} = LME_{-} + K_1 (\tilde{RM} - \hat{RE})$$

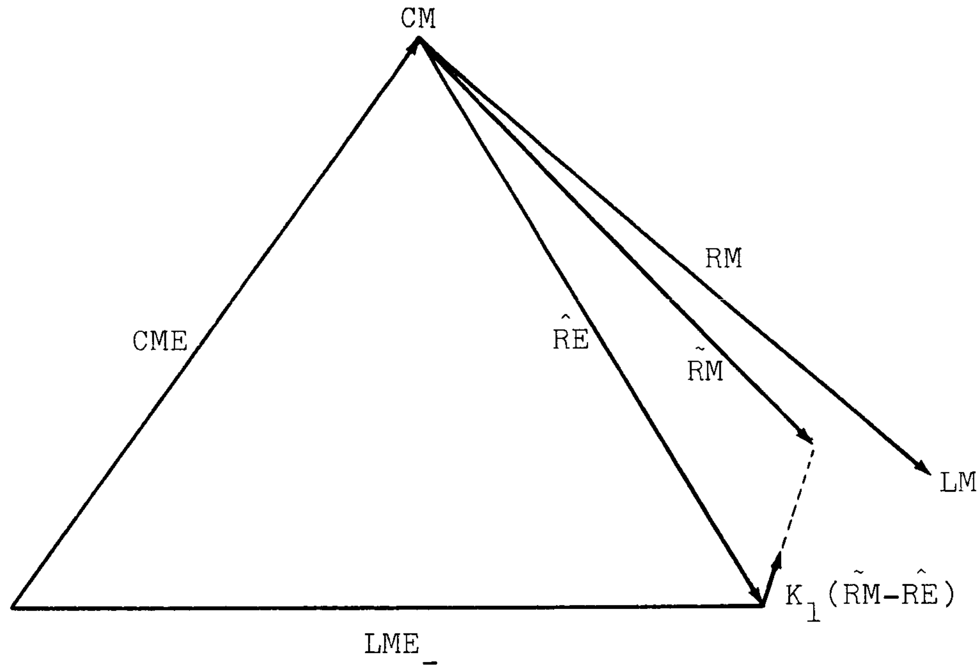
where

LME_{+} = LM state vector estimate after incorporating measurement

LME_{-} = LM state vector estimate prior to incorporating measurement

$\hat{RE}$ = estimate of relative state vector based on prior measurement data

$\tilde{RM}$ = relative state measurement vector (components which are not measured are estimated)



$$LME_{+} = LME_{-} + K\,(\tilde{RM} - \hat{RE})$$

FIGURE 2 - SKETCH OF THE WAY THE LM STATE VECTOR IS UPDATED

K_1 = Gain constant of filter, used as 0.5

The relative state estimate is determined by

$$\hat{RE} = LME_{-} - CME$$

The state vector estimates of LM and CSM can be propagated to each new navigation point by any convenient technique. In the current simulation they were numerically integrated. The velocity vector updates are handled in one of two ways depending upon whether a component of velocity is sensed. If no component of velocity is sensed, the velocity update equation is:

$$V_{REL+} = V_{REL-} + \frac{K_2}{\Delta t} \left[(\tilde{R}_{REL} - \hat{R}_{REL-})_t - (\tilde{R}_{REL} - \hat{R}_{REL+})_{t-\Delta t} \right]$$

Where $K_2 = 0.1$ was used in the short study.

If a component of velocity, such as range rate, is sensed, the velocity update equation has the same form as the position update equation,

$$\hat{V}_{\substack{REL+\\LOS}} = \hat{V}_{\substack{REL-\\LOS}} + K_3 (\tilde{V}_{\substack{REL\\LOS}} - \hat{V}_{\substack{REL-\\LOS}})$$

i.e. the velocity component parallel to the measurement direction is updated. $K_3 = 0.5$ was used.

In practical navigation measurements, all components of the RM may not be measured. Components of the relative state vector which are not measured must be estimated. For example, rendezvous radar which measures range and range rate is assumed to point in the estimated direction from LM to CM. These estimated components are therefore the same as corresponding components in the $\hat{R}E$ vector. If radar aiming angles are known, then all three components of RM can be computed from the measurements.

The value of the gain constant K is chosen to be as large as possible without causing intolerable errors in the state vector estimate. Clearly, larger values of K will cause the filter to converge more rapidly but at the same time will inject large portions of the most recent measurement error into the state vector estimate. The values selected for K depend upon the magnitude of the measurement error relative to tolerable errors in the state vector estimates. If range only is measured, $K_1 = .5$ is selected for position components, whereas $K_2 = .1$ is selected for velocity components.

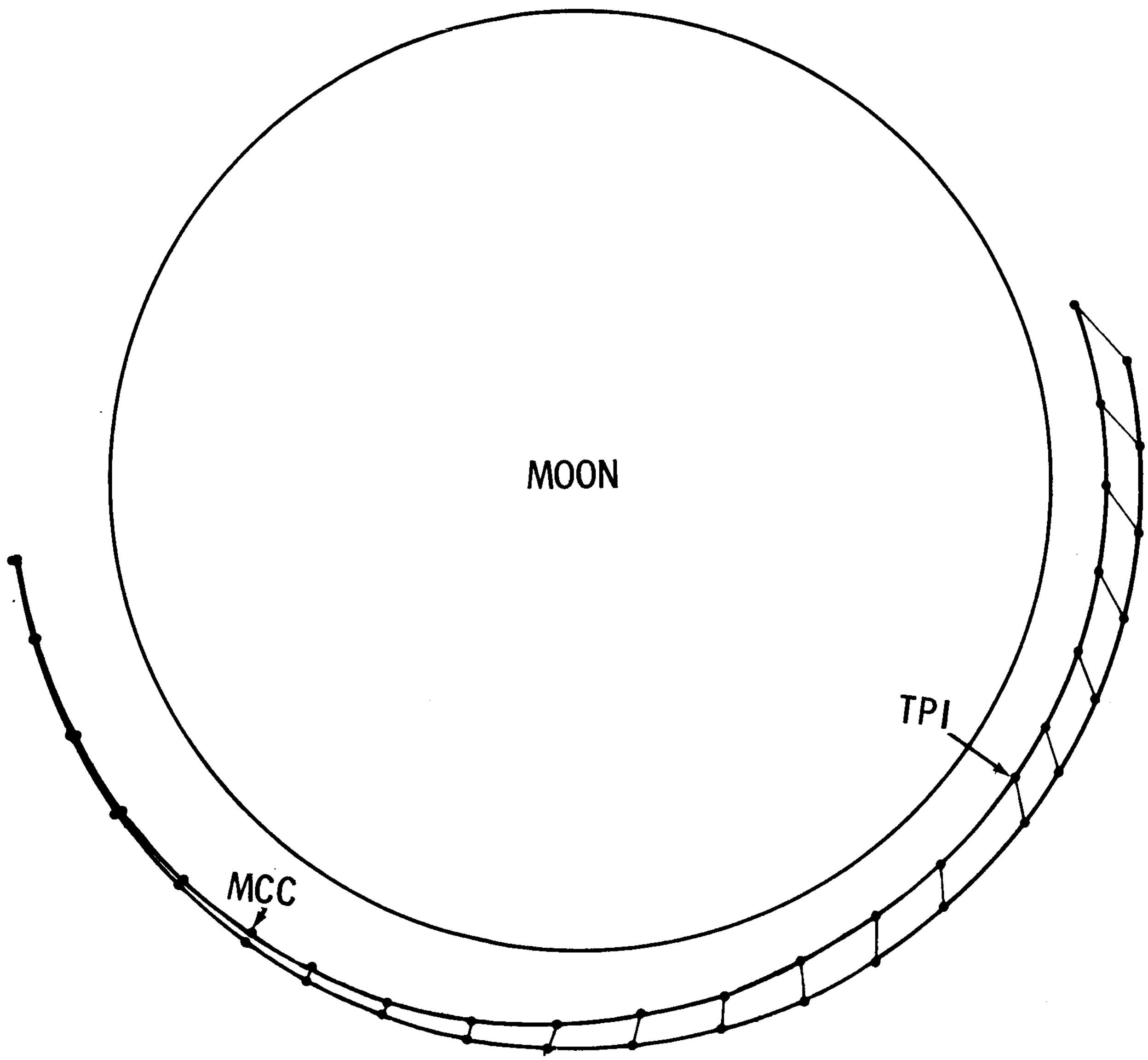
Simulation

To assess the performance of the simplified navigation filter, the LM and CM with navigation sensor measurements were simulated over a portion of the DRM IIA lunar rendezvous sequence. The simulation as sketched in Figure 3, started immediately after LM CDH and included TPI and MCC burns before ending at closest approach rendezvous. The navigation measurement schedule, as sketched in Figure 4, began with a measurement 180 sec after CDH and included one navigation point per minute until TPI. Computer runs were made for each of the following sensor combinations:

(1) RR PNGCS (2) RR AGS (3) SXT with VHF (4) SXT only

FIGURE 3

TRAJECTORY USED IN NAVIGATION STUDY



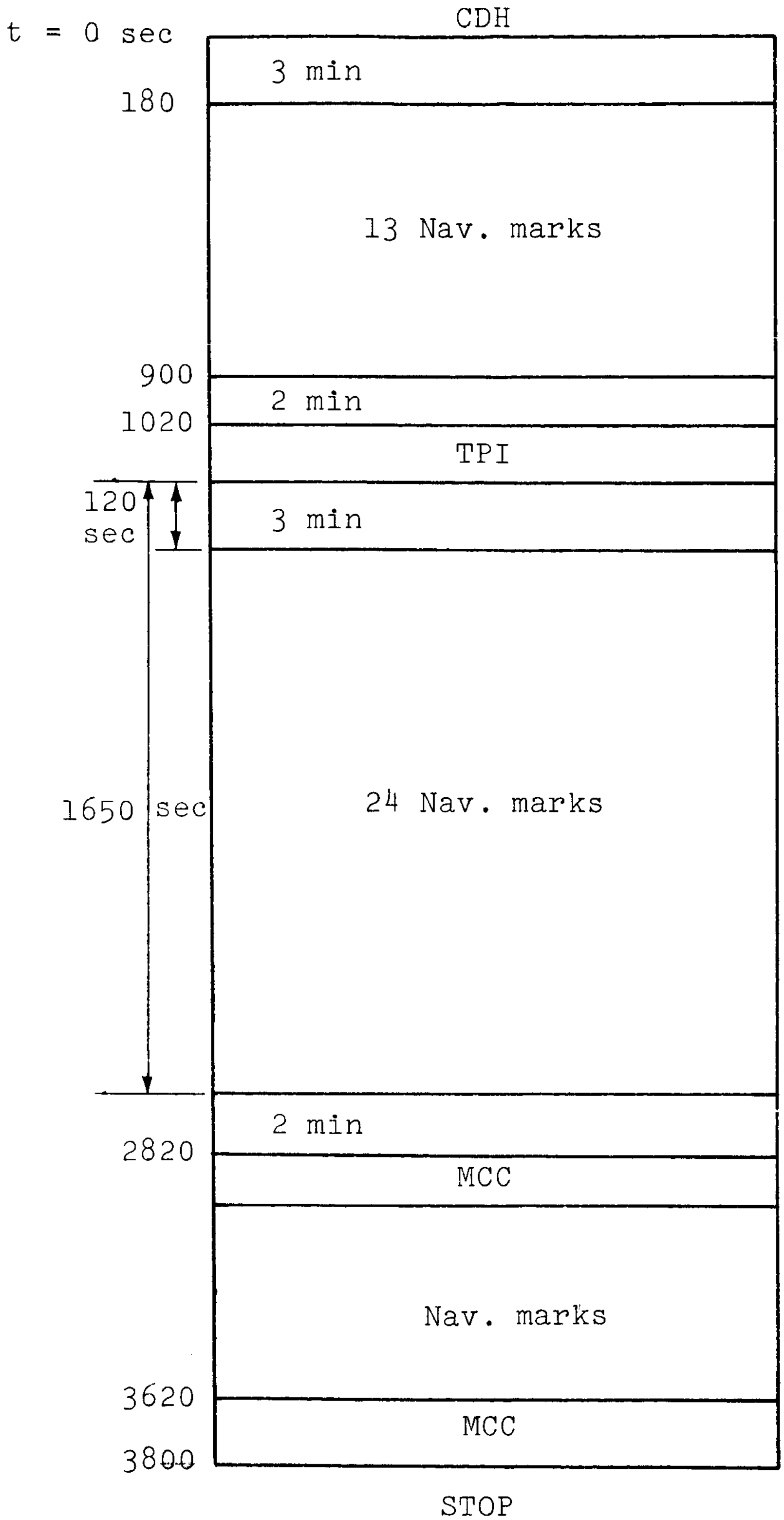


FIGURE 4 - TIMELINE FOR SIMPLIFIED FILTER SIMULATION RUNS

Two types of runs with navigation errors were made: Bias and Random. A perfect navigation run was made for comparison purposes.

Bias Runs:

> Initial condition deviations from nominal were inserted for LM and CM position and velocity. Measurement errors had non-zero means but zero covariance matrices. Components of the measurement error biases were set equal to the corresponding 1σ measurement error used in the Random Runs.

Random Runs:

> Initial condition deviations from nominal were inserted for LM and CM position and velocity exactly as for Bias Runs. Measurement errors were unbiased, uncorrelated Gaussian random variables, i.e. zero means and diagonal covariance matrices.

Each Random Run was essentially a single Monte Carlo run with non-randomly perturbed initial conditions. For both run types, the initial LM and CSM state vectors were perturbed non-randomly in opposite directions. A summary of the significant characteristics of each run type is shown in Table I. The measurement errors which were used are shown in Table II with the data sources referenced. These errors are RSS's of hardware biases and hardware 1σ random errors and, where appropriate, crew reading or pointing error. For Bias Runs, the values shown in Table II were used as bias errors; for Random Runs the same numerical values were used as 1σ zero mean uncorrelated Gaussian random errors.

The navigation errors associated with simplified navigation, as shown in the simulation position and velocity error curves of Figure 5, are tolerable. In this case the rendezvous radar measuring range and range rate was used with the measurement bias error sources indicated on the figure. An initial position error of 3 NM was reduced below 1/2 NM, whereas an initial velocity error of 31 fps was reduced to about 2 fps.

The miss distance at closest approach and the velocity, ΔV, added at the TPI and MCC burns is shown in Table III for simulation runs of various sensor combinations. The sextant only, which produces miss distances greater than 2 NM, is inadequate for navigation measurements when using the simplified navigation technique. The remainder of the

TABLE I

CHARACTERISTICS OF BIAS AND RANDOM RUN TYPES

RUN TYPE	MEASUREMENT ERROR MEAN AND COVARIANCE MATRIX	INITIAL CONDITION OF STATE VECTOR MEAN AND COVARIANCE MATRIX
BIAS	$\mu_m \neq 0, \ \Sigma_m = 0$	$\mu_s \neq 0, \ \Sigma_s = 0$
RANDOM	$\mu_m = 0, \ \Sigma_m =$ [Diagonal]	$\mu_s \neq 0, \ \Sigma_s = 0$

TABLE II

MEASUREMENT ERRORS

ERROR	UNIT	VHF SXT	RR PNGCS	RR AGS
Range Bias	Feet	100.[4,p.2]	500.[1,p.56]	3000.[3,p.13]
Range Scale Factor	None	.005*	.0008[1,p.56]	.0008[3,p.13]
Range Rate Bias	fps	0.	1.0[1,p.57]	1.2**
Range Rate Scale Factor	None	0.	0.[1,p.57]	0.[1,p.57]
Angle Error (Each Axis)	deg.	.0115	.9[2]	1.0***

*Reference 4, page 2 states .002, however this more conservative estimate was used.

**Includes 1 fps bias error (Reference 1, page 57) and an estimated 0.5 fps tape meter reading error.

***Include 0.9 deg. bias error (Reference 2) and an estimated 0.5 deg. crew pointing error.

FIGURE 5

POSITION & VELOCITY ERROR AFTER NAV. UPDATES
PNGCS, RENDEZVOUS RADAR, R & $\dot{R}$

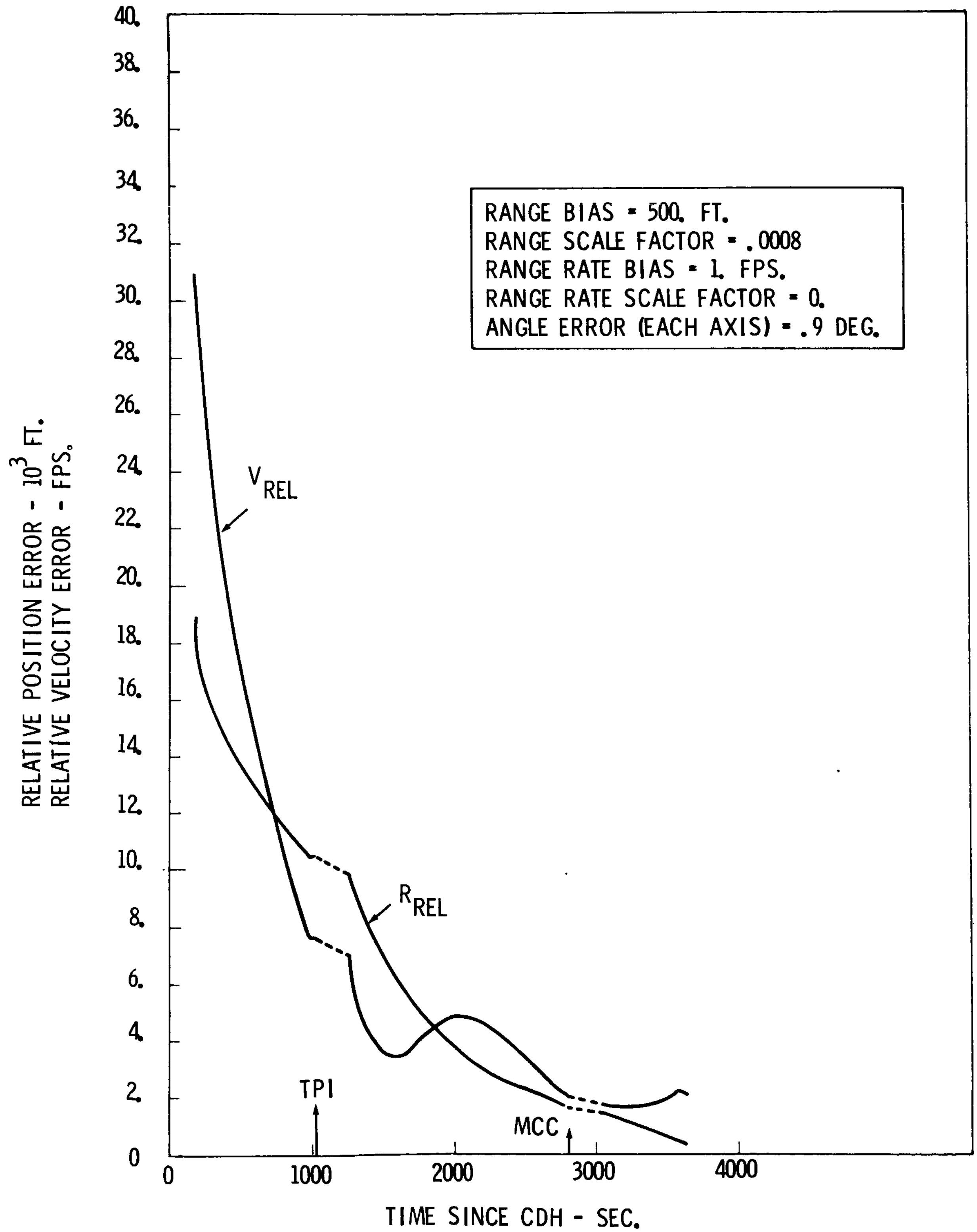


TABLE III
MISS DISTANCE AND INCREMENTAL VELOCITY FOR SIMULATOR RUNS

TYPE	MISS DISTANCE (FEET)	ΔV (TWO BURNS)
Perfect Navigation	484	158.20
PNGCS RR Bias	2611	188.45
AGS RR Bias	3356	185.35
SXT + VHF Bias	3812	175.73
SXT ONLY Bias	167,320	175.66
PNGCS RR Random	3863	83.33
AGS RR	2014	89.14
SXT + VHF Random	3480	95.23
SXT ONLY Random	12,601	107.91

BIAS means +6000 ft. + 10 fps each axis for CM, -6000 ft. - 10 fps axis for LM from nominal trajectory initially, plus (σ bias in instruments)

RANDOM means all of the biases become random - instrument errors are random at each measurement.

sensor combinations produces miss distances of approximately 1/2 NM or less and appears to be adequate as a source of input data for the simplified navigation filter.

It should be pointed out that the simplified filter appears to work well for measurement bias errors; it has been recognized that the Kalman filter tends to diverge if non-zero measurement biases are not included in the state vector. For this situation of biased measurements, the simplified filter appears to perform better than the Kalman filter.

Further analysis and simulations are being made to assess the performance of the simplified filter; this study reports only preliminary information for the limited situations studied.

W. O. Covington

2014-WOC-nr

W. O. Covington

Attachment
References

REFERENCES

1. LSP-370-2E Rendezvous Radar, Transponder and Landing Radar Sections, Guidance, Navigation and Controls Subsystem, Design Control Specification for (U), Grumman Aircraft Engineering Corporation, February 1, 1967.

2. "MIT Presentation of Lunar Module Rendezvous Radar Performance," MSC, September 28, 1967.

3. "LEM/AGS Performance Analysis Test Results Design Mission Computer Program," TRW No. 05952-6058-T000, TRW Systems, November 1966.

4. Memorandum from R. R. Gilruth to Major General S. C. Phillips, "Addition of Ranging Capability to the CSM-LM VHF Voice Data Systems," September 1967.

ATTACHMENT 14

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: CM and LM AGC Computer Programs

DATE: March 1, 1966

FROM: W. C. Hittinger

Dr. G. E. Mueller/M:

Enclosed are copies of the vu-graphs covering our two-week study of the technical feasibility of reducing the size and complexity of the CM and LM AGC computer programs. Written documentation of the study will be furnished later.

We conclude, within the limitations of a very short study, that it is technically feasible to program the AGS within the spirit of the guidelines that you have established. You realize, of course, that the equations and programming philosophy have not been verified - time has permitted looking at only a few off nominal cases and the spectrum of possible abort cases has not been examined.

We are not in a position to evaluate the impact of a parallel development on the total program. It is clear that such a development can only be a backup - to meet the current flight schedules we must rely on the MIT effort. The other factors which must be considered are:

1. Because the AGC will be used, and to be efficient some of the current MIT routines should be used, there will certainly be some impact on the current MIT programming effort. To learn to program the AGC and to use the MIT routines will require some education of the backup contractor by MIT.

2. MSC may not be able to support a second software development in the areas of mission planning, requirements development, equation verification, data development and procedures.

3. Because the backup program will be based on a different philosophy than the current MIT program there will be an impact on crew training if a changeover should occur.

On the positive side it does appear that the program could be developed such that verification for flight could be simpler. If the program size can be held at 18K words then room for future growth to handle the extended Apollo missions would exist. However, historically programs have grown in size during the development phase. MIT originally estimated 4K for the AGC program. Extremely tight control will be required to maintain the backup at 18K.

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Thus the Bellcomm position is that a backup program meeting your guidelines is technically feasible. Because of the impacts noted above we cannot assess the total value to the space program of going ahead with such a development.

W C Hittinger

W. C. Hittinger

WCH:ldh
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