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APOLLO

GUIDANCE, NAVIGATION AND CONTROL

E-2305

MAN/MACHINE ALLOCATION IN THE APOLLO NAVIGATION, GUIDANCE, AND CONTROL SYSTEM

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JULY 1968

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APOLLO

GUIDANCE AND NAVIGATION

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MAN/MACHINE ALLOCATION
IN THE APOLLO NAVIGATION, GUIDANCE, AND CONTROL SYSTEM

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Introduction

Man's role in spacecraft guidance and navigation is that of supervisor of automatic systems and performer of specific sensing and control functions (ref. 1). However, as technology improves in capability and reliability, many of these specific tasks will be replaced by automatic systems. It is unlikely, however, that man's unique flexibility and decision capability will be replaceable in the foreseeable future.

To serve effectively as a supervisor during critical mission phases, the human operator must continuously keep abreast of all vehicle systems and the environment. In the case of certain critical functions, several levels of back-up subsystems demand continual time-shared attention in order that the astronaut know the status of these systems when they are required. The total system must be designed to provide sufficient crew involvement in system functions to enable smooth and rapid transition to a back-up mode.

It is this requirement of continual crew awareness of and involvement in many levels of redundant systems, operating in parallel, which places a heavy burden on vehicle flight crews of aircraft and spacecraft.

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*Associate Professor, MIT

To illustrate, the nature of the astronaut's supervisory and functional tasks and the levels of system redundancy will be explored by reference to the rendezvous phase of an Apollo mission, including the required thrusting maneuvers.

The paper is organized as follows:

a. Description of the nominal lunar orbit rendezvous phase
b. Outline of the navigation process for rendezvous
c. Detailed functional time line for a CSM active rendezvous
d. Exploration of future man/machine allocation, relieving man of specific tasks and expanding his role as supervisor and decision-maker.

Lunar Orbit Rendezvous

For the basic lunar landing mission (Fig. 1) there are fifteen distinct guidance and navigation phases. The lunar orbit rendezvous phase starts when the powered ascent injects the Lunar Module into a safe perilune orbit. Termination of rendezvous occurs when the desired conditions for manual docking have been achieved.

In order to maximize crew safety, the Primary Guidance, Navigation, and Control Systems (PGNCS) on both the Lunar Module (LM) (Fig. 2) and Command Service Module (CSM) (Fig. 3) have been designed to give each vehicle independent "on board" capability (ref. 2, 3, 4 and 5). Fig. 4 is a functional outline of both systems. The systems have also been designed to complement one another. Thus, during the rendezvous phase when one vehicle is active*, the other vehicle is

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* That vehicle is active which performs the thrust maneuver.

constantly monitoring* and updating the state vector† of the active vehicle with data from its own sensors. In addition, thrusting maneuvers by the active vehicle are relayed over the voice link to enable updating of the active vehicle's state vector in the passive vehicle's computer. If problems occur, the vehicle roles (passive to active) can be reversed instantaneously and the rendezvous continued.

The techniques under active consideration for rendezvous are the Stable Orbit Rendezvous (SOR) technique and the Concentric Flight Plan (CFP) (ref. 8). SOR (Fig. 5) is a modified direct ascent rendezvous in that the ascent trajectory is controlled to intercept the orbit of the passive vehicle a given distance ahead of, or behind, the passive vehicle. Additional maneuvers place the active vehicle in the same orbit as the passive vehicle with the desired separation and control the final transfer from the stable orbit point to the terminal rendezvous point.

The CFP rendezvous profile, to which we will devote our attention in detail, is a special parking orbit concept as illustrated by Fig. 6. Initiation of the CFP occurs after the LM has been injected into an elliptical orbit, approximately 50,000 feet by 30 nautical miles, coplanar with the CSM orbit. Initiation is accomplished by a horizontal thrust maneuver approximately 90° (central angle) from the injection point. This Concentric Sequence Initiation (CSI) maneuver, raises the LM apolune to a new altitude at time t_2.

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* During these maneuvers the "ground" systems are constantly monitoring and supporting except for the period when the vehicles are behind the moon.

† Position and Velocity at a specified time.

When the LM reaches this time, t_2, its orbit is circularized by a constant delta height (CDH) maneuver. The LM stays in this new orbit until the line of sight to the CSM achieves a preset angle (approx. 26.5°) with the LM local horizontal plane. At this time, t_3, the transfer phase initiation (TPI) maneuver is performed, placing the LM on a direct intercept trajectory with the CSM. This intercept trajectory covers a central angle of approximately 140°. During this period, midcourse corrections may be made to insure intercept. A series of terminal rendezvous braking maneuvers are performed to establish the desired docking conditions.

Rendezvous Navigation Process (ref. 6, 7, 8 and 9)

In order to compute the ΔV* corrections to establish the proper intercept trajectory, an estimate of the other vehicle's position and velocity must be processed by the on-board PGNC system. The PGNCS carries out this navigation function in all phases of the rendezvous maneuver. The only difference between the two vehicle systems is that navigation data (range, range rate, and line-of-sight angle) is obtained by radar on the LM while only line-of-sight angle data is obtained from a precision optical system on the CSM. A range-only capability is being added to the CSM VHF communications system.

The rendezvous navigation concept is illustrated in Fig. 7. For initialization, each vehicle guidance computer must contain the following information: (a) the state vector for each vehicle (this data may be obtained from previous on-board navigation phases or from

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* velocity change, direction and magnitude

Mission Control via telemetry), and (b) predetermined stored statistical data for the uncertainties in the state vector and each navigation sensor, and a geometry vector ($\underline{b}$) which is a function of the type of navigation measurement.

The estimated LM trajectory ($\hat{\underline{R}}_L$, $\hat{\underline{V}}_L$) and CSM trajectory ($\hat{\underline{R}}_C$, $\hat{\underline{V}}_C$) are computed at designated times by integrating the equations of motion. A radar tracker in the LM tracks the CSM, yielding relative position and/or velocity data. The estimates of the LM position and velocity and the tracking measurement biases are improved by processing the tracking data through an optimum filter. Each measurement (e.g., range rate, elevation angle) is processed separately with the estimates updated after each processing. Measurements are typically processed every 60 seconds during the rendezvous midcourse and terminal phases. In this way the on-board estimate of the LM's position and velocity and the measurement biases are continually improved as the filter smooths the tracking data.

The filter consists of an optimum weighting vector $\underline{W}$, which, when multiplied by the difference between the actual measurement ($\tilde{\underline{Q}}$) and an estimate of the measurement ($\hat{Q}$), yields an estimate of the deviation ($\delta\ \hat{\underline{R}}$, $\delta\ \hat{\underline{V}}$) from the previous estimate of LM position and velocity. These deviation estimates are then added to the previous estimates to produce the improved LM position and velocity estimates, (R_L, V_L)as shown in Fig 7. The filter also produces a new estimate of the measurement bias if desired.

After each measurement processing, W is updated by subtracting the step change (δW) from the extrapolated value of W. During long rendezvous profiles, it is necessary for the astronaut to reinitialize the weighting vector computation (W matrix). Timing for this process is shown in Fig. 11 and 12B.

In the CMC rendezvous navigation program optical tracking data is typically processed once per minute during those phases in which the target vehicle can be tracked. If the magnitudes of the changes in the estimated position and velocity vectors, δr and δv respectively, are both less than preset update alarm levels, the selected vehicle's state vector is automatically updated by the computed deviation, $\delta \underline{x}$. If either δr or δv exceeds its alarm level, the state vector is not updated, and the astronaut is alerted to this condition by a display of δr and δv. In this case, the astronaut should recheck the optical tracking and verify that he is tracking the target vehicle. Under certain conditions it is conceivable that a star could be mistaken for target-reflected sunlight, and it may take a few minutes to identify positively the target vehicle by watching the relative motion of the target and star background in the optics field of view. The CMC automatically points* the optics along the estimated line of sight, so the object that generally follows the optics reticle is the orbiting target, while star images will drift across the optics field. After tracking has been verified, the astronaut has the option of commanding a state vector update if the tracking alarm is exceeded, or of repeating

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* When the crewman desires to point the optics exactly at the target vehicle he does so by changing the optics mode to manual.

optical checks before incorporating the measurement data. Once the target has been positively identified, the state vector update should be commanded regardless of the tracking alarm. The primary purpose of the tracking alarm is to avoid false target acquisition and tracking, and only alerts the astronaut that the state vector update is larger than normally expected. The alarm level can be adjusted in flight. At present it is set to zero to maximize the crew's ability to monitor the rendezvous navigation process.

As shown in Fig. 7, either of the two state vectors may be updated by the tracking process. This option is normally chosen when the rendezvous navigation program is first called, but can be changed at any time by the astronaut. The vehicle having the larger uncertainty in its initial state vector is the one normally chosen for updating. If the relative accuracy of the two state vectors is unknown, the passive vehicle is usually updated.

During an active CSM rendezvous, the CSM state vector is automatically updated during powered rendezvous maneuvers by a routine called Average-G which integrates the output of an orthogonal set of accelerometers mounted on an inertially stabilized platform. If the CSM is monitoring a LM-active rendezvous, the LM state vector in CSM computer memory is updated after a rendezvous maneuver as an impulsive ΔV by a special entry* into the CSM computer display-keyboard (DSKY) during which time optical tracking data is suspended

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*(Maneuver ΔV input of Fig. 7).

In review, the five major operations required of the astronaut during the CSM rendezvous navigation are:

1. Optical tracking of the target vehicle, which includes initial target acquisition followed by a uniform optical tracking and marking operation.
2. State vector update monitoring.
3. Vehicle update option.
4. Input target ΔV maneuvers.
5. Weighting vector reinitialization.

Functional Time Line of the CSM Active Rendezvous

Of the two possible operating modes, CSM active or LM active, the former is the more interesting, for this discussion, because the CSM is manned by only one crewman during rendezvous, whereas the LM has two crewmen. Therefore, in order to illustrate adequately the role of the man as a supervisor and monitor, we shall limit the remainder of our discussion to a CSM active rendezvous from the CDH burn to final braking phase.

During rendezvous, the crew must activate, control, and monitor spacecraft systems for prime and backup navigation, thrust guidance, and attitude control. Vehicle housekeeping systems such as environmental control, fuel cells, and communications must also be monitored.

1. Navigation

a. Primary

The navigation portion of the PGNCS is shown in Fig. 8. Navigation data, in early missions, will be derived from two optical devices, a Scanning Telescope (SCT) and a Space Sextant (SXT). In later missions range data will also be available for navigation. The SCT is a unit power, 60°-field instrument, used for acquisition and pointing. The SXT is a two-line-of-sight, 28X, 1.8°-field instrument used for obtaining high accuracy pointing data. Navigation angle data derived from these units is referenced to the inertial measurement unit (IMU), a three-gimbal, inertially-stabilized platform, which is the primary attitude reference. When the astronaut optically acquires and accurately points the optics at the target vehicle, he signals this event to the Command Module Computer (CMC)* by activating the optics mark computer discrete. This discrete causes

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* The PGNCS Command Module Computer (CMC) is the primary onboard sequence controller. It monitors the sensors' (optics, accelerometers, IMU gimbal angles) information and determines thrust times and vectors, vehicle trajectory parameters, and optics target lines of sight, maintains attitude control, and guides the CSM during thrust maneuvers. The CMC and crew interface at the display/keyboard (DSKY), which consists of a numeric keyboard and electroluminescent digital displays. The CMC sequences are separated into functional blocks called programs (Fig. 9). Two kinds of codes are used for crew/CMC communication: (a) verbs, indicating the kind of action to be taken, and (b) nouns, defining the data to be processed or displayed. Some of the most-commonly-used verbs and nouns are listed in Fig. 9.

the CMC to read the optics and IMU angles and the time of mark. The computer compares the measured target elevation with the predicted value and weighs the data with the W matrix (Fig. 7) to derive vehicle state vector update parameters, δ r and δ v.

b. Backup Navigation

Backup rendezvous navigation has two modes, one for long distances and one for close distances. For long distances, navigation data is determined from ground tracking. Orbital maneuvers for rendezvous are targeted from the ground and accomplished by crew performance of fixed attitude burns using the backup guidance systems.

At close distances (determined by the maximum unaided target visibility) elevation angle data is taken by boresight fixes along the CSM longitudinal axis with the COAS*, a single-power collimated reticle attached to the commander's window. The COAS is used for monitoring the primary system, and for backup rendezvous navigation sightings. The COAS can also be used to align the IMU in a degraded mode.

(continued from previous page)
There are designed-in "holds" in the program sequences to permit crew review of all calculations and operations before enabling execution. Fig. 10 indicates the general nature of the crew/CMC interface. This design permits the crew to exercise complete sequence control, but also causes the crew to monitor the CMC continuously during a mission operation.

* Crew Optical Alignment Sight

For navigation the crewman measures the target vehicle elevation angle above, or below, the local horizontal as a function of time. Range, range rate, and the maneuvers required to achieve intercept are obtained from charts. During the final braking phase the crewman employs a fixed braking schedule based on range, starting at about one mile. Range is determined by the target vehicle angle subtended on the COAS reticle.

2. Attitude Control

Primary attitude control is achieved with a CMC digital autopilot (DAP) which maintains attitude rates and deadbands, and calculates attitude maneuvers based on input values from thrusting directions determined by the CMC targeting programs, by ground targeting, or from the crew. Manual control with the rotation hand control (RHC) through the DAP, the spacecraft Stabilization and Control System (SCS), or directly to the spacecraft reaction control system (RCS) jet solenoids are three levels of redundant attitude control. Manual attitude control through the DAP can be performed in fixed-rate or acceleration modes with the RHC, and an attitude impulse mode (minimum RCS jet firing time) with a pencil-stick minimum-impulse control. RCS manual attitude control through the SCS may be performed in proportional-rate, minimum-impulse, or acceleration modes.

Spacecraft attitude is displayed with respect to the inertially referenced IMU on one of two Flight Director Attitude Indicators (FDAI) which display total attitude, attitude error, and spacecraft angular rate. The backup attitude reference consists of two sets of

BMAG's* (one set for angular rate and one for attitude) and an Euler angle computer called a gyro display coupler (GDC). The GDC is either initialized from the IMU via the DSKY or by the crew using the COAS and stars. A local horizontal attitude reference is available from ORDEAL** which drives the FDAI display in pitch to indicate vehicle attitude with respect to local horizontal.

3. Guidance

Primary guidance is achieved by a CMC thrust vector control (TVC) DAP which controls the RCS roll jets and the Service Propulsion System (SPS)*** gimbal drives. Both fixed and variable attitude guided propulsion maneuvers can be performed by the CMC. Fixed attitude burns may also be made automatically by using the SCS system, or manually through the SCS system in either a rate damped or acceleration modes by RHC control of the SPS gimbals. For these backup TVC modes, thrust is monitored by a single accelerometer mounted on the spacecraft longitudinal axis. The change in spacecraft velocity is displayed to the crew via the backup ΔV counter which is part of the Entry Monitoring Systems (EMS).

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* Body-mounted attitude gyro

** Orbital Rate Display, Earth and Lunar

*** There are two propulsion systems on the CSM. The Service Propulsion System (SPS) is an approximately 20,000 pound thrust, gimballed engine for large translation maneuvers. The reaction control system (RCS) consists of four quads of 100 pound thrust jets. The RCS is used for translations too small for efficient use of the SPS and attitude control.

4. Integrated Crew Functions

There are two stations in the CSM for crew operations. The primary station is at the Main Display Console (MDC) (Fig. 3), located in front of the astronaut couches. Located on the MDC are all the PGNCS, SPS, RCS and SCS switches and controls, a DSKY, the two FDAI' s,and controls for power, environment and communications. The other station is the Lower Equipment Bay (LEB). Here the PGNCS prime sensors, switches and controllers for manually operating the optics, a second DSKY,and auxiliary mounts for the RHC and THC are located. All optics sightings are performed and RCS attitude and translation maneuvers may be executed in the LEB. This arrangement necessitates movement between the MDC and LEB as operations demand (Fig. 12).

Figure 11 is a relative position plot of the Active CSM and passive target during the rendezvous transfer phases. The major navigation and maneuver activities are indicated as a function of time from docking. Figure 12 is a functional time line showing crew activities, DSKY operations and backup system interfaces. During each step, prime and backup systems are initiated and updated in parallel. When the IMU is aligned, for example, the backup attitude gyro package is initialized (periodically it has to be re-aligned to compensate for system drift). Peak activity periods center around the thrusting maneuvers. Satisfactory thrust execution and rapid verification of the expected effect on the rendezvous are prime requirements. As intercept approaches, increasingly manual performance is manifested by increased reliance on desired range and range rate by crew monitoring of target elevation and time, correlated with onboard charts.

The attitude reference for the IMU is established first by optics sightings on two of the thirty-seven navigation stars whose coordinates are stored in the CMC. The transfer-phase conditions are set up by a CDH burn using the CMC targeting program (P-30) and SPS thrust maneuver program (P-40). (The nominal CMC/Crew sequence for these programs is shown in Appendix A.) This sequence places the orbits at a constant-differential height, permitting the CSM to catch the target at a prescribed rate.

The rendezvous navigation CMC program (P-20) (App. A) is selected to maintain the required CSM attitude for optical and radar transponder coverage of the target and to process optics data for state vector updating. Trajectories are defined and sightings are scheduled to maximize target visibility during rendezvous tracking. For the nominal plan, tracking from TPI to intercept is done in darkness and intercept occurs in daylight just after sunrise. Sightings in darkness rely upon a high-intensity flashing beacon mounted on the target. Upon initiation of P-20 the crew specifies which state vector is to be updated, normally that of the passive vehicle. This choice may be changed at any time during the rendezvous by keying in a special DSKY code. The CMC maneuvers the spacecraft to the tracking attitude and automatically points the optics at the target vehicle. Acquisition of the target may be monitored at this time by the crew as described in the first section.

After the P-20 maneuver, the transfer phase initiation (TPI) targeting program (P-34) (App. A) is selected to specify the time of ignition for a target elevation angle at TPI. (At crew option, the TPI solution can be specified to be ignition time instead of elevation angle.)

The duration of the transfer time from TPI to docking is also specified by the crew and the CMC calculates and displays the velocity increments for the TPI and transfer phase final (TPF) maneuvers. Also displayed are post-TPI perigee, time from ignition, and the number of navigation sighting marks made during this sequence. This computation cycle may be executed several times concurrently with the navigation mark process. When the crew desires sighting marks, the optics are enabled to acquire the target automatically. Upon acquisition, the navigator performs manual centering of the target in the optics field. For monitoring purposes, range, range rate, and the target elevation DSKY display may be requested by the crew and compared with the backup charts. As new navigation data is incorporated, the computed values displayed in P-34 will change to reflect this new information.

During the TPI thrust program (P-41) (App. A) the CSM is maneuvered to the thrust attitude. The CMC monitors thrust and calculates and displays the new CSM orbital parameters. The crew is responsible for monitoring attitude and adding the correct ΔV with the translation hand control at the specified time of ignition.

After the burn, further comparison between the CMC target variables (range, range rate, and elevation angle) and COAS target elevation data are made. The rendezvous tracking attitude is reestablished and further optics marks are made in parallel with execution of the TPM targeting program (P-35). Target data are periodically monitored and the transfer phase midcourse (TPM) sequence (P-35, P-41) is performed to minimize dispersions at the specified docking time. If, however, the indicated ΔV corrections from P-35 are very small, this sequence would be omitted. P-35 calculates the thrust vector at an ignition time which is a fixed time delay after program selection. Time from ignition, ΔV values for the midcourse and TPF thrusts, and the post-TPM burn perigee are displayed. P-41 is repeated in the same manner as for the TPI burn.

During this final phase, more reliance is made upon out-the-window monitoring of the target. The residual maneuvers and docking are performed manually without onboard computation assistance.

As the time to TPF is counting down, the crew prepares to burn the RCS manually for the ΔV calculated in P-35. The CSM is maneuvered so that the longitudinal axis is directed at the target, permitting out-the-window monitoring during thrust.

The PGNCS burn monitor program (P-47) is called to display the TPF ΔV along spacecraft axes to verify crew THC ΔV inputs and to enable update of the CMC state vector.

Summary

The crew functions illustrated by Fig. 12 (A, B, and C) can be categorized as follows:

a. Monitoring of and decision making associated with the rendezvous navigation process including the effects of target sighting data on state vector updates and comparison of onboard data with backup charts and ground tracking.

b. Sequencing and initialization of primary guidance, navigation, sensing systems, propulsion and timing systems.

c. Initialization and sequencing of backup systems.

d. Monitoring of the spacecraft housekeeping systems. Note: This function is not shown in Fig. 12 but it is implicit at all times.

Approximately 70 crew functions can be identified, for item a, b, c above. These functions range from a complete IMU alignment sequence to monitoring an automatic spacecraft maneuver. Of these functions approximately half are associated with items b and c. Approximately 25 - 40% of the crew work load is devoted to these items. More exact figures await full mission simulation on the NASA Apollo Mission Simulator tied into the Mission Control Center and actual earth orbit simulations of these lunar orbit mission phases.

Future Trends of Man-Machine Allocation

The advances in component technology coupled with the requirement to increase the efficiency and reliability of each mission (flight), be it airplane or spacecraft, have resulted in continuous pressure to increase the number and complexity of functions performed in the cockpit. For example, Fig. 13 and 14 from ref. 10 detail the increase in cockpit functions from the DC-3 to the DC-9 and from Mercury to Apollo.

To view the man/machine development during this period with some perspective, consider the following model - a fully automatic machine or robot, capable of accomplishing Apollo or performing SST functions without man. At the very least, this mechanism would have large spectral bandwidth with extremely low threshold, high signal-to-noise ratio, great flexibility in choice of signal processing, associated adaptive dynamical systems, the capability to perform many tasks in parallel, and sufficient reliability to achieve efficient, low-cost missions. A machine of this nature does not now exist, nor will it for at least a decade or so. We make up for this lack, interestingly, by using a low bandwidth, low signal-to-noise, serial, but highly-flexible, processor (man) to serve as performer of unique sensing tasks and as monitor and supervisor of redundant systems. Thus, man/machine design has meant utilizing man to fill technological gaps which limit the desired system capability, flexibility, and reliability. At the existing level of technology, such design activity is oriented toward integrating redundant systems that individually do not have the required inherent reliability and/or capability to achieve mission objectives.

The major difficulty of this technique is indicated in Fig. 13 and 14 . How many functions that interface directly with man can we add to the cockpit load before chaos results? Fortunately the next generation of airborne computers and integrated display techniques appears to offer hope of reducing this trend.

Large scale integrated circuits coupled with the multiprocessor approach (ref. 11 and 12) will allow a central processor design with capability and reliability high enough to allow man to be removed from continual direct control over the multiplicity of subsystems. This technique should be a strong tool for reducing crew workloads as follows:

1. Minimize the details of vehicle pre-flight checkout. All systems should be capable of being selected and sequenced by the computer. Thus, automatic checkout and sequencing may be employed in one of two modes:

a. All checkout is performed automatically; man might be required only if some system could not pass the required tests.

b. Certain sequences could be performed cooperatively; i.e., the computer would do the sequencing, but at each point in the checkout where a measurement is made, the computer would hold the sequence until the crew verified the data. This method would probably be used on mission- or safety-critical systems. The crew could also have the option of changing the cooperative system checkout list in flight.

2. Maximize crew safety during mission-critical phases by automatically initiating all redundant systems. During these periods, the central processor would continuously monitor all active systems for failure or marginal performance and would keep the crew informed. Switchover from a marginal or failed system to a good system could be automatic or manual depending on crew option.

3. Maximize information availability while minimizing display space by utilizing integrated general-purpose graphical displays. These flexible graphical display devices may be used:

a. To reduce the total number of individual displays by time-sharing. For example, during boost, entry, and thrust vector control, data may be displayed which are relevant and/or unique to that mission phase. Thus, panel clutter and instrument weight can be substantially reduced.

b. To display time-shared graphical displays of rendezvous, entry, navigation, etc., to aid the crew in visualizing the problem in process. Diagrams for system maintenance may also be integrated into this type of display. Displays of target vehicles, maneuvers, scientific data, etc. may be more easily utilized by the crew than purely digital data.

c. To act as an integral part of the onboard training package necessary to re-train the crew prior to a mission-critical phase (midcourse or entry maneuver) during very long space flights. It is expected that airborne programs would be modified or changed during long flights by ground uplink and this, too, would require inflight training.

To summarize, present technology permits heavy burdening of flight crews by forcing performance of extensive redundant system integration to accomplish mission objectives. New technology offers a chance to reduce the cockpit load and increase the number, complexity, and reliability of mission functions.

By placing the crew in a more administrative or supervisory role and limiting their use to unique, carefully-prescribed decision processes, the burden on the crew can be reduced significantly. At the very least, the crew can be released to perform other functions (scientific experiments, onboard data analysis) divorced from aircraft or spacecraft control.

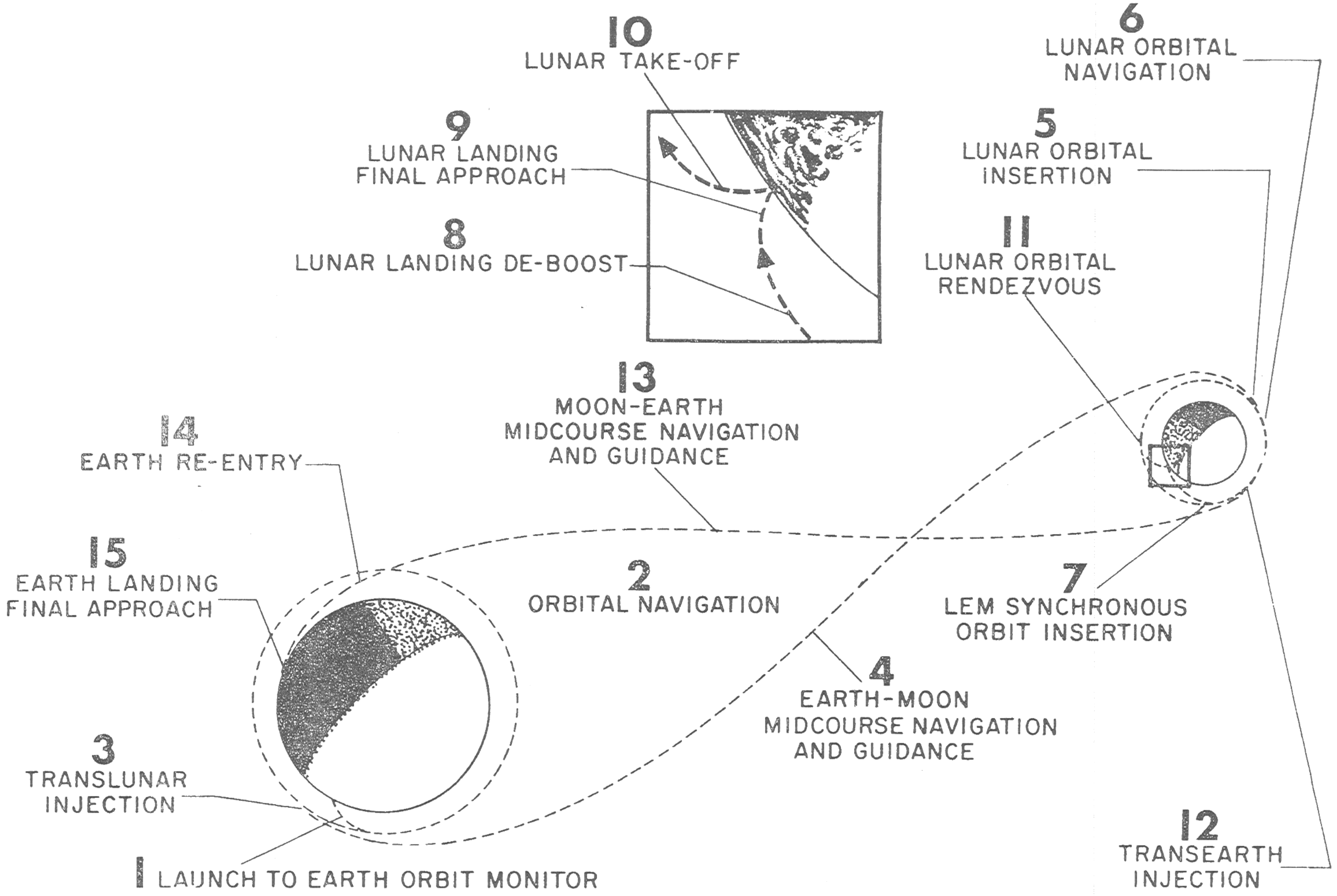


Fig. 1 Mission Phase Summary.

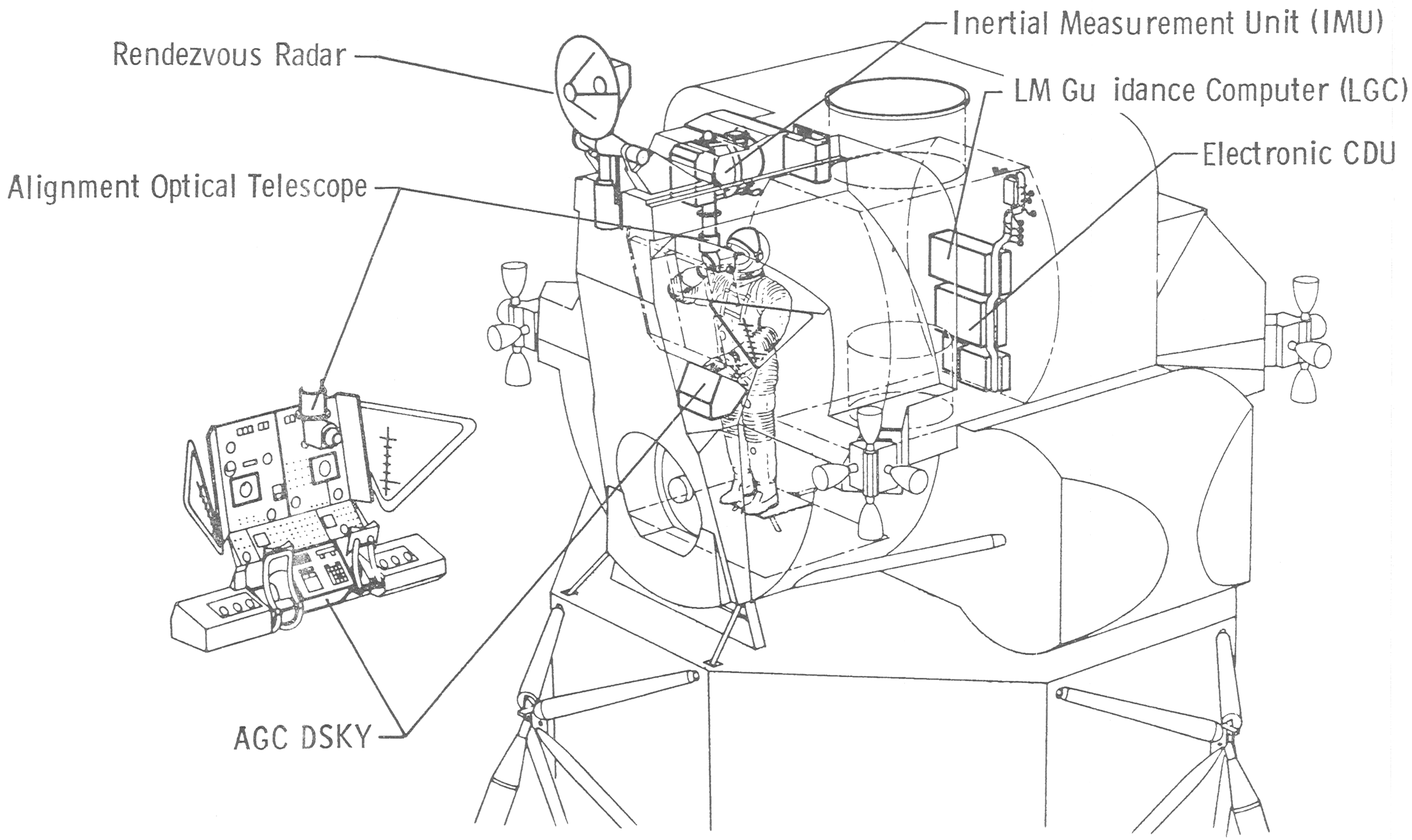


Fig. 2 LM PGNCS Installation.

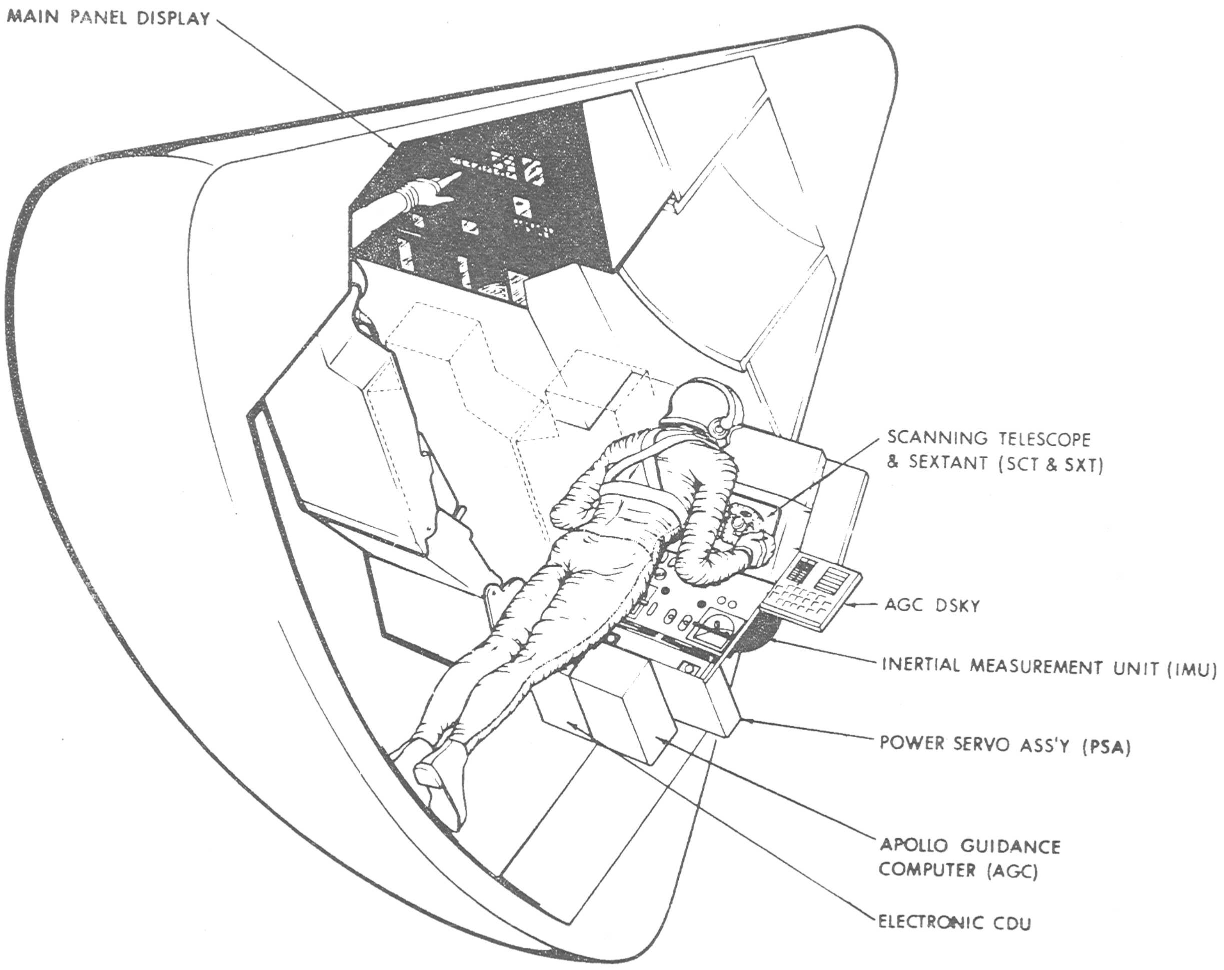


Fig. 3 AGE Spacecraft Location.

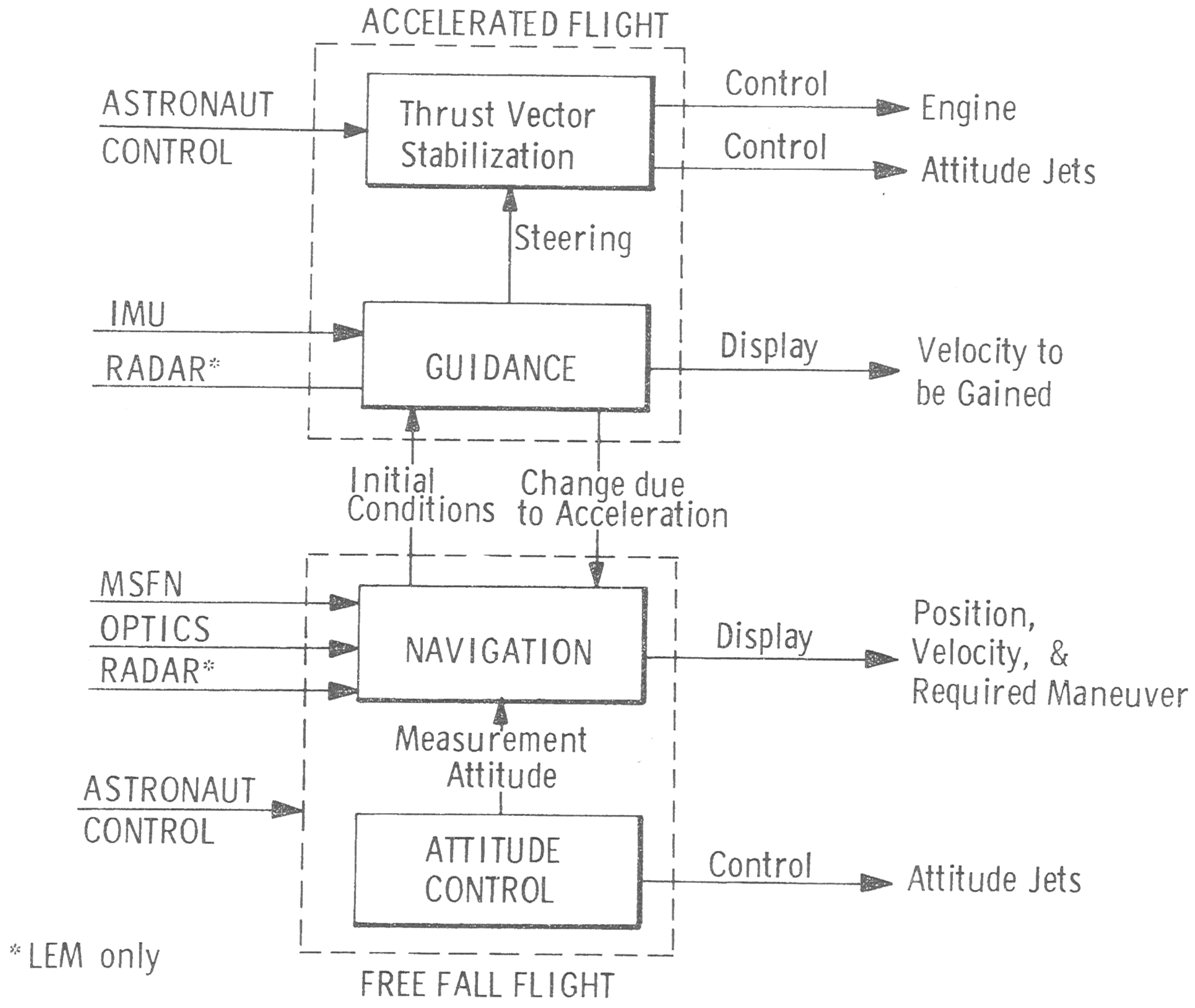
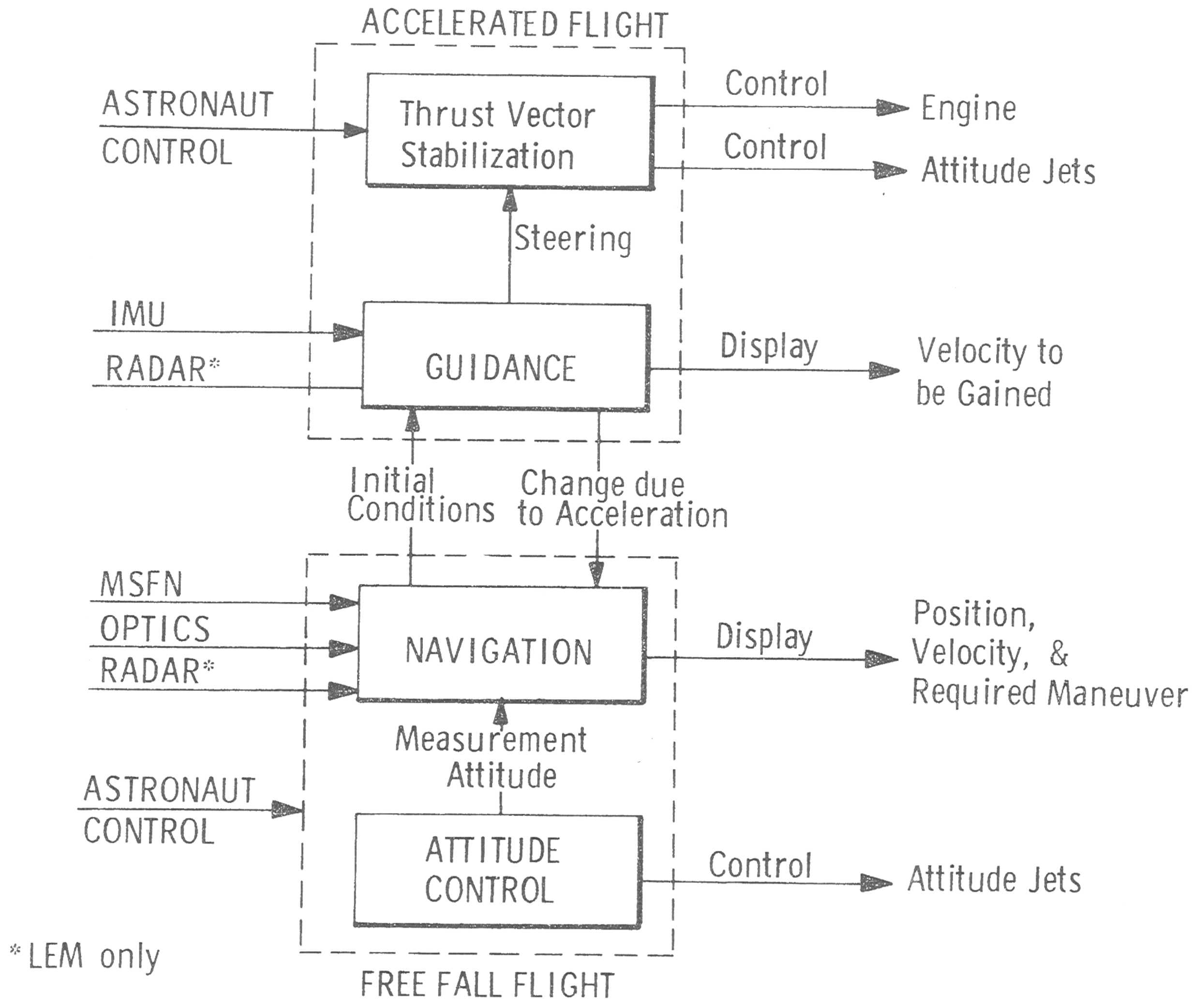


Fig. 4 Apollo Guidance and Navigation Function Flow.



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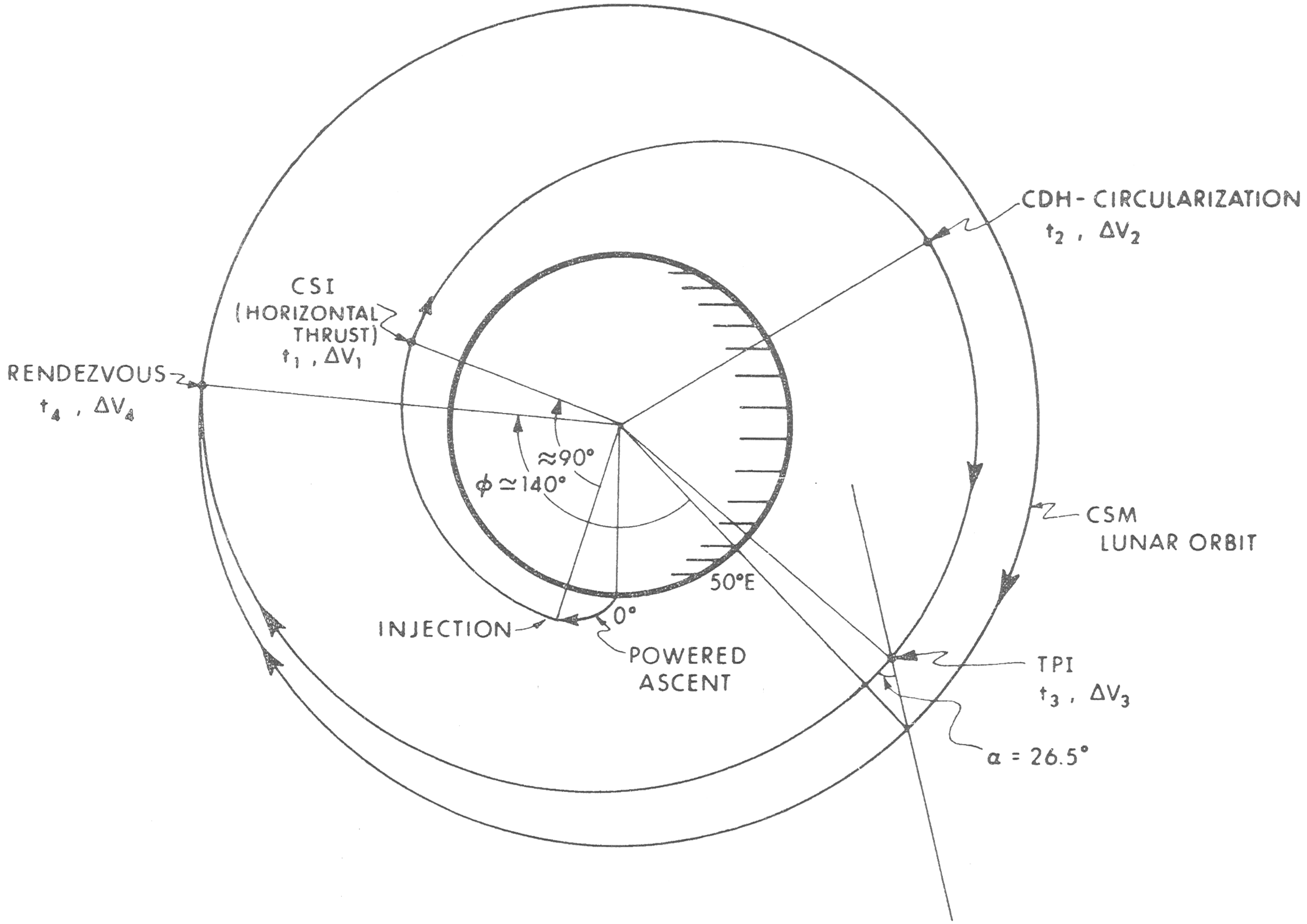
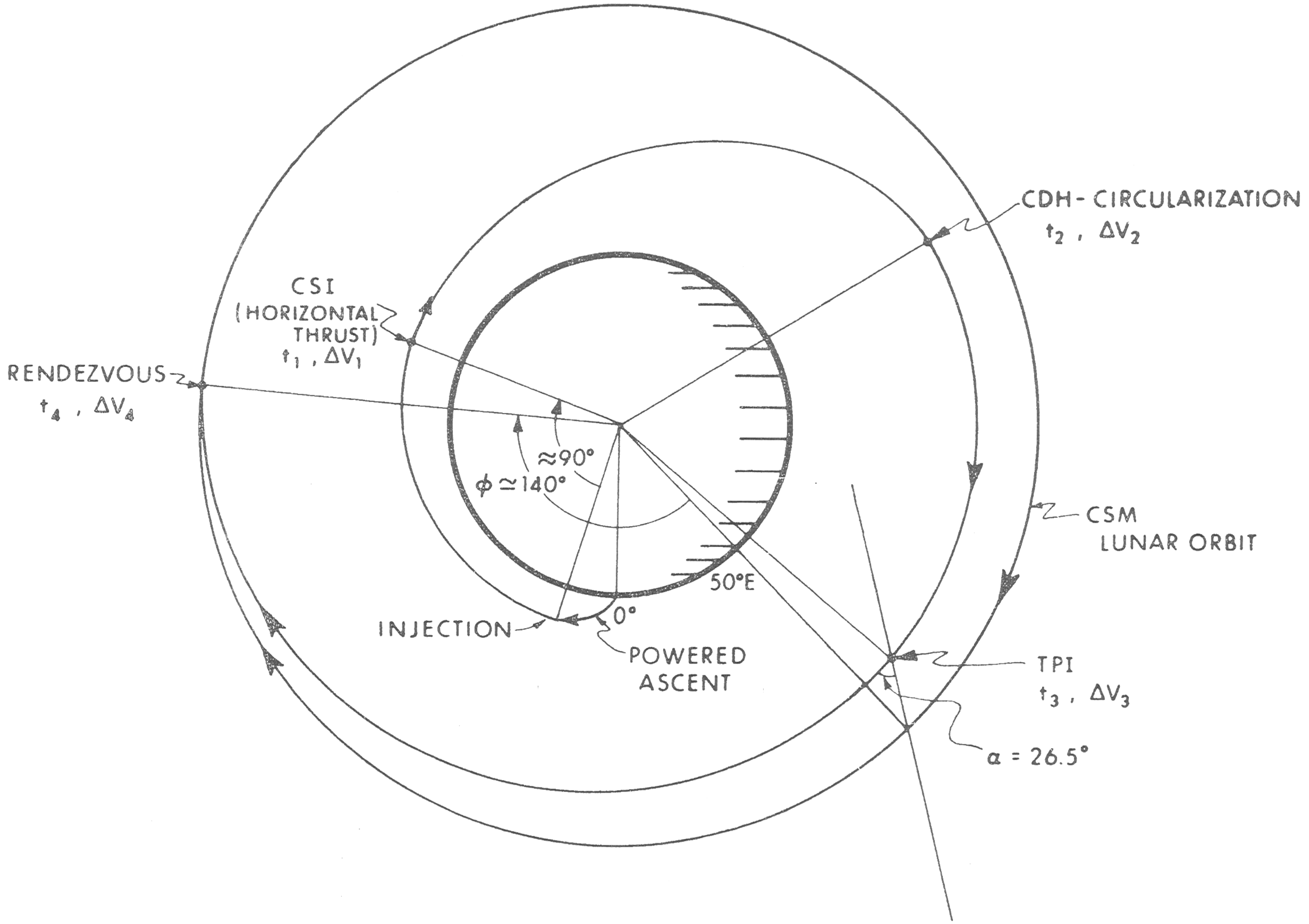


Fig. 6 Concentric Flight Plan Rendezvous Profile



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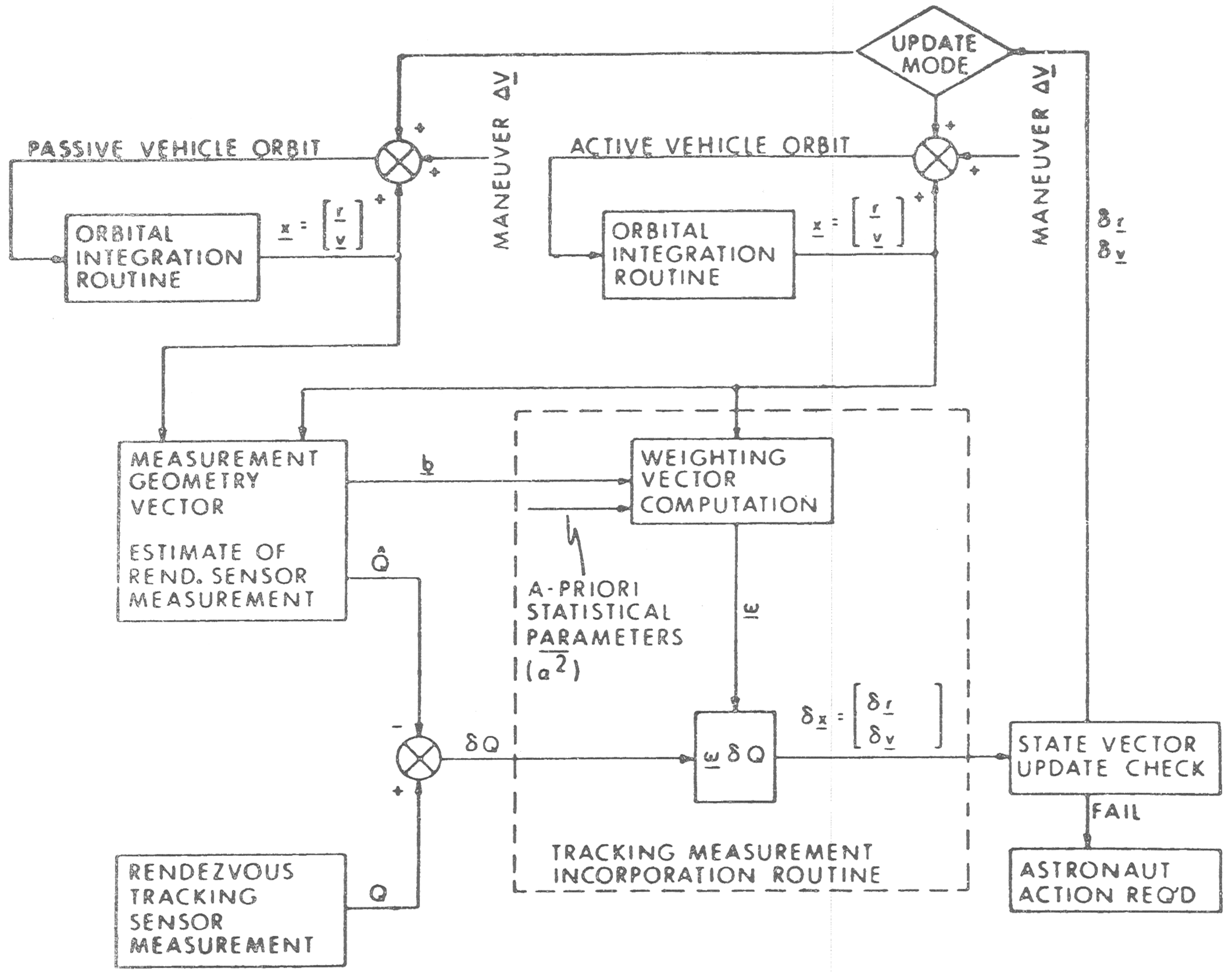


Fig. 7 Rendezvous Navigation Computation

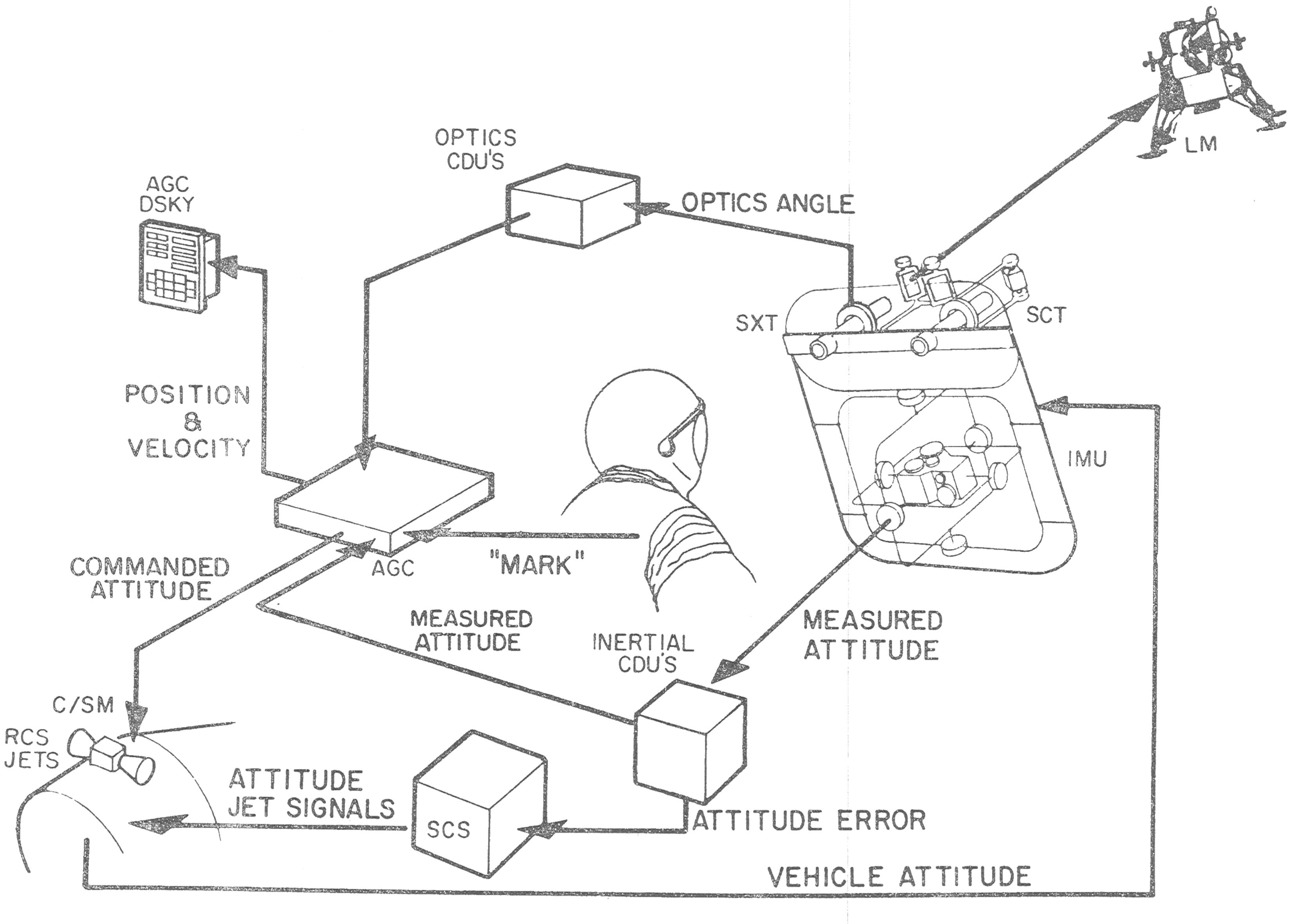
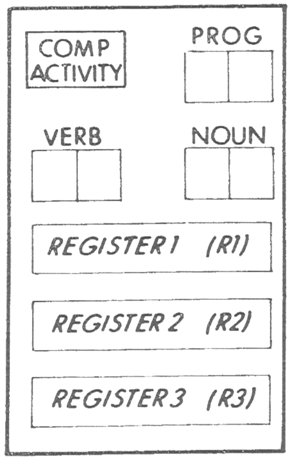


Fig. 8 PGNCS Rendezvous Navigation.

CM	LM	No.
Prelaunch and Service		(P0-)
Boost	Ascent	(P1-)
Navigation		(P2-)
Pre-Thrusting		(P3-)
Thrusting		(P4-)
IMU Alignment		(P5-)
Entry	Descent	(P6-)
Aborts and Backups		(P7-)



CMC DSKY

VERBS	
16	Monitor decimal
21	Load R1
22	Load R2
23	Load R3
25	Load R_1, R_2, R_3
32	Recycle
33	Proceed
34	Terminate Function
37	Change Program
48	Load DAP data
50	Perform option
51	Mark Target
57	Start Rend. Marks
71	CMC Update
86	Initialize W Matrix

NOUNS	
20	ICDU 's R, P, Y
22	New ICDU 's
25	Checklist (used with V50)
32	Time to Perigee hrs, min, sec.
33	Time of Ignition
35	Time to event
39	ΔT Transfer
43	Lat, Long, Alt.
46	Autopilot config.
47	S/C Inertias & Weight
48	Engine Trim
49	State Vec Update
54	R, $\dot{R}$, Tgt Elev.

Fig. 9 CMC Program and Code Summary.

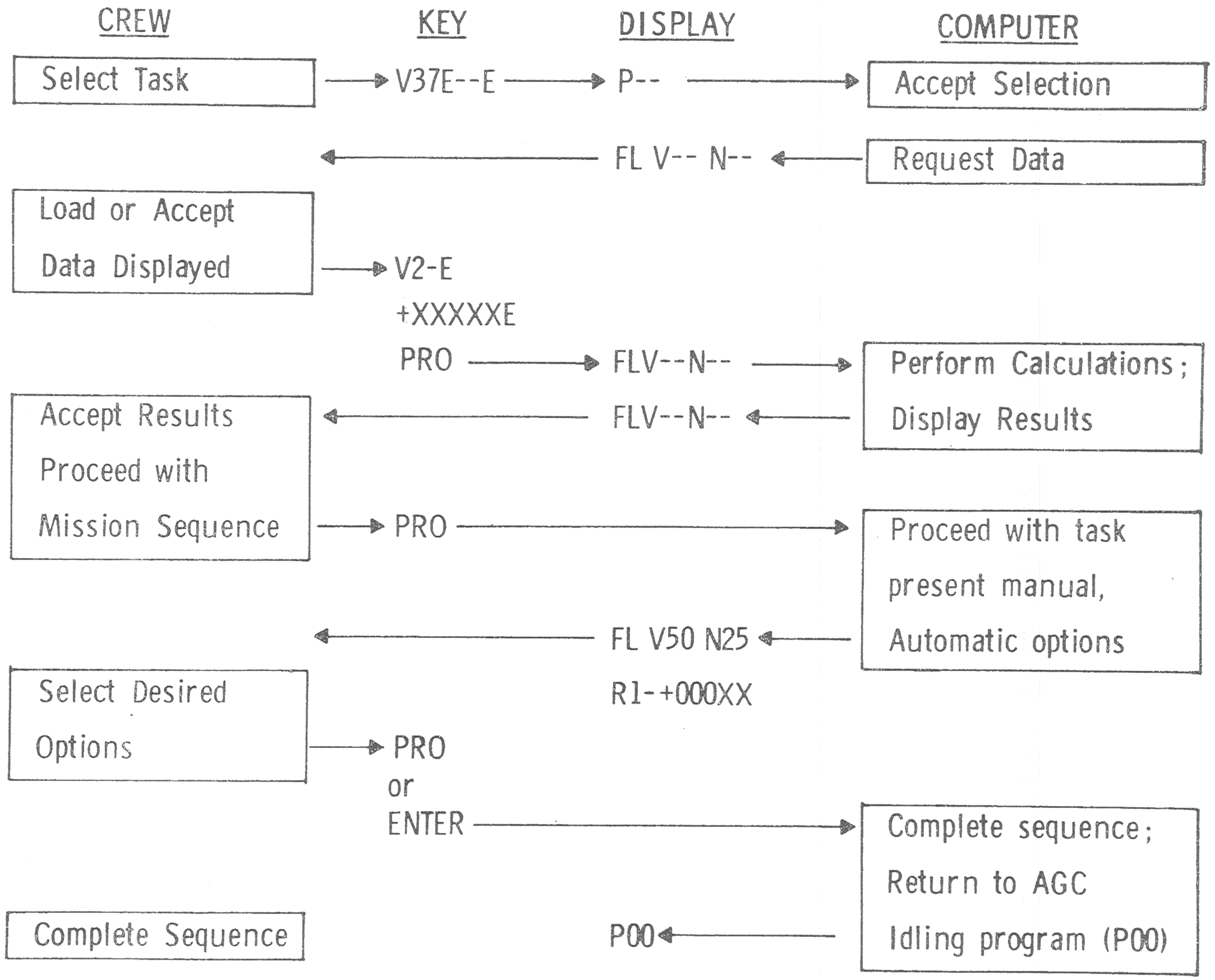


Fig. 10 Crew/Computer Interface.

FUNCTION	SYSTEM		DISPLAYS		
	PRIME	BACKUP	DSKY	FDAI	OTHER
NAVIGATION	OPTICS: SCT/SXT		✓		Reticle, Counters
		COAS		✓	Reticle
ATTITUDE REF.	IMU		✓	✓	
		BMAG & GDC		✓	
ATTITUDE CONTROL	CMC(DAP) to RCS (Auto or RHC)		✓	✓	
		1) RHC thru SCS		✓	
		2) RHC direct to RCS		✓	
GUIDANCE	CMC(TVC DAP)		✓	✓	
		1) SCS Auto		✓	
		2) MTVC (RHC)		✓	
ΔV MONITOR	CMC		✓		
		EMS			ΔV Monitor Counter

Fig. 12A Crew Functions - Summary.

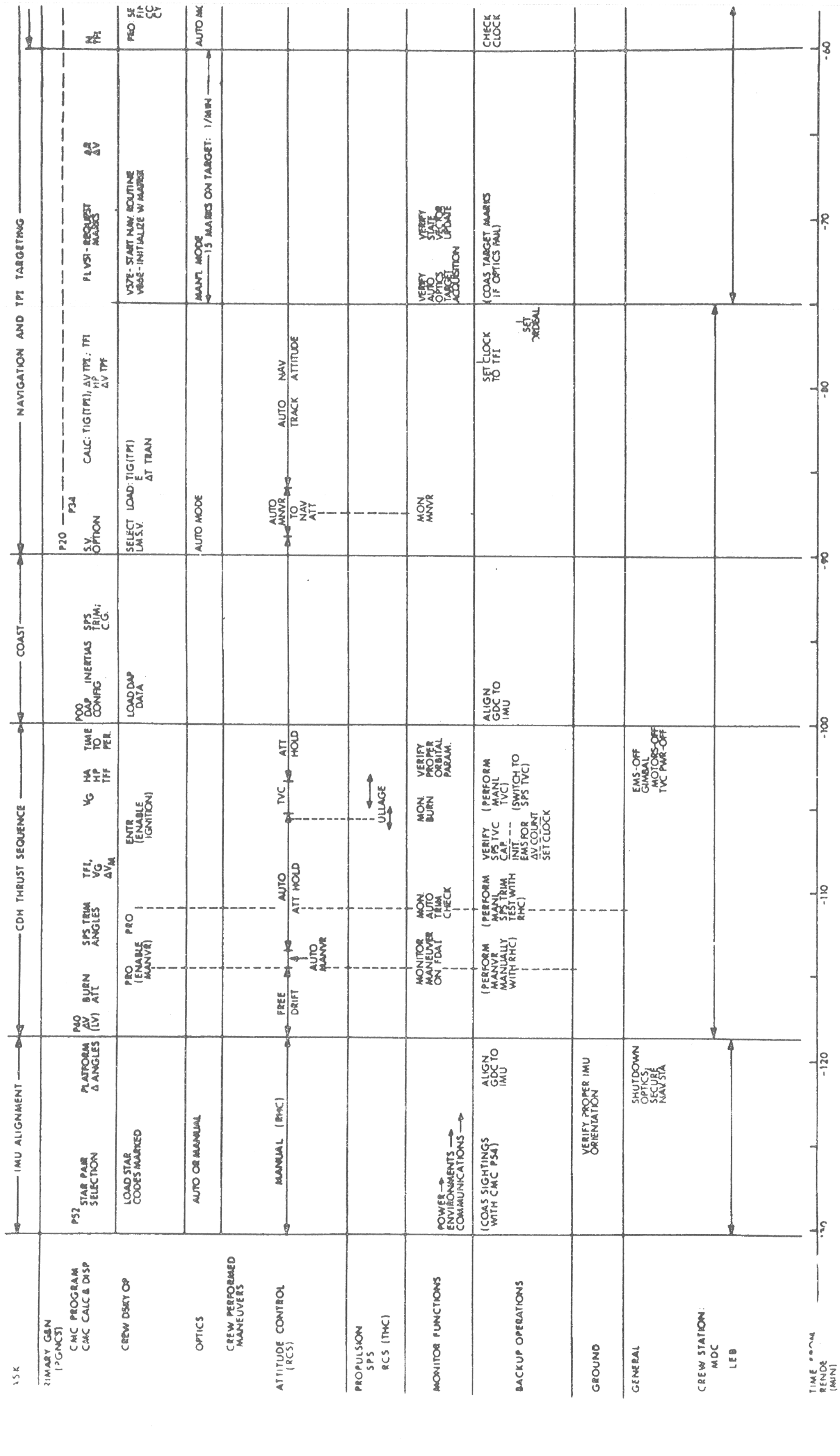


Fig. 12B Rendevous Functional Time Line.

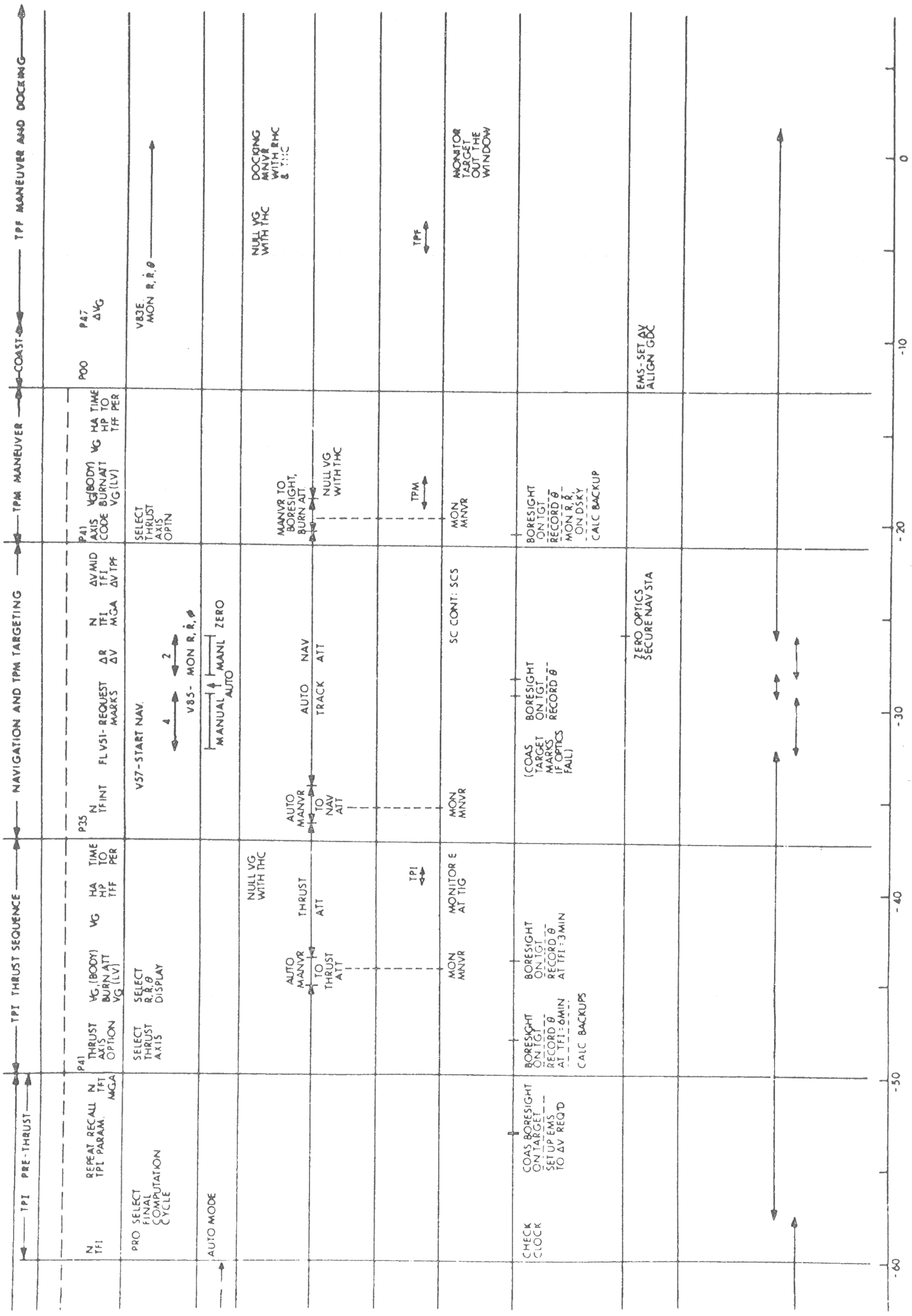


Fig. 12C Rendevous Functional Time Line.

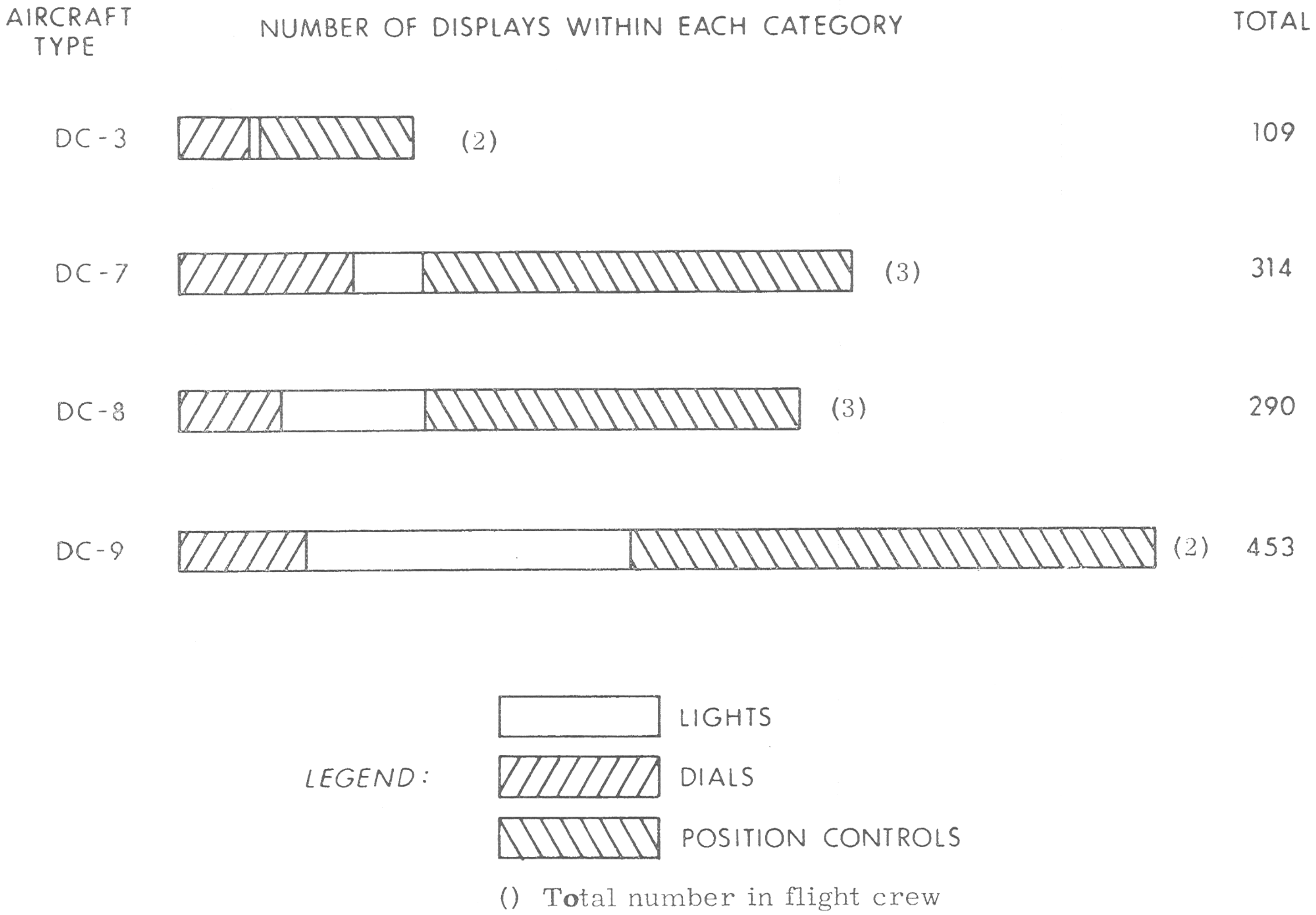


Fig. 13 Evolution of the Displays Used by the Pilot and Copilot in Douglas' DC Series.

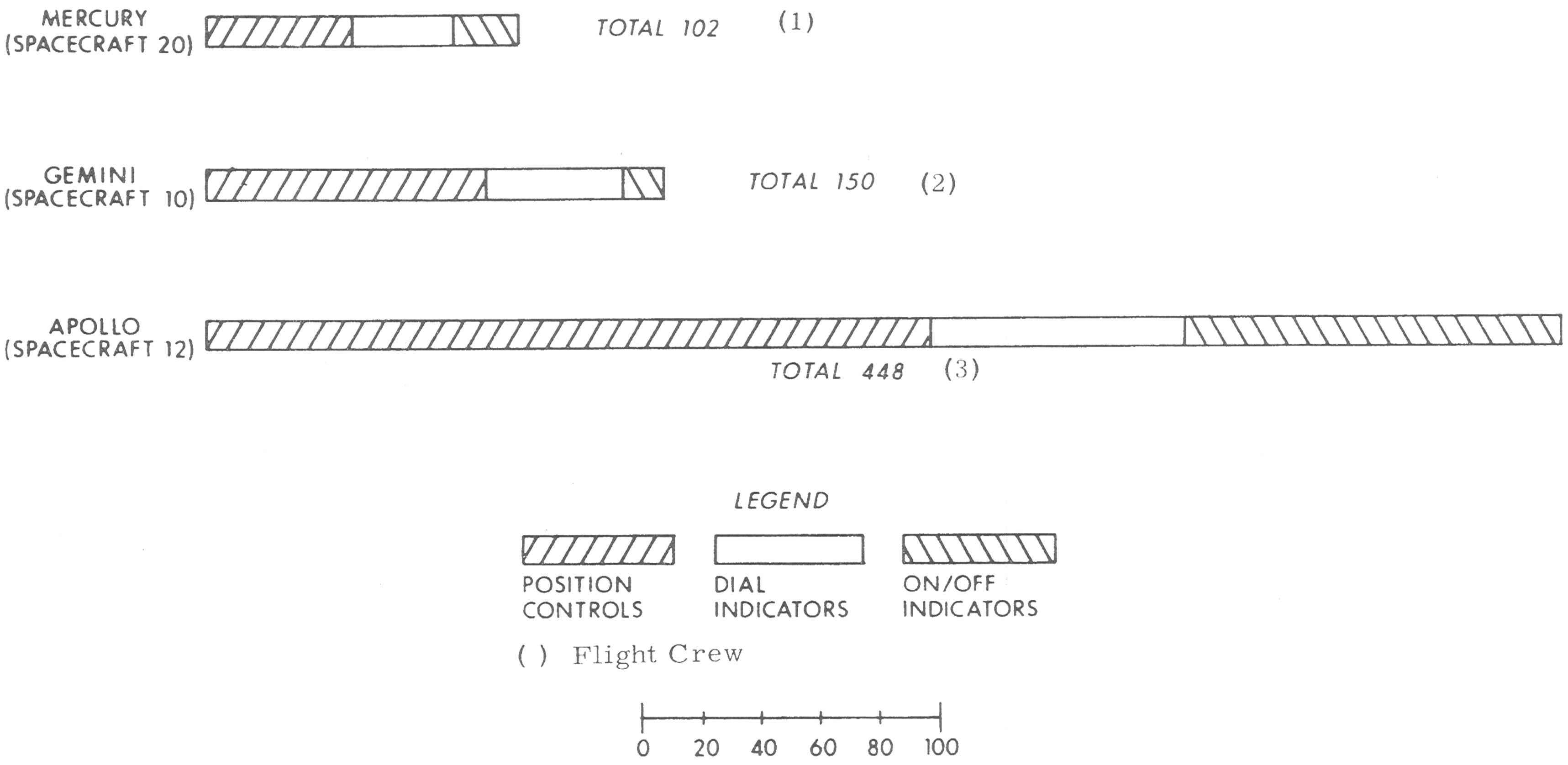


Fig. 14 Approximate Breakdown of the Various Types of Displays in Existing Spacecraft.

CSM - Rendezvous Navigation

Program 20

Step	Function	DSKY Display	Definition	Crew Action	Definition	Remarks
1.	Prog. Selection			V37E20E		
2.	Mode Selection	*PF V50 N25 R1-00203	Request PGNCS auto attitude control	Switches: SC Cont - CMC CMC Mode - Auto Key ENTR	Select PGNCS Auto	
				or		
				Key PROCEED	Reject Auto	Maneuver will be performed manually
3.	Option Code	**F V04 N06 R1-00001 R2-0000X	State Vector Update	V22E		
				1E; PRO	Update LM	
				or	or	
				2E; PRO	Update CSM	
						Maneuver and Acquire target in optics
4.	Prog. Termination			V56E	Terminate P20 only	
				or		
				V37E00E	Terminate all current programs	

* Possible flash
** Flash

V: verb
E: enter

CSM - TPI Targeting

Program 34

Step	Function	DSKY Display	Definition	Crew Action	Definition	Remarks
1.	Prog. Selection			V37E34E		
2.	Option Selection	F V04 N06 R1-00003 R2-0000X	TPI Data Load	V22E 1E; PRO or 2E; PRO	Load TIG, Compute E Load E, Compute TIG	
3.	Data Load	F V06 N37 or V06 N55	TIG (TPI) E(Elev Ang)	PRO or V2XE	Accept Displayed Values Load New Data	
4.	Poss. Alarm	PF V05 N09 R1-Code	No TIG for E	V32E or V37E34E		Return to E Load, Step 3 or Reselect P34
5.	Data Load	F V06 N39	ΔT TRANS	PRO or V25E		
6.	Poss. Alarm	PF V05 N09 R1-Code	Hyperbolic Vel would result if ΔTTrans were used	V32E		Return to Step 2
7.	Computed Data	F V06 N55 or F V06 N37	E or TIG	PRO	Accept Data	
8.	Computed Data	F V06 N58	ΔV (TPI) Per Alt (TPI) ΔV(TPF)	PRO	Accept Data	
9.	Computed Data	F V06 N35	Time from Ignition	PRO	Accept Data	Set clocks; if not first time through computation cycle, go to Step 11

CSM - TPI Targeting

Program 34 (Con't)

Step	Function	DSKY Display	Definition	Crew Action	Definition	Remarks
10.	Computed Data	F V06 N45	No. Marks Time from Ignition	PRO	Go to Step 7 if first time through comp. cycle	
				or V32E	Go to Step 7	
11.	Computed Data	F V06 N45	No. Marks TFI Middle Gimbal Angle	PRO	EXIT P34	

CSM - TPM Targeting

Program 35

Step	Function	DSKY Display	Definition	Crew Action	Definition	Remarks
1.	Prog. Selection			V37E35E		
2.	Computed Data	F V06 N35	Time from Intercept	PRO	Accept Data	Set Clocks
3.	Computed Data	F V06 N45	No. Marks TF INT	PRO	Accept Data	
				or V57E	Initiate Navigation Mark Routine	P20 must be Running
4.	Poss. Alarm	F V05 N09 R1-Code	Hyperbolic Vel would result if burn scheduled at specified TIG	V32E	Return to Step 2	
5.	Computed Data	F V06 N45	No. Marks TFI (MID) MGA	P 20	Accept Data	
6.	Computed Data	F V16 N59	ΔV (MID) TFI ΔV (TPF)	PRO	Exit P35	
				or V32 E	Return to Step 2	

CSM - SPS Thrust

Program P40

Step	Function	DSKY Display	Definition	Crew Action	Definition	Remarks
1.	Prog. Selection			V37E40E		
2.	Computed Data	F V06 N86	V_G (LV)	PRO	Accept Data	
3.	Computed Data	F V06 N22	Attitude at Ignition	PRO	Accept Data	
				or V37E52E	Perform IMU Realignment	
4.	Mode Selection	PF V50 N25 R1-00203	Request PGNCS auto attitude control	Switches: SC Cont - CMC CMC Mode - Auto Key ENTR or Key PROCEED	Select PGNCS Auto	
					Reject Auto	Maneuver will be performed by SCS or manually
5.	Test Enable	F V50 N25 R1-00204	Request SPS Engine Gimbal Drive Test	PRO	Bypass Test	
				ENTR	CMC Performs Auto SPS drive test	
6.	Poss. Alarm	F V05 N09	TIG less than 45 secs from now	PRO	Establish new TIG = 45 sec from now	
				V34E	Terminate P40	
7.	Computed Data	F V06 N40 R1-TFI R2-V_G R3-ΔV_m	Time from Ign, Vel-to-be-gained; ΔV measured	Monitor Display Begin Ullage at TFI = 15 sec.		ΔV_m should =0 until translation begins

CSM SPS Thrust

Program <u>P40</u> (Con't)

Step	Function	DSKY Display	Definition	Crew Action	Definition	Remarks
8.	Engine On Enable	F V50 N99 R1-TFI R2-V_G R3 -ΔV_m	Request Engine On Enable at TFI = 5 sec	Key ENTR	Enable Engine on at TIG	If ENTR keyed after TIG, engine ignition is immediate
9.	Thrust Data	F V06 N40 R1-TGO R2-V_G R3-ΔV_m	Time from Engine Cutoff	Monitor Burn and proper engine cutoff PRO when engine cuts off		Early cutoff may require use of RCS to achieve full V_G.
10.	Data for Trim	F V16 N85	V_G in CSM axes	Null remaining V_G with RCS PRO		
11.	Computed Data	F V06 N44 R1-Apo Alt R2-Per Alt R3- T of free fall	Orbital Parameters	PRO	Accept Data	TFF = 59:59 for stable orbit
12.	Computed Data	F V06 N32	Time to Next Perigee	PRO	Accept Data	EXIT P40

CSM - RCS Thrust

Program 41

Step	Functions	DSKY Display	Definition	Crew Action	Definition	Remarks
1.	Program Selection			V37E41E		
2.	Option Code	F V04 N06 R1-00004 R2-0000X	Assumed Translation Axis	V22E 1E PRO	Select +X axis xlation	
3.	Computed Data	F V06 N85	V_G (CSM Axes)	PRO	Accept Data	
4.	Computed Data	F V06 N22	CSM Att. at ignition	PRO	Accept Data	
5.	Computed Data	F V06 N86	V_G (local vertical)	PRO	Accept Data	
6.	Mode Selection	PF V50 N25 R1-00203	Request PGNCS auto attitude control	Switches: SC Cont-CMC CMC Mode -Auto Key ENTR or Key PROCEED	Select PGNCS Auto	
					Reject Auto	Maneuver will be performed by SCS or manually
7.	Computed Data	F V06 N22	CSM Att. at ignition	Monitor Maneuver		
8.	Computed Data	F V50 N19	Request Att. Trim	PRO	Reject Trim	Present Att. Satisfactory Return to Step 7
				or ENTR	Trim Attitude	

CSM - RCS Thrust

Program 41 (Con't)

Step	Function	DSKY Display	Definition	Crew Action	Definition	Remarks
9.	Computed Data	F V06 N85	V_G (CSM Axes)	Null V_G with THC PRO		Display first Appears 15 sec. before TIG
10.	Computed Data	F V06 N44 R1-Apo Alt R2-Per Alt R3-T of free fall	Orbital Parameters	PRO	Accept Data	TFF =59:59 for stable orbit
11.	Computed Data	F V06 N32	Time to Next Perigee	PRO	Accept Data	EXIT P41

CSM Thrust Monitor

Program 47

Step	Function	DSKY Display	Definition	Crew Action	Definition	Remarks
1.	Prog. Selection			V37E47E		
2.	Data Display	F V16 N83	ΔV (CSM Axes)	Perform Thrust as desired; Monitor ΔV increments on DSKY		
				PRO	Terminate P47	
				V32E	Zero N83 to monitor another translation maneuver	

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