MANNED LUNAR LANDING VIA RENDEZVOUS

Fred E. Digesu
Chief, Advanced Studies Branch
Astrionics Division
George C. Marshall Space Flight Center

In any mission description, the vehicles, the flight profiles, and the astrionics hardware to implement the mission are all tightly interwoven things. A final result evolves only after many iterations to the solution are made. This paper will describe one of these iterations in the Saturn C-5 Earth Orbit Rendezvous approach to the Manned Lunar Landing Program. Since the iteration to be described in an n^{th} one, there exists some basis for the hope that the perturbation from the final solution is small.

This paper is not concerned with the landing itself, but only with those operations leading to injection of the space craft into the lunar transfer trajectory. However, as is to be expected, it is the target conditions which set the pace for the overall operation. The entire operation must be sized to culminate at a time and place which places the lunar target in an attainable position. The procedure would call for a burst of activity lasting over a relatively short time as compared to the long and extensive preparations leading up to it.

The activity must be aimed at the opening of the lunar "launch window". Figure 1 illustrates the variation in the velocity increment required to launch a vehicle into a lunar transfer trajectory from a 485 kilometer earth orbit. The minima are at irregularly spaced intervals and are a function of the inclination of the lunar and the earth satellite planes and of the position of the moon in its orbit around the earth (i.e., the day of the month). In an operations analysis these spacings will influence the number of vehicles on the launch pad (primary and back-up), their state of readiness, the firing rate, and also the flight profile to be chosen. Whether it is decided to go by "connecting" or by "tanking" mode, the objective must be to get the spacecraft in the launch ready state at the opening of one of these launch windows. It may be desired that the first vehicle be capable of remaining in a functionally capable state even after bridging one or more of the gaps between the windows. This consideration will influence the design of the vehicles as well as the operational modes to be designed into the flight control hardware. For example, a sleep switch may be desirable from the standpoint of savings in battery weight.

The Flight Profile

With the end objective in mind, let us start to define the flight profile.

First it can be established that the "parking orbit" technique is the desired approach. This is dictated by the possible lift-off delay of the chaser vehicle. If launch from other than the equator is considered, there is a two fold constraint upon the lift-off time. These are:

1. The chaser and target orbits are coplanar ("dog-leg" requirement).

2. That chaser and target meet at the same point in space and time.

(Phase Requirement)

It is possible to meet both of these requirements (even when launch is not in the equatorial plane) by so called "Rendezvous Compatible Orbits". These are orbits whose periods are an integral division of the earth sideral period. Thus, they can be designed such that, for instance, once a day the satellite target will appear over a given earth subpoint. Since, however, this condition holds for only an instant of time, a "direct ascent" rendezvous would presume zero launch delay time. The velocity penalty for lift-off delay can be placed into two categories.

1. That required to "dog-leg back into the target plane".

2. That required to "catch up" with the target.

The "dog-leg" requirement versus time (plotted in terms of velocity increment, ΔV) for a chaser launched at 90 degrees azimuth from Cape Canaveral into a 225 kilometer orbit inclined at 28.3 degrees to the equator is shown in Figure 2. If a variable azimuth launch for the second vehicle is presumed, the curve takes the shape indicated in Figure 3.

In the second category, the "catch up" maneuver manifests itself in essentially an off optimum ascent trajectory for the second vehicle. The ΔV penalty versus time for launch delay of a vehicle to meet the target depends on the ability to optimally guide to the desired end conditions. For one minute launch delay the cost can be of the order of 80 m/sec to meet a target in a 250 n. mi. orbit.

It can be seen that the second constraint poses a stiff requirement upon lift-off delay, while the first is a much more lenient restriction.

Since lift off delays are inevitable (the present best estimate for Saturn is that launch delays up to fifteen minutes are possible, even in the operational version, not considering the manned aspect) it is clear that launch into the plane of the first vehicle be the designing criterion and that other less expensive means be found for the phasing problem.

It turns out that a parking orbit for the chaser meets the requirements. If the orbits of the two vehicles are of different altitudes, the period difference automatically causes a catch up in phase. When the proper constellation is reached (Figure 4) the chaser can be injected into a Hohmann transfer ellipse to rendezvous with the target. Of course launch delays can occur even here, but these can be shown to be much less severe. Consider Figure 5. The ΔV required for transfer of the chaser from the lower to the upper orbit can be shown as a function of the central angle traversed (from perigee) by the chaser. This can be translated into "lead angle" deviation from nominal by the target, which in turn (due to the difference in angular velocity of the two vehicles) can be plotted as ΔV penalty versus chaser launch delay time, as is shown in Figure 5.

A parking orbit for the chaser can therefore be established as a requirement.

The next consideration concerns the sequence of launchings, manned first or second? This has ramifications in tracking requirements as well as abort precautions for the manned vehicle, since a variable azimuth launch will be mandatory for the second vehicle.

There are, however, two primary factors to be considered.

1. That man should not be committed until as late as possible in the operation, and

2. The first vehicle may have a prolonged orbit stay time. This means that (a) it must be penalized with heavy insulation and micrometeorite protection, and (b) the possibility must exist for turning off all equipment so that weight savings in the primary power source can be attained. Thus, the unmanned vehicle should go first.

The question as to which vehicle (the manned or the unmanned) should do the orbital maneuvering would tend to indicate a profit in propellant if the lighter of the two were the chaser. Since the maneuvering will probably be done with auxiliary propulsion systems with lower specific impulse than the primary propulsion, and since the chaser will be required to take the penalty of the reserve ΔV's in the velocity budget, it would seem expedient not to thus burden the manned vehicle.

The desirability of the tanker being the chaser in the Tanking Mode can be established; while evidence seems to favor the R-1 vehicle being the chaser in the Connecting Mode, it is not quite so conclusive as in the Tanker case.

The point of view taken here is that the unmanned vehicle does the maneuvering and that this maneuvering in the terminal (that is, the actual rendezvous phase) is done under close radio and radar supervision of the manned vehicle. It is quite likely that the actual docking (that is, from the last few feet up to mating of the vehicles) will be done by "man across the loop".

We are now in a position to establish the operational flight profile.

The unmanned vehicle is launched into the lower parking orbit. The lead time over the opening of the Lunar Launch window will be determined by the nominal time required to ready and launch the second (manned) vehicle. The stay time in the parking orbit can vary from as little as an hour or two, to as much as ten days. This is, of course, a direct function of the firing rate at the cape, lift-off delay of the second vehicle (and thus the phasing which must be done) and the opening of the first and subsequent (if necessary) lunar launch window.

Provision will be made for a chaser "midcourse" maneuver. This midcourse correction is defined as that impulse which either (1) injects the chaser into the perigee of a Hohmann transfer up to target acquisition point, or (2) corrects the existing chasing ellipse so that chaser will come into nominal target acquisition conditions. Thus, it can be seen that the question of a circular versus elliptical parking orbit looses significance as far as guidance is concerned.

The manned vehicle will be placed into the higher orbit by a launch phase which injects it into the perigee of a transfer ellipse with a subsequent circularizing apogee kick. Ground tracking will establish the two ephemerides and ground computation will determine the time, magnitude, and the direction of the chaser midcourse impulse.

The chaser ascent ellipse is so designed that at nominal acquisition range of 40 kilometers the relative closing velocity is 63 m/s and the angular rate of the line of sight in inertial space is zero. See Figure 6. The guidance logic is to hold the line of sight rate zero (thus maintaining a collision course) while breaking the closing velocity to essentially zero at some small distance ahead of the target. It is to be noted that this breaking thrust is in actuality boosting the chaser into the higher energy orbit and for the normal case is essentially nothing but a well timed apogee kick.

Docking can be accomplished by allowing a small closing velocity to remain after the rendezvous thrust. Alternately, the chaser can be brought to a virtual stop and the vehicles guided together under the close supervision of the man across the loop. In either case the touching velocities and the docking dynamics will be very much determined by the close in accuracy of the sensors and it is quite likely that some form of optical sensor, perhaps TV, will have to be used for this phase.

After docking, and loxing in the case of the tanking mode, the space craft must be checked out and counted down for orbital launch. Although a given lunar launch window may be open for several hours, the actual orbital launch point will be optimum at only one place in the orbit. Specifically this means that inside of the larger launch window are smaller windows (in the order of several minutes) which occur once each orbit. The actual launch must take place inside these smaller launch windows, and since these windows appear at about one and a half hours intervals, provision for extended launch ready standby as well as updating of the guidance system must be made. This will be required if holds are encountered which cause misses of these windows.

Velocity Budgets and Error Analysis

Earlier in this paper, mention was made of the propellant contingency which must be designed into the chaser. This section will point out those places where tolerances must be designed in and will indicate the propellant contingency requirement in the form of a velocity budget. A detailed description of the analyses made is beyond the intent of this paper. However, it is felt that a description of the overall approach which was taken to establish the maneuver velocity budget would be informative. The impact of the analysis upon vehicle design can be felt when it is realized that the chaser burns about 100 lbs. of propellant per meter per second velocity increment.

Error sources which contribute to the velocity budget must be analyzed in detail and in themselves are an interplay between operational analyses, guidance scheme and hardware inaccuracy, navigational uncertainties, and the influence of physical phenomena upon the flight profile.

Under the heading of operational analysis we have the earth and orbital launch window influence. The effect of launch delay in the required velocity has already been shown. It remains that a figure of merit for the launch operation be established. This in itself requires a detailed evaluation of checkout and countdown procedures. The degree to which automation is involved in the countdown must be evaluated, among other things against the number of operations to be done, the number of checks which are to be made, and the length of time the various subsystems can be expected to reliably remain in the launch ready state. As has been already mentioned, the tolerance upon Saturn launch is established (at least for purposes of mission analysis) as 15 minutes. The probability of liftoff is a constant over the 15 minute interval.

A mitigating circumstance is that the Saturn ascending burn is long enough (that is, the central angle traversed is large enough) so that the maneuver of doglegging on the way up begins to pay off. In other words, the turning is done at the relatively lower velocities and, as is indicated in Figure 7, an overall saving in ΔV can be made. This is being investigated within the framework of the Saturn path adaptive guidance concept and thus the notion of variable azimuth launch begins to include the idea of some variation in the azimuth during Saturn boost to injection.

Two things come immediately to mind under the heading of the influence of physical phenomena. These are (1) the effect of atmospheric drag upon the lower vehicle, and (2) the difference in the nodal regression rates of the orbital planes of the upper and the lower vehicles.

The lower orbital altitude is a compromise between vehicle performance, remaining atmospheric drag, and the desirable difference in period between the upper and the lower vehicles (i. e, the "stalking" rate). At the lower orbits considered, the air density is appreciable but its magnitude is unknown to the extent that calculations of orbit lifetime must be given a tolerance of 50%. This is an error source which must be accounted for in the velocity budget.

The regression of the nodes of satellite is, as is well known, due to anomalies in the attracting body's gravitational field. For the earth these regression rates are a function of the inclination of the satellite and earth equatorial plane and of the orbital altitude of the satellite, and can be fairly well predicted. Figure 8 shows the nodal regression rate as a function of altitude for an orbital inclination of 28.3 degrees to the equator. This shows that the difference in rates can lead to an inclination of 1.1 degrees between Target and Chaser orbital planes for a stay time of one day. This, of course, translates into a dogleg requirement for the chaser which must be budgeted.

The velocity budget for hardware and scheme errors can be determined by the following approach.

An analysis of the accuracy with which the target and chaser can be injected into their respective orbits is made after error distributions of the guidance components and of the guidance schemes have been established. This will indicate what must be added to the nominal ΔV for transfer between the nominal orbits.

Next an error analysis is made of the accuracy with which ground tracking can establish the ephemerides of the two vehicles. A correlation of the accuracy with which the chaser ardware can deliver the midcourse maneuver impulse is then made to establish the accuracy with which the chaser can be injected into its transfer ellipse. This is in turn reflected into the deviation of the chaser from its nominal at the target radar acquisition point. Error in target orbit determination can be analyzed as being deviation of the chaser state variables from nominal at the acquisition point.

Having now established an error space at radar acquisition point, the error analysis proceeds to establish the velocity budget for the terminal phase. These errors fall into the two categories of hardware and scheme. Since the rendezvous guidance is basically a homing technique, the errors are convergent and the attainable end conditions (docking velocity zero at a given relative range) are directly functions of the close up sensor accuracy. Accuracy at the longer ranges (as well as dispersions from nominal conditions at acquisition) can be visualized as causing extraneous maneuvering which results in over expenditure of velocity increments.

The accuracy required of the sensors close up can be analyzed in a docking dynamics study as shown in Figure 9.

Let it be hypothesized that the conditions necessary for docking are that the cone tip and velocity vector of the chaser enter the cone of the target. (These are, of course, not sufficient conditions since relative closing velocity, cone angle, and friction coefficients will determine the cone penetration and thus the actual docking and latching).

The hypothesized necessary conditions can be thought of as guidance criteria which can be used to define the sensor inaccuracies (as well as the minimum controllable impulse and cutoff dispersion) as illustrated in Figure 10.

Hardware Instrumentation

This section will discuss the various items of the Saturn instrumentation package and their functional use in the given flight profile.

Consider the tanking mode and the unmanned tanker first launch. The instrumentation package is contained in the instrument unit shown in Figure 11. A typical component layout is shown in block diagram form in Figure 12, where the basic Saturn inertial guidance and control system is enclosed by a dotted line.

The inertial unit is a four gimbal, full angular freedom, platform stabilized in inertial space by three air bearing gyros. Three mutually orthogonal pendulum integrating gyro accelerometers are mounted on this stabilized reference frame.

The guidance computer will be an all solid state, core memory type, general purpose computer. Triple redundancy with voting circuitry will be employed in order to ensure reliability of operation. In addition to the guidance computation, one of the major functions of the computer will be its role in the orbital checkout procedure.

The Guidance Signal Processor is the input/output box for the computer. The computer communicates with this device only. It contains resolvers to transform the Euler angles computed in the computer into direction cosines which the platform gimbal resolvers translate to the missile body axes. In addition it can be seen that the GSP accepts and distributes the intelligence commands generated on board or externally. It thus can be thought of as the central distribution point for the guidance and control signal flow.

The control computer contains the servo electronics which position the control elements (engine throw angle, attitude control jets, etc.) in pitch, yaw, and roll in accordance with the guidance commands and the error signals resolved from the platform gimbal angles. The control computer also contains the electrical shaping networks required for rigid and flexual mode stability.

Injection of the tanker into the lower parking orbit is done by the basic Saturn Guidance and Control system just described.

In order to save power in the parking orbit, all subsystems except a command receiver and a tracking beacon are de-energized and the vehicle is permitted to tumble randomly. Should the required orbit stay time turn out to last for several days, a check is maintained on the operational hardware by subjecting the vehicle to an orbital checkout at least once a day. The orbital checkout will be initiated by ground request through the digital radio command link. See Figure 13.

The inputs to the high capacity solid state multiplexer contain all the analog channels to be used in the checkout. The multiplexer samples each channel at preselected rates. An analog to digital converter in the Vehicle PCM Telemetry System digitizes each analog data sample and formating and logic circuitry interlaces other data that originated in digital form in the vehicle. Thus, each channel of data in the vehicle becomes available for analysis by the spaceborne computer, transmission over the telemetry system and/or recording in the vehicle for later playback.

The spaceborne computer is programmed with sub-routines for total vehicle checkout, stored conditions for channel analysis, and self checking sub-routines. The checkout sub-routines would perform such test as, for example, "static" tests (tank pressures, temperatures, vehicle voltages, etc.) "dynamic" tests (such as hardware response to test stimuli) and tests of the control and guidance systems (torquing of rate gyros, accelerometer readouts, etc.). The computer would analyze the results for a go-no-go decision. In case of a "no-go", some fault isolation could be done by prestored analytical subroutines. In other cases these analytical programs will have to be transmitted up from the ground and into the computer through the radio command link.

The manned vehicle is now launched into orbit. Again the primary launch guidance will be the inertial system. Injection will be into a "coast up" ellipse which has an apogee at the operational orbit attitude. The inertial system will compute and direct the velocity increment for the circularizing kick in the apogee. This will be verified for the astronauts by ground tracking and command prior to execution.

Once in orbit, the ephermis of the manned vehicle will be established by ground tracking. The time, direction, and magnitude of the velocity increment required for chaser midcourse will be computed on the ground.

A command to reactivate subsystems is transmitted to the chaser via its radio command link. The chaser goes into a search mode until its horizon sensors acquire earth. Once this is done the platform is re-erected by the gyro compassing technique and the ground computed impulse is transmitted into the chaser guidance computer through the radio command link. Thus, at the proper time the inertial equipment on the chaser supervises and measures the required impulse which will place the chaser at the acquisition point.

The target radar acquires the approaching chaser near the acquisition point. The radar measures the relative range, range rate, and bearing angle of the chaser. This information is displayed in the capsule, while at the same time it is processed in the target computer where the stored guidance logic transforms it into velocity increment commands for the chaser. These commands are sent to the chaser where they are executed in much the same manner as was the midcourse maneuver. Since the capsule control the command link, guidance override and preemption of the command function will be at the descretion of the astronaut.

The docking maneuver has so far been, of necessity, studied by analog computer and scale models. In this way the impact loading on the docking structures, mating mismatches, and vehicle dynamics can be studied in terms of the threshold inaccuracies of the docking sensors. Undoubtedly for the first times in orbit the astronauts will have available much more information than will actually be needed. Radar, optics, TV, even a window or a periscope! The instrumentation and actual techniques as well as man's effect across the loop can evolve only after actual docking experimentation in orbit.

Finally it might be mentioned that by and large the job of earth orbital rendezvous can be done with relatively conventional approaches to the guidance and control hardware. Only modest (at most) advances are called upon. More exotic implementations like cryogenic gyros and computers can be left for later developmental effort. Such exotic systems, while quite attractive for talking purposes, are not considered safe risks for hard planning purposes as yet.

It is considered that the overriding philosophy for the on board instrumentation will be a requirement for reliability and long life. The greatest advances will be required in these areas and it is expected that design and testing of components will be geared to this goal.

References

1. E. D. Geissler and W. Haeussermann, "Saturn Guidance and Control", Missiles and Rockets Magazine, dated February, 1962.

2. Lockheed/STL, "Orbital Docking Test Study", Contract Number NAS-8-864, dated July, 1961.

3. J. Jensen, G. Townsen, J. Kraft, J. Kork, "Design Guide to Orbital Flight", McGraw Hill Book Company, Inc., dated 1962.

4. W. E. Minor and D. H. Schmeider, "The Path-Adaptive Mode for Guiding Space Flight Vehicles", ARS Guidance, Control and Navigation Conference, Stanford University, August 7-9, 1961.

5. W. G. Thornton, "The Simulation of Docking Dynamics", MSFC Inter Office Memo, dated April 9, 1962.

6. J. Harden, Jr., "Minimization of the Angle Between Target and Chaser Orbital Planes with Reference to Chaser Launch Time Variations", MSFC Technical Memo, MTP-ASTR-A-62-3, dated January 30, 1962.

7. R. Weber, W. Pauson, and R. Burley, "Lunar Trajectories", NASA Technical Note D-866, dated August 1961.

8. J. Ladner and F. Speer, "Orbital Lifetimes of Tanker and R-Vehicles", MSFC Internal Memoranda data February 15, 1962, March 6, 1962, and March 26, 1962.

9. P. J. deFries, "Saturn C-5 Flight Profile for Rendezvous", Astronautics Magazine, dated April, 1962.

10. Georg von Tiesenhausen, "Engineering Problems of Orbital Operations", Astronautics Magazine, dated April 1962.

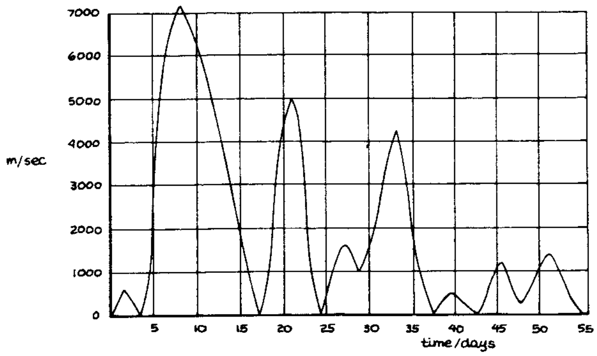
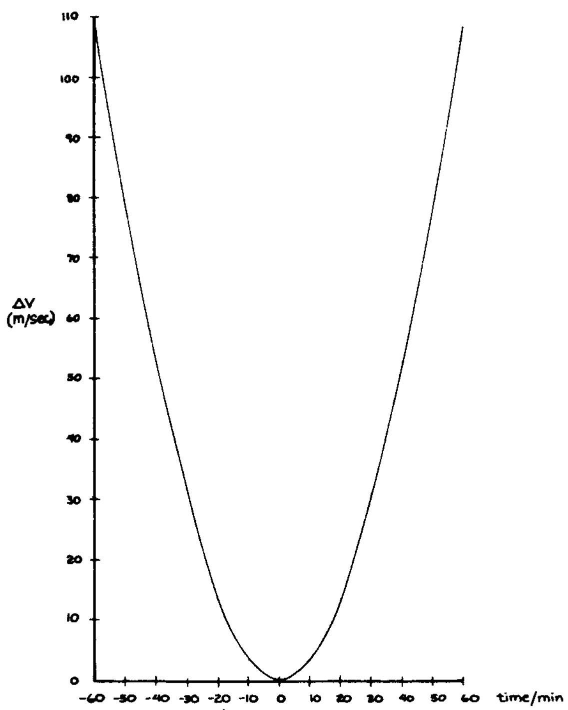


FIGURE 1

VELOCITY REQUIREMENT OVER NOMINAL FOR DEPARTURE FROM EARTH ORBIT TO MOON. LUNAR ORBIT INCLINATION 18.50°, EARTH ORBIT ALTITUDE 485 KM. EARTH ORBIT INCLINATION 30°



VELOCITY INCREMENT VS. TIME OF LAUNCH FOR ROTATION OF A 225 KM ORBIT INTO THE DESIRED ORBITAL PLANE (VARIABLE LAUNCH AZIMUTH)

FIGURE 3

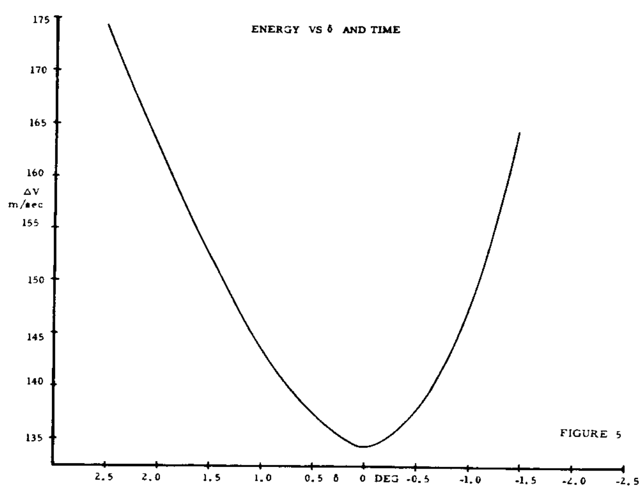


FIGURE 5

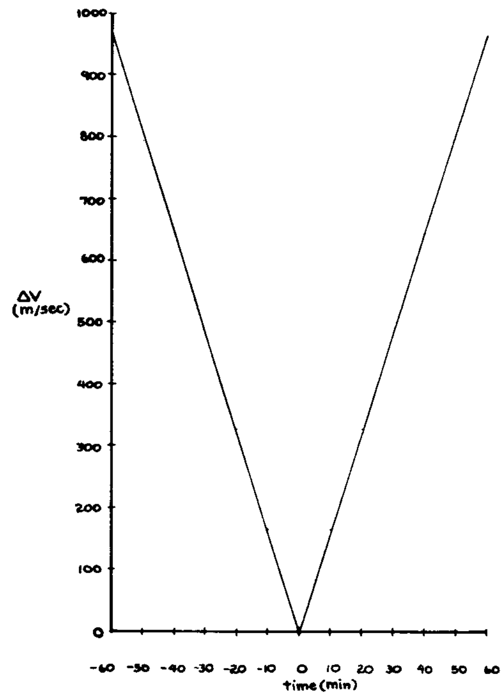


FIGURE 2

VELOCITY INCREMENT VS. TIME OF LAUNCH FOR ROTATION OF A 225 KM ORBIT INTO THE DESIRED ORBITAL PLANE (NON VARIABLE LAUNCH AZIMUTH)

ORBIT TRANSFER CONSTELLATION

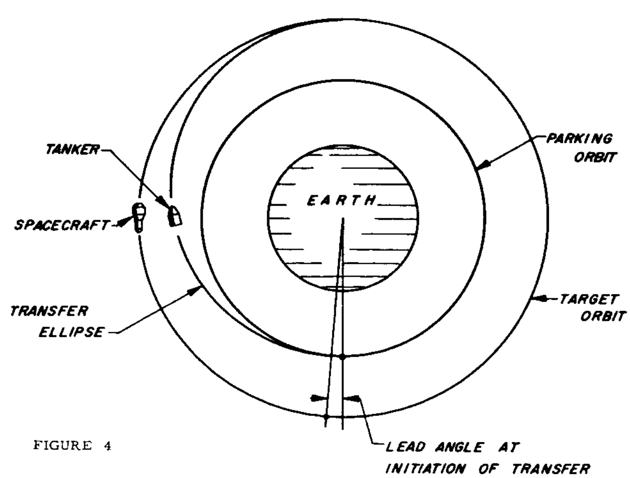
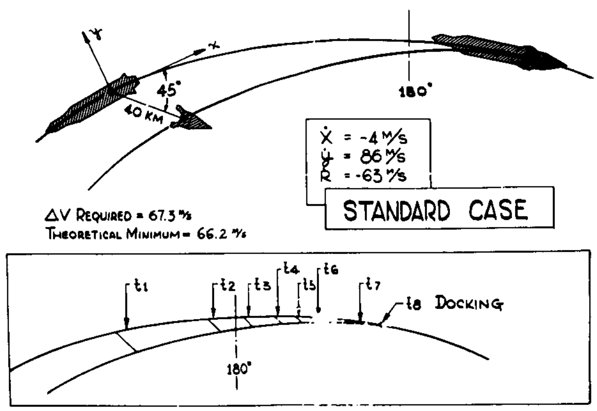


FIGURE 4

RENDEZVOUS MANEUVER

FIGURE 6



ACQUISITION CONDITION (t=0)

$-.06\,°/\text{SEC.} < \omega_p < +.06\,°/\text{SEC.}$

$-.06\,°/\text{SEC} < \omega_y < +.06\,°/\text{SEC.}$

$20\,\text{KM} < R < 60\,\text{KM}$

$-100\,\text{m/s} < \dot{R} < -45\,\text{m/s}$

t	SEC	EVENT	$\dot{R}$ (m/s)
t_0	C	ACQUISITION	-63
t_1	600	BRAKING PHASE BEGINS	-67
t_2	626	FIRST ω_p CORRECTION BEGINS	-34
t_3	637	FIRST ω_p CORRECTION ENDS	-20
t_4	650	BRAKING PHASE ENDS	-.50
t_5	652	SECOND ω_p CORRECTION BEGINS	-.50
t_6	656	SECOND ω_p CORRECTION ENDS	-.49
t_7	760	ω_p CORRECTION FOR < 1 SEC	-.50
t_8	769	DOCKING	-.51

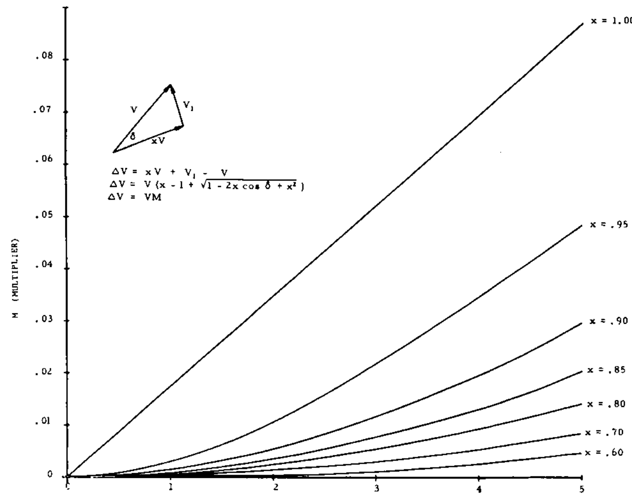


FIGURE 7 δ (deg)

VELOCITY REQUIRED FOR DOG LEG DURING ASCENT

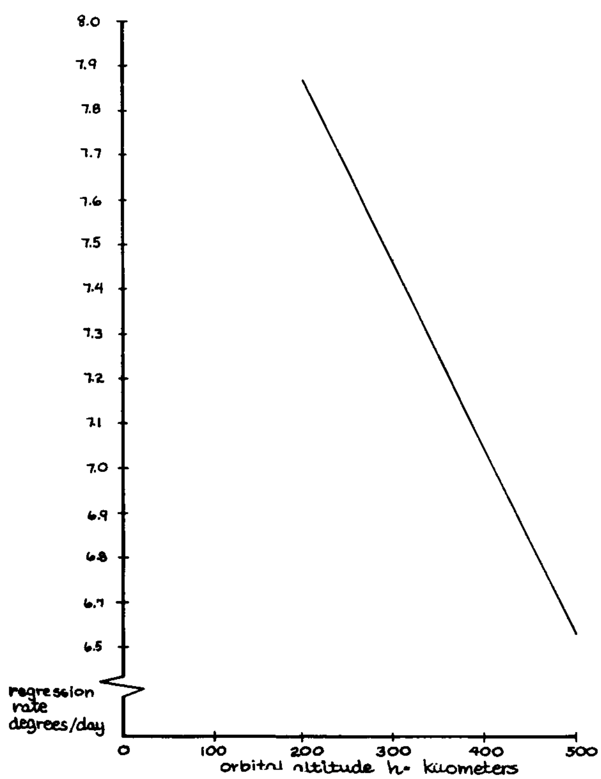
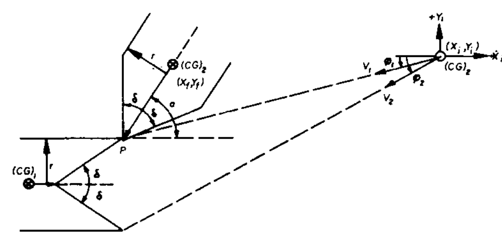


FIGURE 8 NODAL REGRESSION RATE VS. ORBITAL ALTITUDE (i = 28.3°)

DOCKING GUIDANCE ANALYSIS

FIGURE 9



I DOCKING CRITERIA

A. POINT P LIES WITHIN DOCKING CONE AT INITIAL CONTACT

B. EXTENDED VELOCITY VECTOR OF CHASER $(CG)_2$ PENETRATES CONE

II ANALYSIS

$$\left.\begin{aligned} X_f &= X_i + (Y_f - Y_i)\frac{\dot{X}_i}{\dot{Y}_i} \\ Y_f &= Y_i + (X_f - X_i)\frac{\dot{Y}_i}{\dot{X}_i} \end{aligned}\right\} \quad (1)$$

$$\tan\varphi = \frac{\dot{Y}_i}{\dot{X}_i} \quad (2)$$

III APPLYING CRITERIA

A LIMITING POSITION OF (X_f, Y_f) IS ALONG ANGLE $a = 90 - \delta$ WITH P WITHIN THE CONE

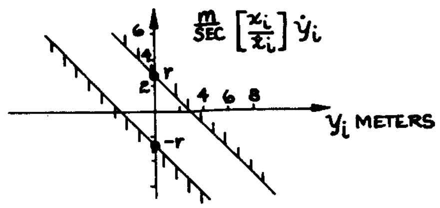
B. $\tan\varphi_1 = \frac{Y_i - r}{X_i} = \frac{\dot{Y}_{i1}}{\dot{X}_i}$

$\tan\varphi_2 = \frac{Y_i + r}{X_i} = \frac{\dot{Y}_{i2}}{\dot{X}_i}$

WHERE $\dot{Y}_{i1} \le \dot{Y}_i \le \dot{Y}_{i2}$ AND $\frac{\dot{X}_i}{X_i}(Y_i - r) \le \dot{Y}_i \le (Y_i + r)\left(\frac{\dot{X}_i}{X_i}\right)$

$\dot{Y}_{i1} = \left(\frac{\dot{X}_i}{X_i}\right)(Y_i - r)$

$\dot{Y}_{i2} = \left(\frac{\dot{X}_i}{X_i}\right)(Y_i + r)$



$$(y_i - r) \le \left[\frac{x_i}{\dot{x}_i}\right] \dot{y}_i \le (y_i + r)$$

Example:
$r = 3$ meters
$\dot{x}_i = 0.5$ m/s
$y_i = 2$ meters

then if $x_i = 50$ meters
$\dot{y}_i$ can have 2 values
$\frac{1}{100} \le \dot{y}_i \,(m/s) \le 5/100$

FIGURE 10

PERMISSIBLE LATERAL VELOCITIES AND POSITIONS AS FUNCTIONS OF CLOSING VELOCITY AND POSITION

INSTRUMENTATION PACKAGE

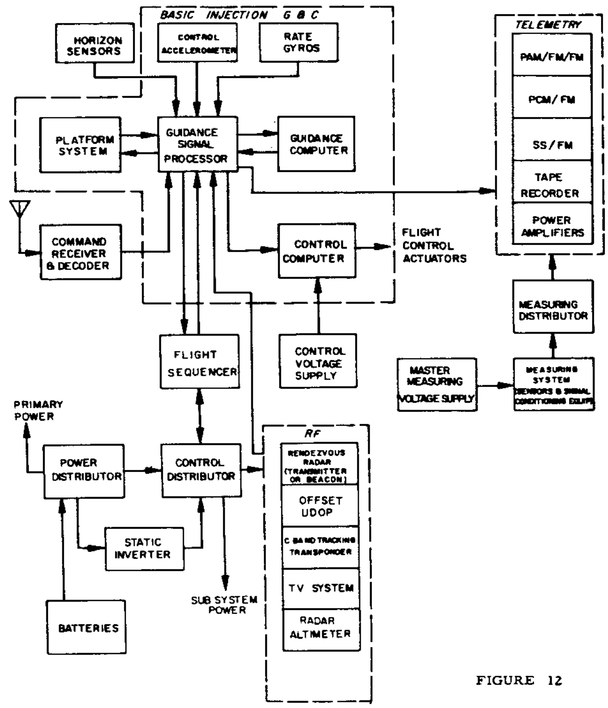


FIGURE 12

INSTRUMENT UNIT LOCATION

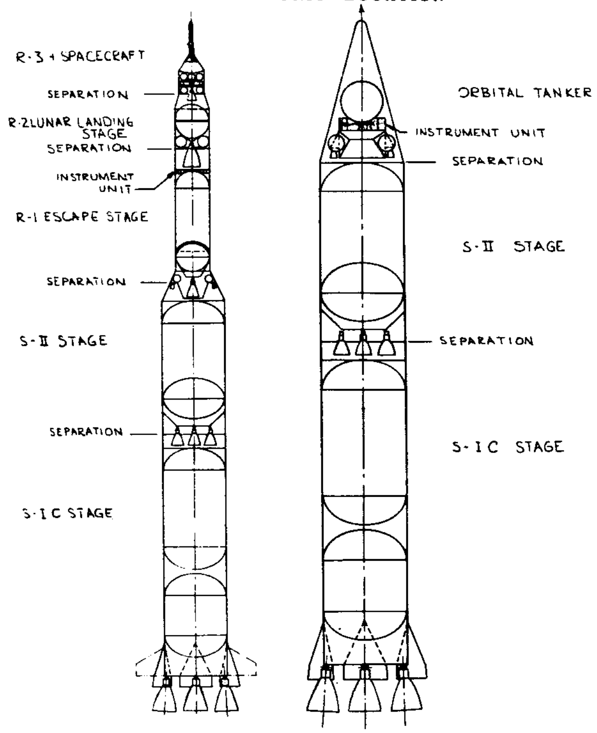


FIGURE 11

NOT TO SAME SCALE

ORBITAL CHECKOUT

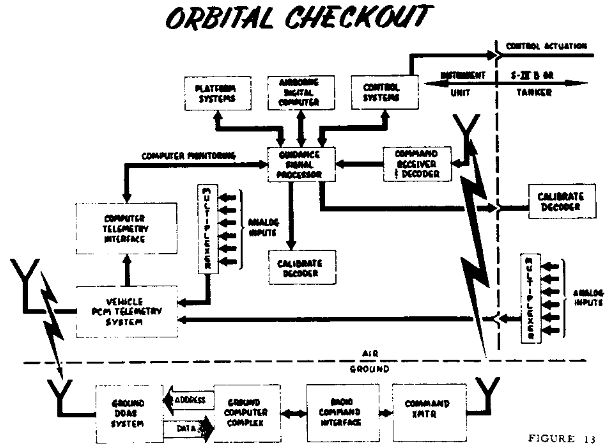


FIGURE 13