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Technical Report

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(U) INTERPLANETARY RE-ENTRY MECHANICS AND ASSOCIATED RECOVERY RANGE COMMAND AND CONTROL APPLICATIONS

OCTOBER 1963

TASKS 6182(22)
6182(78)

AIR FORCE MISSILE DEVELOPMENT CENTER

DEPUTY FOR FOREIGN TECHNOLOGY

AIR FORCE SYSTEMS COMMAND

HOLLOMAN AIR FORCE BASE

NEW MEXICO

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FOREIGN TECHNOLOGY REPORT

AFMDC-TR-63-5

(Title Unclassified)

INTERPLANETARY RE-ENTRY MECHANICS AND ASSOCIATED RECOVERY RANGE COMMAND AND CONTROL APPLICATIONS

October 1963

Tasks 6182(22)
6182(78)

Prepared by:

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(U) PREFACE

The information contained in this Technical Report has been prepared primarily for the use of AFSC personnel engaged in the study of Soviet space technology. This contribution emphasizes postulated planetary re-entry and land area recovery mechanics, and will be of interest to those analysts concerned with future Soviet planetary space technology. (S)

(U) PUBLICATION REVIEW

This Foreign Technology document has been reviewed and is approved for distribution within the Air Force Systems Command. (U)

FOR THE COMMANDER

Howard L Conkey

HOWARD L. CONKEY
Lt Col, USAF
Deputy for Foreign Technology

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(U) SUMMARY

Purpose

This Technical Report was prepared in accordance with requirements established by the Foreign Technology Division in Tasks 6182(22) (page 28), Planetary Exploration System (PES) Command and Control Applications at Recovery Range; and 6182(78) (page 100), Planetary Exploration System (PES) Land Area Recovery Range, of the Soviet Planetary Exploration Program TOPS. (S)

Conclusions

a. Recoverable scientific planetary space missions are believed to be included in future Soviet planning.

b. Recoverable planetary space mission programming is tied directly to state-of-the-art advances in the fields of inertial guidance equipment, tracking technology, power equipment, etc.

c. Optimum launch windows for planetary space vehicles restrict and minimize test conditions for advanced technological concepts.

d. Because of the sophistication of these missions and the estimated size and relative inaccuracies of present Soviet spacecraft guidance/control systems, it is not felt that any such recovery will be attempted by the Soviets prior to 1972.

e. Only meager evidence exists which would suggest that the Soviets intend to expand or otherwise improve their deep space tracking capability, evidently being content for the present with the Crimean facility. ~~(S)~~

Background Highlights

A comprehensive review of both U.S. and Soviet literature indicates that recoverable planetary probes are programmed for future scientific missions. The dates involved, however, will be dependent on the state-of-the-art advances in such critical areas as inertial guidance, tracking systems, power supply systems, electric propulsion, etc. (U)

A brief look at the deep space experiments thus far conducted by both countries gives some insight into the complex mission requirements. Problems encountered by the Soviets in their deep space missions have undoubtedly delayed any time schedule which may have been planned for recoverable planetary missions. Recent proposals in U.S. technology suggests a 1972-75 time scheduling for a recoverable planetary vehicle in this country. This time period, however, is extremely flexible and again is dependent upon state-of-the-art advancements. (U)

Because the U.S. does not have a recoverable planetary vehicle on the drawing board at this time and no information exists in the

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area of Soviet recoverable planetary missions, this report deals primarily with advanced technological theory supported by current Soviet recovery mechanics. (S)

Discussion

The first Soviet attempt to inject a vehicle into a deep space trajectory occurred in 1960. On 10 and 14 October 1960, two ill-fated Mars probes were launched from TTMTR. These vehicles used the Category A ICBM for initial boost and a new "third stage" for injection into an earth orbit. This new stage was previously unobserved and has a thrust of ~65,000 pounds. The first successful flight of this system was on 4 February 1961; however, a fourth stage used to inject the payload from a parking orbit into a Venus trajectory apparently failed on that date. A second Venus probe attempt on 12 February 1961 did achieve a Venus fly-by indicating that all the stages functioned satisfactorily; however, the communications link failed. This deep space probe program has continued in August and September 1962 (3 Venus failures), and October and November 1962 (2 Mars failures and 1 successful Mars probe). There has been no intelligence evidence or official Soviet announcement to indicate that the USSR had any program to launch a recoverable planetary probe. Indeed, the engineering problems of interplanetary navigation, attitude control, communications, re-entry, and recovery are much more complex for such a planetary mission than for near-earth or cislunar missions; any Soviet program to accomplish such a shot would have to be accompanied by a tremendous sophistication of their astro-inertial guidance and spacecraft control system. Such self-contained systems are necessary for a precisely controlled fly-by of another planet, and for a well-controlled return to earth. (S)

SECTION I

INDICATORS OF A RECOVERABLE SATELLITE PROGRAM

Preliminary to the launch of an actual recoverable planetary vehicle will be tests designed to prove the reliability and feasibility of equipment for such missions. Indicators which would reflect that the Soviets are pursuing such a program include:

1. Boosted re-entry ballistic missile firings: Such tests would subject components and vehicles to the re-entry velocities anticipated for interplanetary orbits - on the order of 35,000-43,000 ft/sec. (U).

2. Test orbit: It is possible to launch a recoverable space probe on a round trip orbit into space and back requiring only six months (see Fig. 1). The probe is launched with a velocity relative to the earth which is normal to the ecliptic. Thus the plane of the resulting vehicle orbit makes a substantial angle with the plane of the ecliptic and the orbit has a line of nodes through the launching site. The other end of the line of nodes is through the recovery site which is reached six months later. If the same magnitude of velocity were used for this lobbing shot as is used for a recoverable interplanetary space probe, the vehicle would reach the maximum distance of 15 million miles from

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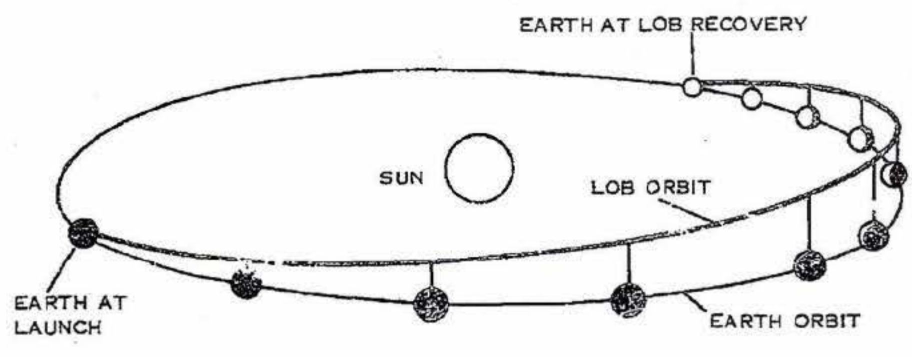

FIG 1 A LOBBING DEEP SPACE PROBE

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earth. The advantage of such a test orbit is that its short range allows a considerable amount of data on vehicle performance to be transmitted back to earth. Further, the use of a directional antenna would not be required; thus, a breakdown in the attitude control system of the vehicle could be diagnosed by telemetry. If such a breakdown occurred at the far greater ranges of an interplanetary probe, it might not be diagnosed inasmuch as the communications channel depends upon the gain of the directional antenna pointed at earth. (U)

3. Development of additional deep space tracking stations: The precise orbit determination required for recoverable vehicles will result in the need for additional sophisticated optical and electronic tracking facilities in the USSR. (U)

4. Structural heating: Many problems exist concerning heating of superorbital re-entry vehicles; a monitoring of this field can yield some insight as to Soviet plans for recoverable planetary missions. (U)

5. For a true interplanetary round trip - such as the example of the earth to Venus surface to earth surface - a tremendous mass ratio is required of chemical propellants. In fact, with a respectable specific impulse of 310 seconds and a

velocity reduction factor C of .85 (this is a design factor dependent upon the velocity losses of each stage due to drag and gravity effects; it normally ranges from .65 to .85), it requires 250,000 pounds launched per payload pound returned. The much simpler mission of earth surface to Venus orbit to earth surface requires 10,000 pounds launched for each payload pound returned (specific impulse of 310 seconds and C = .85). For practical purposes this eliminates chemical propulsion systems in favor of powerful nuclear heat exchanger systems, nuclear electric drives, or eventually some form of fission or fusion pulse drive. Development of these latter engines will be clear indication of a vigorous space exploration program. (~~S~~)

SECTION II

RECOVERY FROM INTERPLANETARY ORBITS

A. Introduction

In any planetary recovery program, there are three basic categories of recovery capsules. In order of increasing size, weight, and complexity, they are:

- the instrument capsule
- the biocapsule
- the astronaut capsule (U)

The primary reasons for recovering instrumented probes include: (1) to determine the effect of the space environment on space-borne equipment; (2) to attempt to simplify the equipment, reduce its complexity, and hence improve its reliability; and (3) to develop recovery techniques for the benefit of future orbital glide programs. The scientific data collected in the vicinity of the target planet will normally have been telemetered back to earth; however, a close review of the actual package will enhance the value of the collected information. Future experiments will permit the development of biosatellites with the assurance of regaining the specimens for proper evaluation of results. Further, a well-designed space capsule with a low $\frac{W}{C_D A}$ may become for the astronaut what the parachute is to the aviator - a chance for survival in case of failure of the main vehicle. Such a human-carrying biocapsule would have to be equipped with stabilizers, jet and aerodynamic, to prevent rotation and tumbling. (U)

B. Ballistic Vehicles

Our discussion will be limited to the recovery of purely ballistic interplanetary vehicles inasmuch as that will most probably be the design of all initial vehicles in a space recovery program. The sphere is the present prototype of a ballistic space capsule which can be used to advantage for recoverable planetary missions. Of course, every symmetric body at zero angle of attack is also a nonlifting (i.e., ballistic) body and could be utilized for the flight. For a sphere, the drag coefficient stays close to unity most of the time; and if it loses little weight due to ablation, its ballistic coefficient, $\frac{W}{C_D A}$, remains essentially constant. (U)

The trajectory which a typical superorbital (planetary) ballistic re-entry vehicle follows is defined by:

- initial re-entry velocity, v_i
- initial re-entry angle, γ
- ballistic coefficient of vehicle, $(\frac{W}{C_D A})$

Most re-entry vehicle trajectories are treated as a two-body problem, involving a central force field, with the assumption of a nonrotating atmosphere and zero thrust. Starting with a known initial velocity, altitude, and angle between the velocity vector and the local horizontal, trajectory parameters determined are velocity, altitude, flight path angle, tangential acceleration, rate of altitude change and the geocentric angle - all as functions of the independent variable time (see References 13 and 15). As can be seen, these variables will provide an infinite number of re-entry trajectories, until the actual vehicle and mission parameters are described. As a practical matter, re-entry angles will probably be kept under -10°, with initial velocities on the order of 35,000 to 43,000 ft/sec. (U)

C. Re-Entry Corridor

Well known is the fact that the re-entry angle determination is quite critical. If entry is too shallow, the vehicle will lightly pierce the atmosphere and continue out into space (or into a geocentric orbit); on the other hand, a too-large entry angle yields heating and structural (deceleration) problems which could adversely affect the vehicle. This gives rise to the concept of the re-entry corridor which is defined by an undershoot and overshoot boundary. The corridor width, w, is given as the distance between the conic perigees of the two boundaries (see Fig. 2). As is apparent from

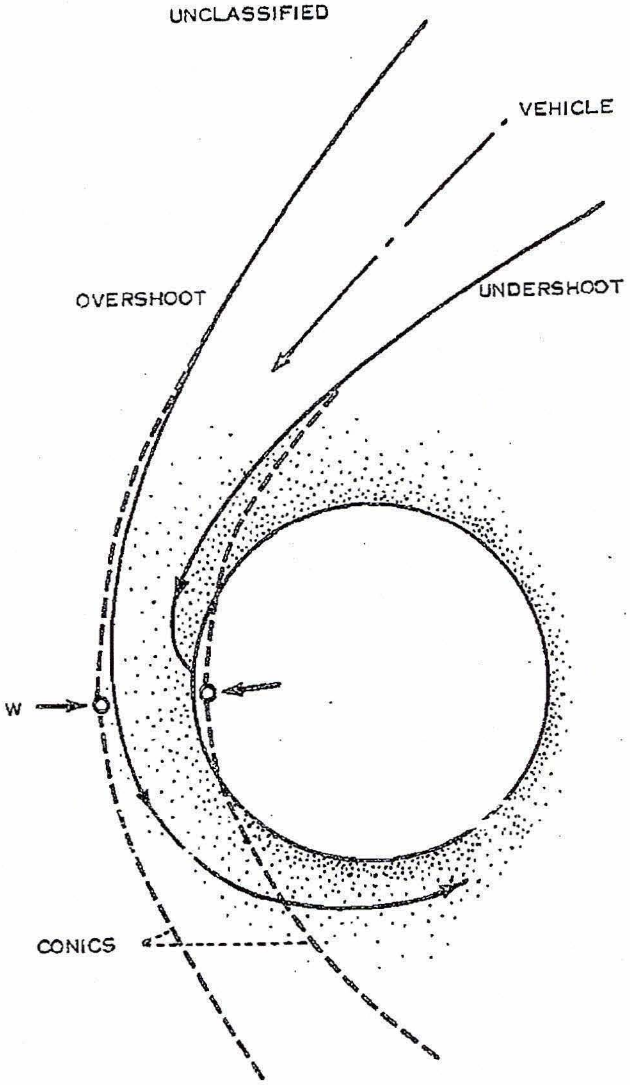

FIG 2 LANDING CORRIDOR WIDTH

the preceding discussion, the width of the corridor will be a function of entry velocity, entry angle, and ballistic coefficient. (Lift/drag ratios and modulation methods also influence w for nonballistic bodies.) Consequently, for a given vehicle and entry velocity, the corridor may be represented by a range of acceptable entry angles. Conversely, if the range of acceptable entry angles is known, the corridor width is determined by calculating the Keplerian perigees and measuring the difference between perigee altitudes. Corridor width, as defined by either w or γ may therefore be used to stipulate the maximum tolerable errors for a guidance system. (U)

D. Space Vehicle Velocities

The velocities associated with space missions are:

circular: $v^2 = \frac{K}{r_1}$

ellipse: $\frac{K}{1} < v^2 < \frac{2K}{r_1}$

parabolic: $v^2 = \frac{2K}{r_1}$

hyperbolic: $v^2 > \frac{2K}{r_1}$

where K is the gravitational parameter, a characteristic constant of the attracting body M (for earth it is $1.407523 \times 10^{16}\ ft^3/sec^2$) and r_1 is the radial distance from the attracting focus. All earth satellites possess either circular or elliptical velocities. In a parabolic orbit, the vehicle's entire kinetic energy is used up in overcoming the central body's entire potential energy between the instantaneous distance and infinity. The vehicle therefore has a velocity of 0 at "infinity" - that is, after escape. In a hyperbolic orbit, the vehicle has more kinetic energy than necessary for overcoming the gravitational potential of the central body. Therefore, even after escape from the body, the vehicle has a finite remaining velocity which is used to change its orbital energy with respect to the central force field of the next higher order - namely, the solar field. (U)

E. Re-Entry Maneuvers

The maneuvers that are required to convert the hyperbolic approach orbit which develops as the vehicle begins to respond to the influence of the gravitational field of its destination to an orbit which leads to safe penetration of the planetary atmosphere will now be discussed (see Fig. 3). The criteria for a satisfactory landing trajectory include the concept of a landing corridor described above. Safe landing paths are correlated with the configuration of the spacecraft and with the altitude of the Keplerian perigees for the approach trajectory. This is the perigee which would occur if the approach trajectory were not perturbed by the effects of

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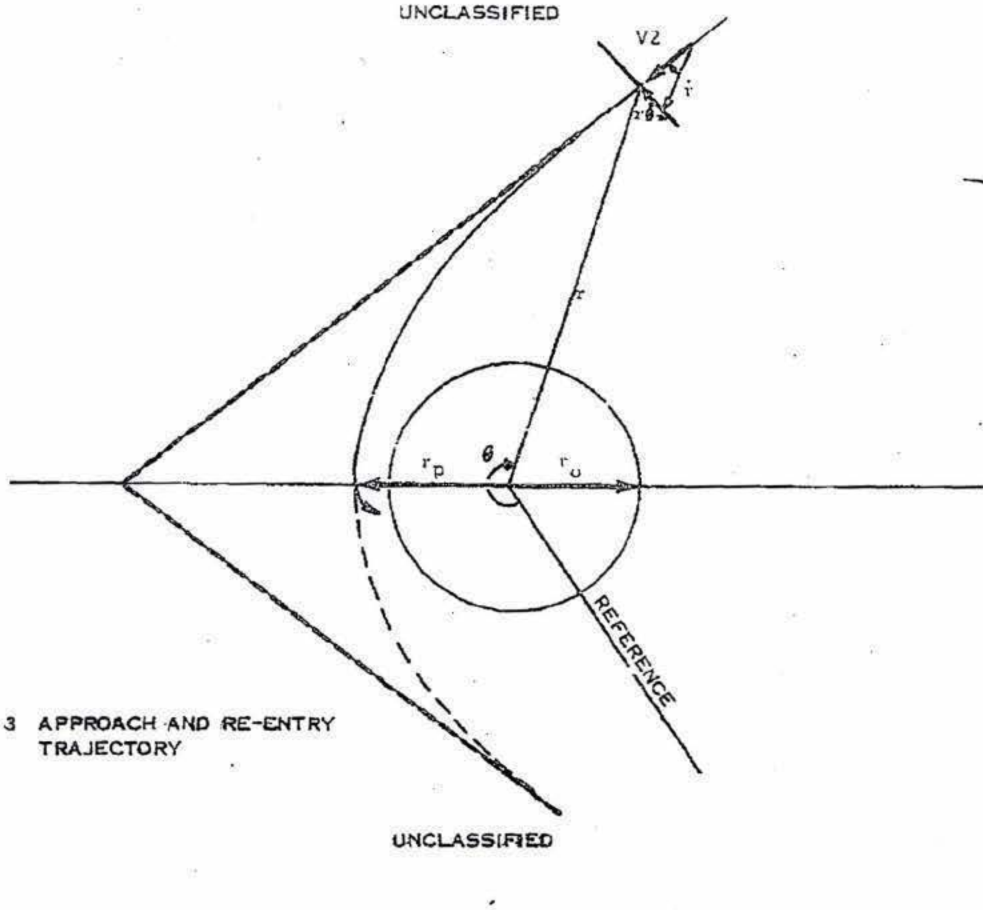

FIG 3 APPROACH AND RE-ENTRY TRAJECTORY

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atmospheric drag. Using this correlation, the control of the spacecraft to place it in a landing path will be accomplished if it is controlled in such a way as to place its Keplerian perigee within a designated landing corridor. This is accomplished by maneuvers which modify the approach trajectory until the orbit relations which may be measured indicate a value of perigee radius, r_p, and velocity, v_p, which are within the indicated landing corridor. It is envisioned that such a return problem will be accomplished by the application of corrections (coarse and fine) to the return trajectory, as follows:

1. First corrections are applied at a point sufficiently distant to assure that the trajectory will close with the earth such as to allow fine corrections at a later time. This first correction is rather simple in application and for best results, it will be performed at the greatest possible distance from earth in order to conserve fuel. The distance is limited by the accuracy with which the correction can be computed and applied. (U)

2. The second correction adjusts the approach trajectory causing it to pass through the narrow landing corridor. This correction is applied fairly close to earth when precision measurements of the trajectory are feasible. Without the first correction,

excessive propulsion energy might be required to perform the second correction. (U)

3. The third correction is required to convert the approach orbit from one which has its perigee in the chosen landing corridor to one which actually re-enters the atmosphere and is recovered. This will normally be accomplished by retrorocket deceleration, but in some cases, an atmospheric-induced drag could be used. (U)

The accuracy of the guidance required for landing control is thoroughly reviewed in a NASA report by Chapman (Reference 3). The consequences of an inaccurate solution to the re-entry homing problem are considered to be intolerable; therefore, the use of an open loop solution is rejected in favor of a closed loop guidance system. In the latter system, flight conditions are continuously monitored and corrections applied as needed; this system requires an accurate prediction system on board to forecast motions of the vehicle. Corrections may be introduced through application of continuous thrust or through impulses at selected intervals. (U)

F. Re-Entry Guidance System

A postulated interplanetary homing re-entry guidance scheme using on-board components is depicted in Fig. 4. Measurements of range, r, range rate, $\dot{r}$, and the rate of change of the line of

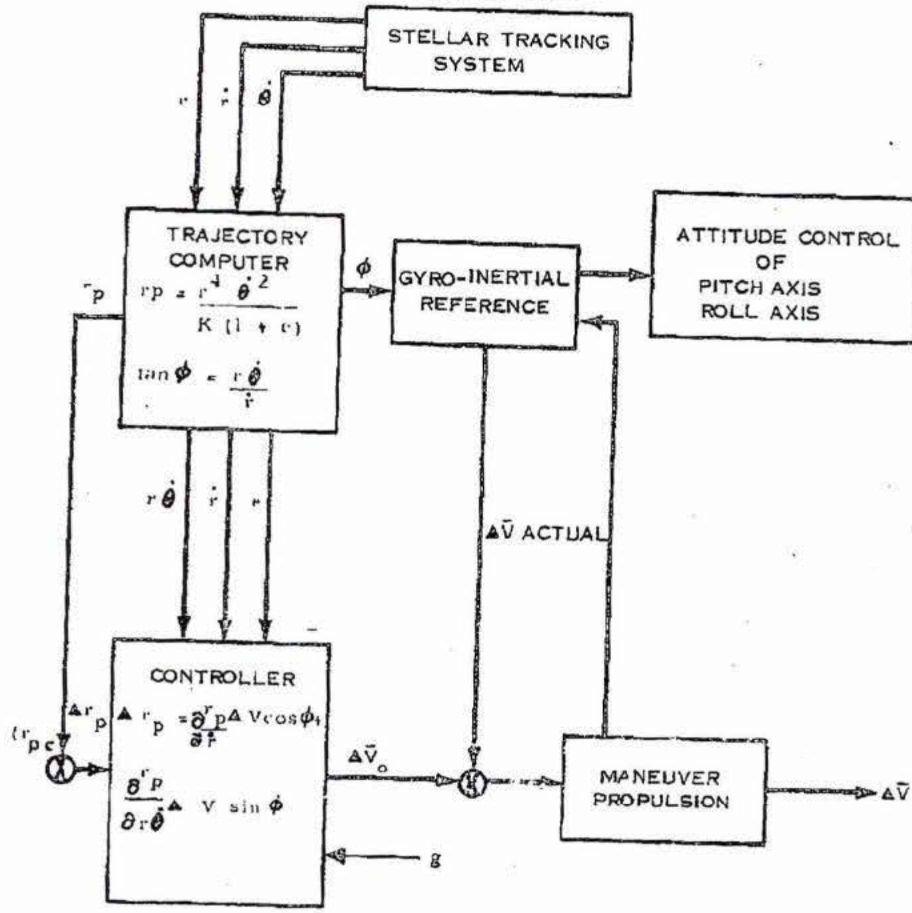

FIG 4 ON-BOARD LANDING GUIDANCE SYSTEM FOR INTERPLANETARY VEHICLE

sight, 0, are required in order to compute the necessary flight parameters for determination of the maneuvers required. (U)

The optimum time for conversion from the hyperbolic arc approach trajectory to a planetary orbit or landing ellipse is at the Keplerian perigee. The error in prediction of the perigee will be related to the errors in measurement of the orbit parameters. Deviations between the perigee of the approach trajectory and the center of the landing corridor are expressed in terms of these parameters by

$$\Delta r_p = \frac{\partial r_p}{\partial r} dr + \frac{\partial r_p}{\partial r} d\dot{r} + \frac{\partial r_p}{\partial \theta} d\dot{\theta}$$

where: $$\frac{\partial r_p}{\partial r} = \frac{2r_p(1+e)}{re} - \frac{r\dot{\theta}^2}{ge} - \frac{r_p^2}{er^2}$$

$$\frac{\partial r_p}{\partial \dot{r}} = - \frac{\dot{r}}{eg}$$

$$\frac{\partial r_p}{\partial \dot{\theta}} = \frac{1}{\dot{\theta}}\left[r_p \left(\frac{1+e}{e}\right) - \frac{r^2\dot{\theta}^2}{eg}\right] \qquad \text{(U)}$$

Unit contributions to the error in predicted perigee define a quantity which may be designated the coefficient of unit error. This coefficient expresses the error in measurement of a single flight coordinate which will contribute an error or unit magnitude in the desired parameter. Since the desired parameter is the perigee distance, r_p, the coefficients for the re-entry guidance example are:

Error in r leading to unit error in perigee:

$$\delta r = \frac{1}{\partial r_p / \partial r}$$

Error in $\dot{r}$ leading to unit error in perigee:

$$\delta \dot{r} = \frac{1}{\partial r_p / \partial \dot{r}}$$

Error in $\dot{\theta}$ leading to unit error at perigee:

$$\delta \dot{\theta} = \frac{1}{\partial r_p / \partial \dot{\theta}}$$

For an earth approach trajectory, these coefficients of unit error have been calculated and are shown in Table I. From the data listed in the table, it is apparent that although the requirements are exacting, instruments may be constructed which will provide the required accuracies to permit adjustment of the space vehicle's trajectory in order to place its perigee within a landing corridor which is 5 to 10 miles wide, provided that a one micro-radian accuracy may be obtained from optical measurements of a planet disk. Accuracy of instruments to meet this quality has been predicted in U.S. state-of-the-art journals. Range rate, $\dot{r}$, accuracy requirements will be satisfied if the differentiation intervals of 100 seconds or more are employed. As the vehicle

TABLE I

UNIT ERROR VALUES ALONG AN EARTH APPROACH TRAJECTORY

	$r/r_o = 100$	$r/r_o = 50$	$r/r_o = 25$	$r/r_o = 10$
Error in r leading to a 1-mile error in r_p	0.3 mile	0.15	0.075	0.0375
Error in $\dot{r}$ leading to a 1-mile error in r_p	0.0159 mi/sec	0.0156	0.0155	0.0153
Error in $\dot{\theta}$ leading to a 1-mile error in r_p	$0.145X10^{-9}$ rad/sec	$4.85X10^{-9}$	$19.7X10^{-9}$	$100X10^{-9}$

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SECTION III

(U) PLANETARY LAND RECOVERY RANGE

In planning future programs, such as recoverable planetary probes, the results are contingent upon the immeasurable time-phasing of advanced technology. (U)

Soviet literature does not discuss in any detail the preplanning for future planetary programs. It does, however, clearly indicate that the future of the Soviet planetary research program is currently dependent upon successful planetary observation probes. In attempting to fulfill this requirement, the Soviets have attempted ten planetary missions of which only two were partially successful. ~~(S)~~

Based on current technology in the areas of guidance, tracking, communications, and propulsion, combined with the difficulties being encountered in their planetary program, it is not believed that recoverable planetary missions are planned for the 1964-72 time period. (U)

By way of comparison, NASA-advanced studies do not include recoverable planetary probes during this time period. The last planetary system currently on the drawing board is the Voyager vehicle which is ultimately designed to orbit its intended planet

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as an observation probe with the future possibility of ejecting capsules for planetary impact. (U)

For the purposes of this report, it is assumed that scientific interest in the planets will continually increase and a recoverable planetary probe will ultimately be developed by the Soviets. The need for a land recovery area will be a natural consequence of such a program. (U)

As pointed out in a previous AFMDC technical report (AFMDC-TR-63-1) concerning postulated land recovery areas for lunar vehicles, the best suited locale for recovery within the USSR is bounded by 56°N-60°E, 56°N-80°E and 44°N-60°E, 44°N-80°E. This conclusion was reached by reviewing the following standard site selection criteria and using it as a working base:

1. Security.
2. Safety.
3. Terrain.
4. Climatology.
5. Logistic Support.
6. Recovery Command and Control Network.
7. Search and Recovery Network. (S)

These parameters, although critical to the success of any recoverable mission, are essentially controlled by the vehicle design characteristics and the geometrical constraints of the entry mission. Without these parameters, it is impossible to accurately predict a land-based aim point. (U)

As discussed in Section II, it is believed that the Soviets will initially utilize a pure ballistic recoverable planetary probe. Use of such a vehicle narrows the tolerable entry corridor into the earth's atmosphere and places final impact accuracy in the hands of very accurate on-board guidance/control systems and deep space tracking stations. These stations will be responsible for determining guidance corrections necessary prior to earth entry on the final leg of the trajectory. (S)

Fig. 5 points out those areas of the USSR which most closely satisfy all standard land recovery range site selection parameters without respect to postulated guidance accuracies. This figure was originally derived for a recoverable lunar vehicle with controllable entry, thus the three impact areas. It is believed that in the 1972-era, or before if technology permits, the Soviets will have tracking accuracies and propulsion systems capable of achieving entry into the larger range area as depicted in the figure. As pointed out in AFMDC-TR-63-1, the optimum recovery area lies within the primary zone and constitutes an optimum aim point for recoverable planetary probes as well as lunar return vehicles. (S)

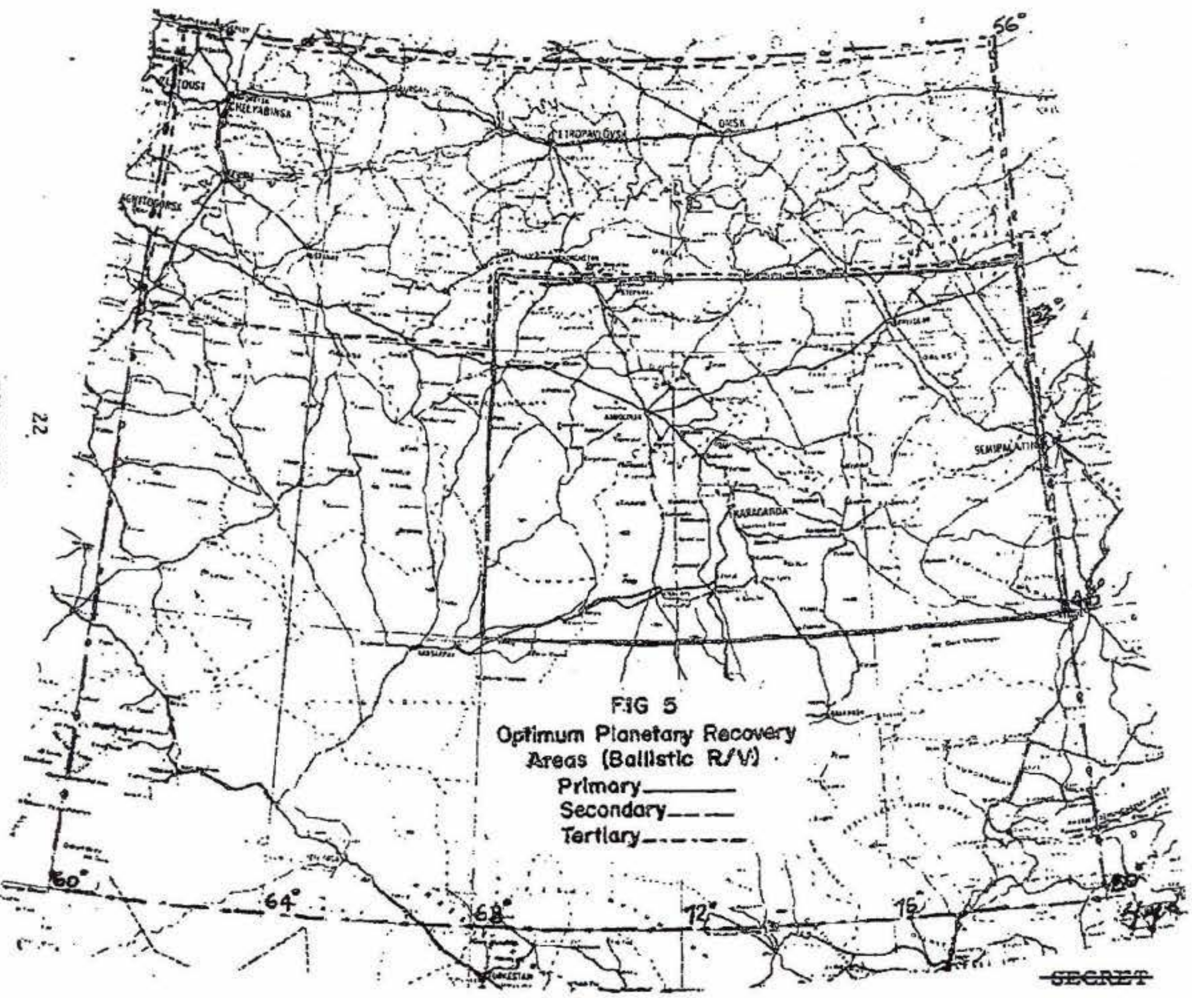

FIG 5
Optimum Planetary Recovery
Areas (Ballistic R/V)
Primary
Secondary
Tertiary

Due to the guidance inaccuracies anticipated during initial interplanetary flights, provisions for emergency landing sites should also be considered. Recovery within any land mass in the Soviet Union or Soviet Bloc countries could provide relative security to the mission and in most cases still be located by mobile recovery forces. Water impact, an additional hazard, should also be considered during early missions. The recovery vehicle should be capable of surviving a water impact in case of emergency and still provide adequate safety to its scientific payload. In addition, recovery forces should be capable and deployed for water retrieval as well as land recovery. ~~(S)~~

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SECTION IV

(U) PLANETARY RECOVERY RANGE COMMAND AND CONTROL

As previously discussed, the engineering problems connected with recoverable planetary flight are much more complex than those encountered in recoverable earth orbit programs. When discussing recovery range programming, however, the command and control aspects need not be more complex in design. (U)

Assuming that the Soviets will use a ballistic type re-entry vehicle during the initial phases of the program, the current command and control network should prove adequate for assuring timely recovery. Fig. 6 shows the command and control network which is believed used by the Soviets during current earth orbit recovery exercises. ~~(S)~~

The existing structure enables the Soviets to have design simplicity combined with operational effectiveness. This system is believed to make use of a recovery range controller who exercises overall control of all search/recovery forces in the planned impact zone, and at the same time receives computed impact data and first echelon directions from the mission control center. [redacted]

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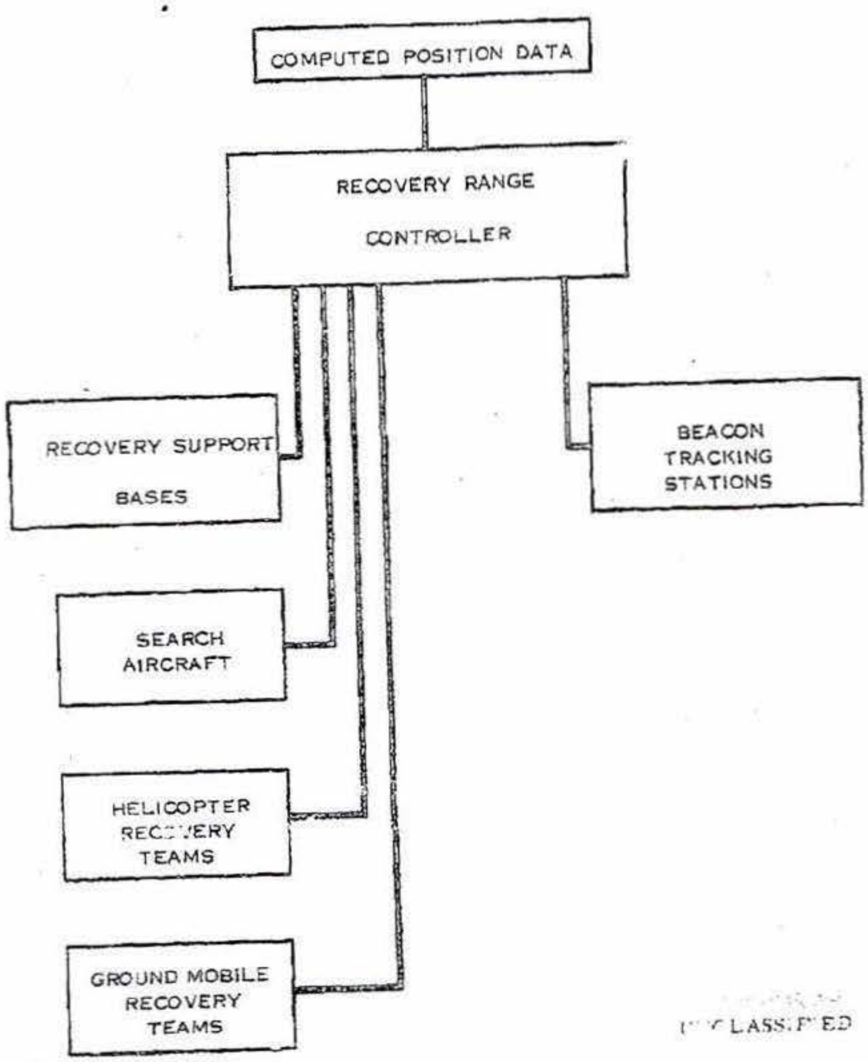

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FIG 6 (U) RECOVERY RANGE COMMAND AND CONTROL NETWORK

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[redacted] This technique would provide the central controller with timely directional information and thus enable him to dispatch search recovery forces to the general recovery area almost immediately. (S)

Although the recovery of planetary probes is not expected to alter the current recovery range command and control structure, it will probably introduce some complexity into the operational aspects of recovery. As discussed previously, the tracking and guidance inaccuracies inherent in a ballistic planetary re-entry system are such that the proposed recovery zone will probably be increased in size. This will constitute a requirement for additional personnel, equipment, and staging areas in order to provide more broad recovery range coverage. Due to the predicted time lag between launch and recovery of planetary probes, initial deployment of the recovery forces will also be governed by the guidance accuracies achieved throughout the entire flight. The recovery forces should be moved into the preplanned recovery zone no later than one week prior to the calculated re-entry date. Check-out of the recovery range command and control network could be accomplished during the interim period with redeployment carried out if necessary. (U)

Even though the use of a ballistic re-entry vehicle alleviates the need for range controlled terminal guidance applications during re-entry, it places extreme accuracy requirements on the deep space tracking facilities. As pointed out earlier, final impact accuracy will be dependent upon the tracking and guidance accuracies achievable during the final leg of planetary flight. (U)

Soviet literature has on occasion credited the deep space tracking facility in the Crimea with the capability of tracking planetary probes. Although the tracking accuracy of equipment located at this facility is currently unknown, some insight may be gleaned from Soviet releases at the July and October 1961 international scientific meetings in Washington D. C. They reportedly gave the following data on the Soviet interplanetary tracking radar:

Antenna type - Tracking dish

Radio frequency - About 700 mc/s

Power flux density - 250 mw/steradian

Oscillator stability - 1 part in 1 billion

Duty factor - 0.5

Pulse length - 64 or 128 milliseconds

Transmit polarization - Circular

Receive polarization - Linear

Power incident on surface of Venus at very center of beam illumination - 15 watts

Although these characteristics cannot be equated to any one Soviet radar, it is believed that the Soviets have three deep space tracking radars with the necessary large tracking dishes (nominal 72'), one each located at Moscow, Novosibirsk, and in the Crimea. ~~(S)~~

The use of these radars could provide the Soviets with deep space position information but are not presently believed capable of accuracies necessary for recoverable planetary vehicles. ~~(S)~~

Additional information suggests that the Soviets could be attempting to establish tracking facilities in both Chile and Indonesia. If such a network could be established, it would probably closely approximate the U.S. Deep Space Instrumentation Facility (DSIF) with stations at JPL, Goldstone Facility, California; Woomera, Australia; and Johannesburg, South Africa. This would then provide the Soviets with worldwide tracking coverage useful in both near and deep space missions. ~~(S)~~

If such a space tracking network is intended by the Soviets, all tracking information would probably be fed into a central data reduction center similar to the U.S. Goddard Space Flight Center.

As shown in Fig. 7, this center would then act as overall mission controller directing all command and control aspects of the operation. (U)

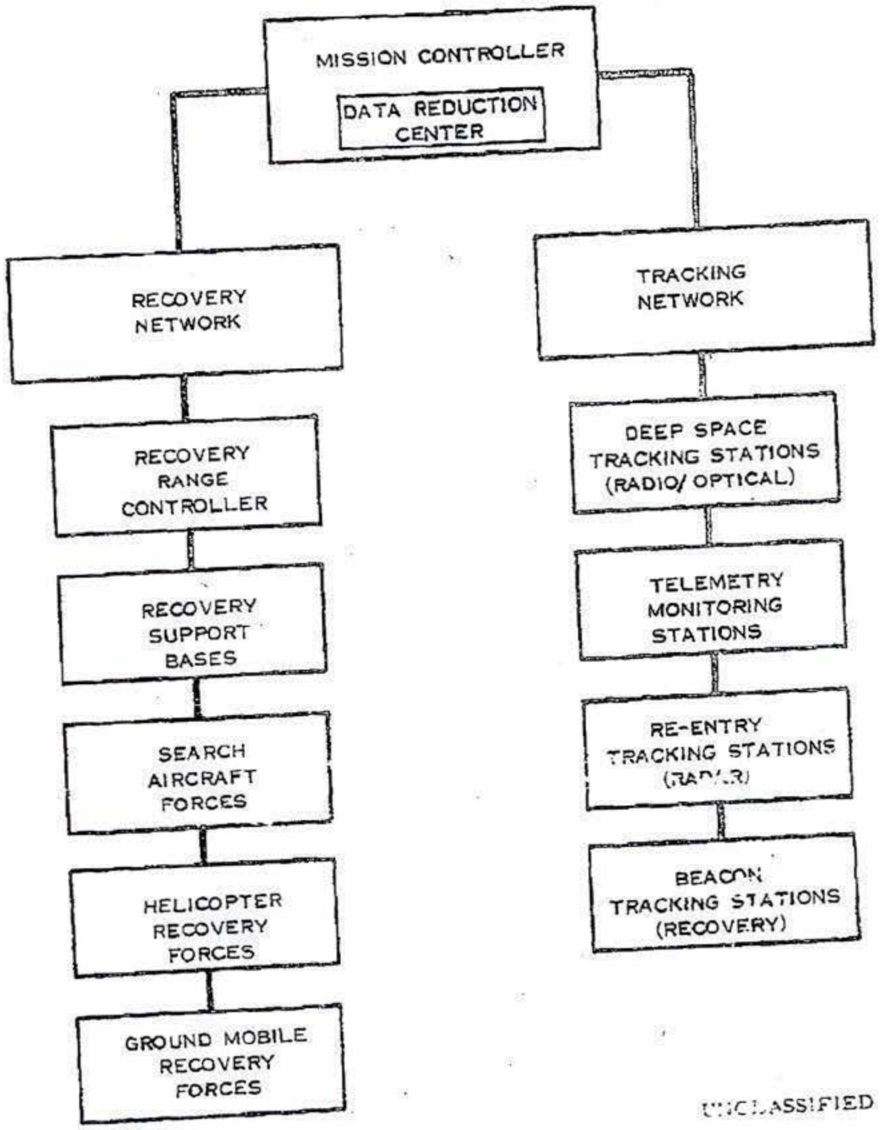

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FIG 7 (U) MISSION COMMAND AND CONTROL

APPENDIX I

GLOSSARY OF TERMS

A - reference frontal area

C - velocity reduction factor

C_D - drag coefficient

e - eccentricity

g - acceleration due to gravity of body

K - gravitational parameter

M - mass

r - radial distance (range)

r_o - radius of planet

r_1 - radial distance from principal focus

r_p - radial distance to periapsis

$\dot{r}$ - range rate

v - scalar velocity

$\bar{v}$ - vector velocity

v_p - velocity at periapsis

v_i - initial re-entry velocity

W - weight

w - corridor width

γ - initial re-entry angle (flight path angle)

θ - line of sight angle, angular distance along trajectory measured at principal focus from a reference direction to position vector of the vehicle

$\dot{\theta}$ - turning rate of the line of sight

ϕ - elevation angle, coflight path angle measured from local horizontal (where horizontal is defined as the plane normal to the geocentric position vector of the vehicle)

~~SECRET~~

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AREA MAP SHOWING LOCATION OF AFMDC