DST-1000S-293-76

DEFENSE INTELLIGENCE AGENCY

BALLISTIC MISSILE GUIDANCE AND CONTROL — USSR AND PRC (U)

Appendix H

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BALLISTIC MISSILE GUIDANCE AND CONTROL - USSR AND PRC

DST-1000S-293-76

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PREFACE

The following definitions apply to terms used in this study:

1. Estimate - Judgment regarding the present or a historical period about which there is substantial validated information; extended to include the next 2 years.

2. Projection - Judgment regarding the time frame from 2 years hence with decreasing confidence as the time span increases.

3. Almost Certain - Indicates approximately a 90:10 chance that an event will occur. Related terms: will, shall, is expected, is anticipated.

4. Probable - Indicates approximately a 75:25 chance that an event will occur. Related term: likely.

5. 50:50 - Even chance an event will occur.

6. Unlikely - Indicates approximately a 25:75 chance or less; i.e., only one chance in four it will happen. Related term: improbable.

7. Better Than Even Chance - Greater than a 50:50 chance but less than approximately a 75:25 chance.

8. Less Than Even Chance - Less than a 50:50 chance but greater than approximately a 25:75 chance.

This revision contains significantly new and revised information which affects previously published assessments. A delta (Δ) preceding a paragraph, heading, or caption indicates that the discussion, table, or figure is reporting significantly new or revised information.

METRIC CONVERSION TABLE

Parameters and measurements in this document are reported in metric units, to bring Department of Defense intelligence reporting into consonance with internationally accepted standards for measurement. For the benefit of users who require parametric values expressed in customary units, the accompanying table lists each metric unit applicable to this document, along with its corresponding value in customary units.

The metric system used is the International System of Units (SI), along with certain non-SI metric units approved by the International Committee on Weights and Measures (ICWM) for use with the SI system. Certain non-metric, international customary units such as degree of arc, and the customary units of time also have been approved by ICWM for continued use with the SI system and will continue to be employed where appropriate. US customary units also will continue to be employed for items which are defined or named in terms of customary units, such as 1-megaton warhead, .38-caliber pistol, and two-by-four. An asterisk (*) in the Customary Unit's column indicates that the customary unit will continue to be used for the indicated parameter.

PARAMETER	VALUE IN METRIC	METRIC UNIT	METRIC SYMBOL	VALUE IN CUSTOMARY	CUSTOMARY UNIT
		ACCELERATION, SPEED, VELOCITY			
Acceleration	1	meter/second^2	m/s^2	3.280840	ft/sec^2
Velocity	1	meter/second	m/s	3.280840	ft/sec
		RANGE, ALTITUDE, DIMENSION, LENGTH			
Range/Distance	1	kilometer	km	0.5399568	NM
Altitude dimension	1	meter	m	3.280840	foot
Dimension	1	millimeter	mm	0.03937008	inch
Thickness, wavelength	1	micrometer	μm	0.03937008	mil
		MASS (WEIGHT)			
Gross weight, payload	1	kilogram	kg	2.204623	pound (m)
Mass (bulk)	1	metric ton (tonne)	t	1.102311	ton
		ENERGY (J = W · s = N · m), TORQUE (N · m)			
Torque	1	newton-meter	N · m	0.7375621	lbf-ft
		PRESSURE, STRESS, STRENGTH (Pa = N/m^2)			
Pressure (gage)	1	kilopascal	kPa (gage)	0.1450377	psig
Pressure (absolute)	1	kilopascal	kPa (abs)	0.1450377	psia
		ANGLE			
Plane angle	1	radian	rad	57.29579	*degree
Angular velocity	1	radian/second	rad/s	9.549279	*rpm

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SECTION I

SOVIET SURFACE-TO-SURFACE BALLISTIC MISSILE GUIDANCE AND CONTROL (U)

1. Introduction

A method of improving the accuracy of an inertial guidance system, especially mobile launchers, is to use a star tracker. A star tracker can correct for both position and azimuth errors. It can greatly reduce the need for frequent updatings of the navigation system on the mobile launcher. The amount of improvement depends on the quality of the inertial components used in the inertial guidance system on the missile and in the navigational system used on the mobile launcher.

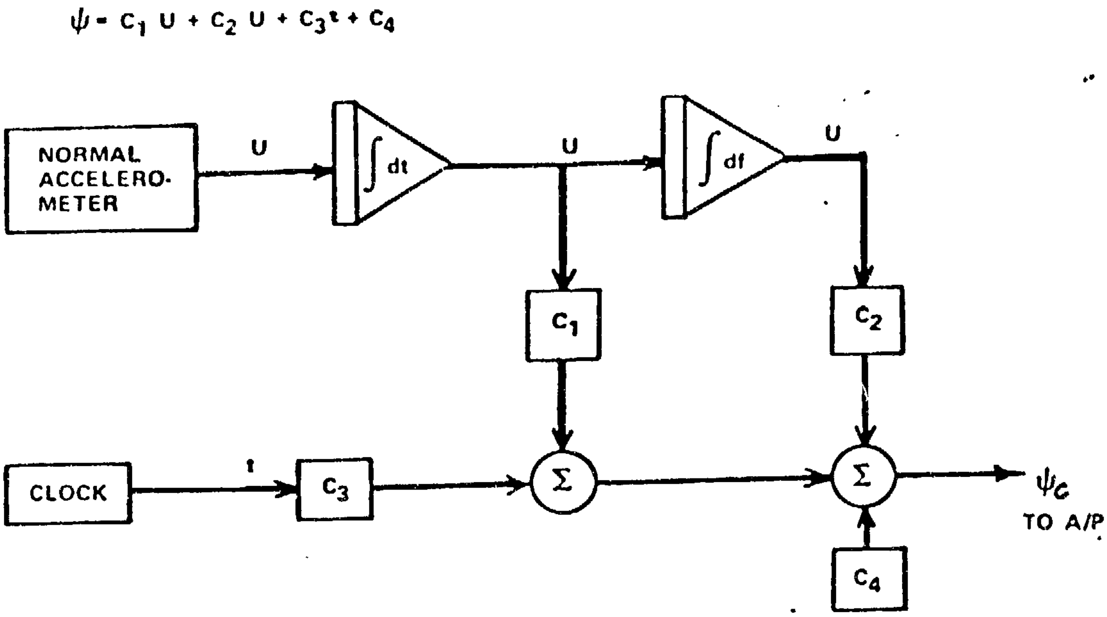

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Fig. 16 Pitch Steering Loop

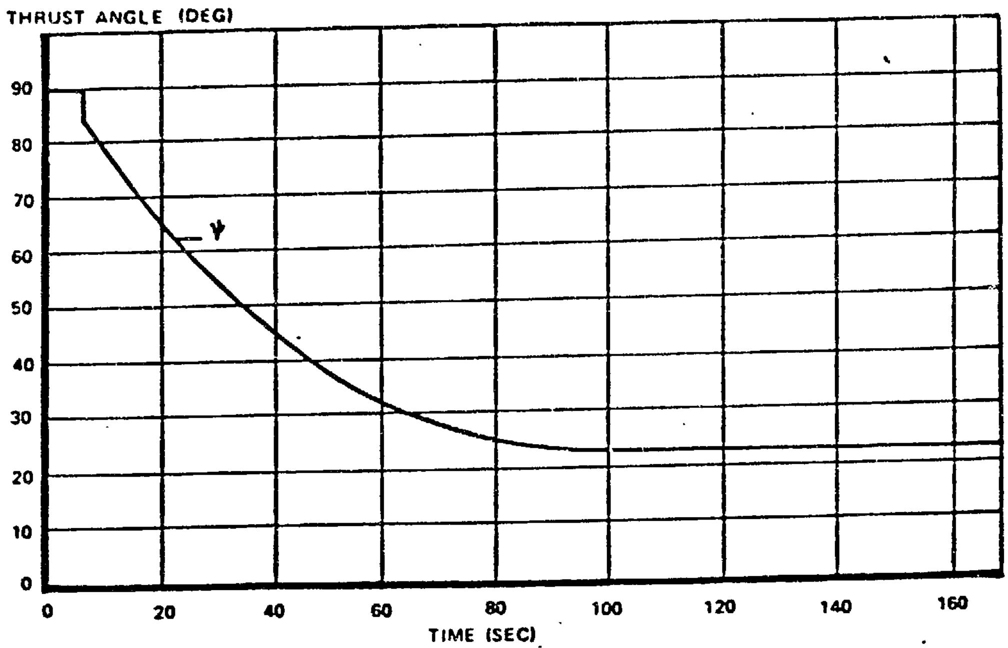

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Fig. 17 Nominal Pitch Command

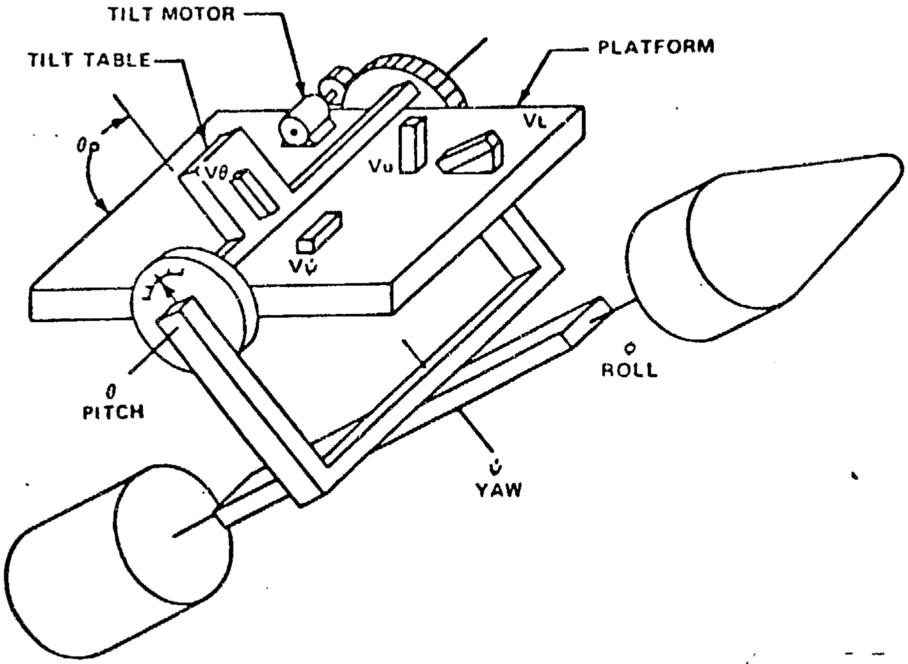

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Fig. 18 Stable Platform After Pitchover

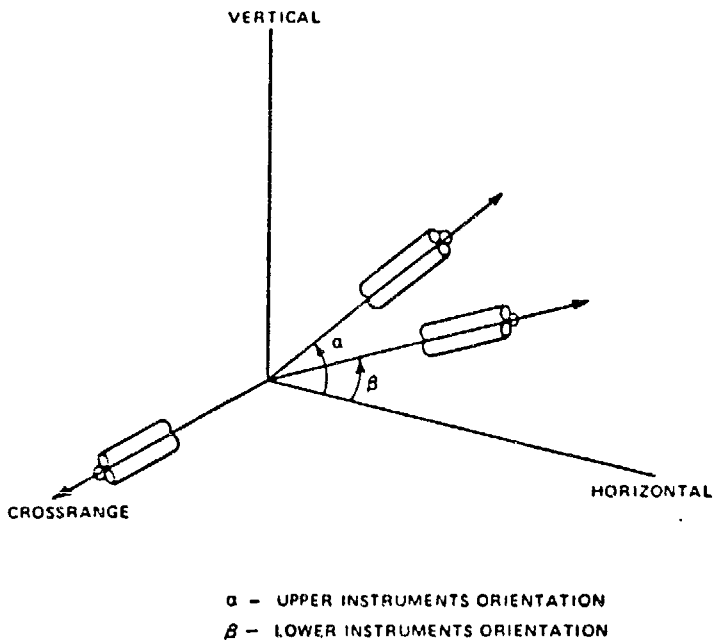

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Fig. 20 Accelerometer Orientations

A possible guidance system limitation does exist for any future MIRV use; i.e., use against targets hundreds of miles apart. The guidance concept now employed may not easily adapt to MIRV use. The introduction of a high capacity digital computer and a change in guidance philosophy will likely have to take place to obtain the flexibility and onboard navigation capability required for advanced multiple RV delivery systems.

Gyro error sources (Table XII) can be separated into two main categories: (1) systematic torques and (2) random torques.

Systematic torques are defined as torques which can be measured and whose characteristics can be correlated with some parameter. It is theoretically possible to compensate either directly or indirectly for the errors produced by the systematic torques. The drifts due to systematic torques are basically of two types: (1) acceleration-sensitive drifts generally caused by mass unbalance, and (2) acceleration-squared-sensitive drifts generally caused by structural compliance-anisoelasticity.

Random or unpredicted variations will introduce uncompensated errors which must be charged to the gyro. These errors are defined as "constraint" torques and are the most significant errors. "Constraint" errors are the limiting accuracy factor affecting gyro performance.

Δ TABLE XII

SOVIET INERTIAL COMPONENT TECHNOLOGY CAPABILITIES

The literature indicates that the Soviets have a good mastery of the theory of gas bearings, and the techniques they are reportedly investigating closely parallel European and US designs. There are many Soviet publications on gas bearings dating back as far as the early 1950's.

Successful developments of these instruments could lead to their use in ballistic missile guidance systems where high accuracy, low cost, ease of maintenance, and reliability are extremely important.

The technical literature on the ESG has indicated the Soviets have a good understanding of its potential capabilities. Since the late 1960's, the Soviets have progressed rapidly from treating the basic operating principles of suspension to analyzing the more complex problems such as resistance to vibration and acceleration, manufacturing tolerances, deformations under rotation, residual friction effects upon duration of spin, pickoff, and control torquing; they appear to be well along in the development. It is apparent that they recognize that a drift rate of 0.0001 degree per hour can be achieved with the ESG.

Δ TABLE XVI

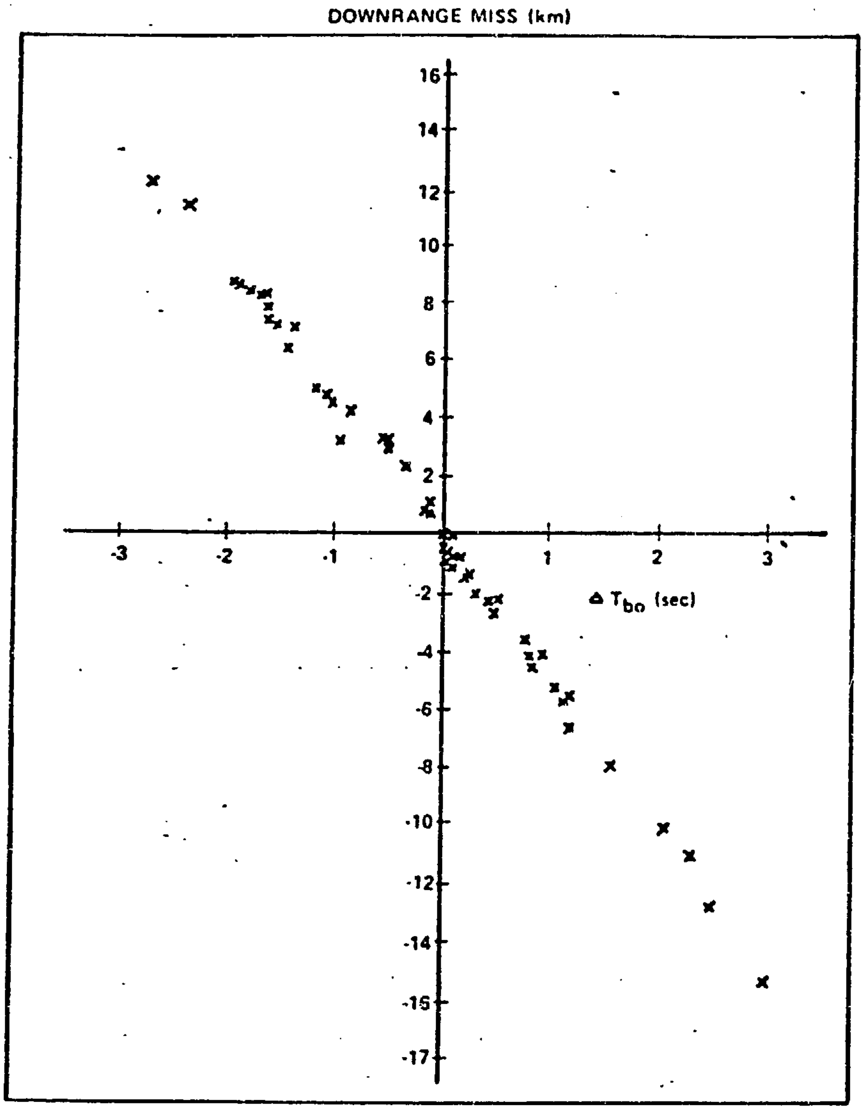
THREE-SIGMA PARAMETER UNCERTAINTIES

PARAMETER	3-SIGMA UNCERTAINTY
Specific Impulse	2.0%
Initial Weight	3.0%
Mass Flow Rate	2.5%
Thrust Misalignment	
In-Plane	0.1 deg
Cross-Plane	0.1 deg

Δ TABLE XVII

GUIDANCE ERRORS FOR SYSTEM WITH TIME COMPENSATION

IMPACT RANGE (km)	DR MISS (km)	CR MISS (km)	CEP (km)
480	1.41	0.30	1.00
960	2.78	0.67	2.00
1,870	4.72	1.46	3.46
2,950	6.22	2.59	4.70



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Δ Fig. 50 Downrange Miss vs ΔT_{bo} - 2,950-km Trajectory (U)

Δ TABLE XIX

GUIDANCE ERRORS FOR SYSTEM WITH TIME COMPENSATION (U)

IMPACT RANGE (km)	DR MISS (km)	CR MISS (km)	CEP (km)
480	0.22	0.44	0.39
960	0.33	0.93	0.72
1,870	0.41	1.98	1.41
2,950	0.63	3.28	2.30

APPENDIX II

PRINCIPAL ERROR AXIS CONCEPT

The principal error axis (PEA) concept is presented in this Appendix. Figure 52 portrays the launch centered inertial (LCI) computational coordinate system used where $\overline{Y}$ is along the geodetic vertical, $\overline{X}$ is directed downrange, and the $\overline{Z}$ axis forms a right-hand coordinate system. The vehicle thrust axis is in the $\overline{X}$, $\overline{Y}$ plane. The range from launch to impact is a function of the position and velocity vectors at thrust termination. For the purposes of this discussion, it will be assumed that the earth is nonrotating; therefore, a flight time dependency is not explicitly included. However, in the results to follow, flight time is implicitly included. Also, the crossrange effects will not be explicitly handled in the development of the range control.

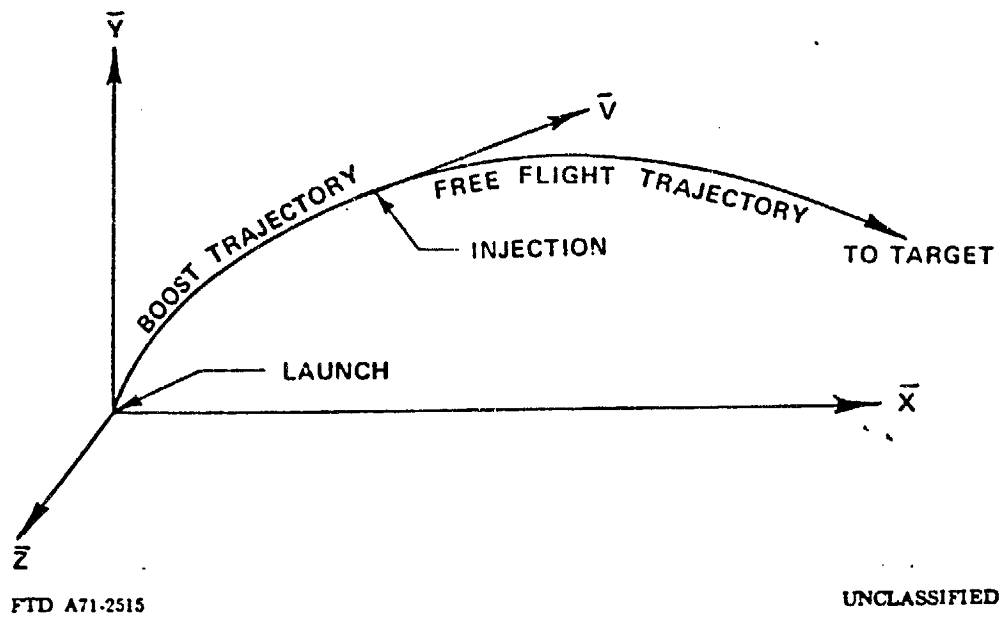

Fig. 52 Trajectory Coordination System (U)

(U) Taking the injection condition of a reference or nominal trajectory, a Taylor Series Expansion of the range equation can be developed as follows:

$$\Delta R = \frac{\partial R}{\partial X}\Delta X + \frac{\partial R}{\partial Y}\Delta Y + \frac{\partial R}{\partial \dot{X}}\Delta \dot{X} + \frac{\partial R}{\partial \dot{Y}}\Delta \dot{Y} + \ldots\ldots \quad (1)$$

where

$\Delta X = X - X_o$ = difference in X coordinate from the reference (nominal) condition X_o. Other Δ terms are defined analogously.

Only first order terms are retained. The principal position axis coordinate system is developed by defining it as a rotation about the $\overline{Z}$ axis of the LCI coordinated system as portrayed in Figure 53.

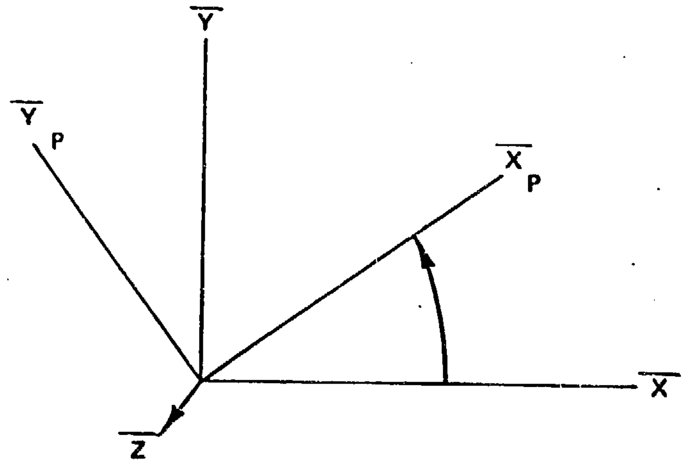

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Fig. 53 Principal Axis Coordinate System

The transformation of LCI errors into this coordinate system is

$$\begin{matrix}\Delta X_p \\ \Delta Y_p\end{matrix} = \begin{bmatrix}\cos\theta_p & \sin\theta_p \\ -\sin\theta_p & \cos\theta_p\end{bmatrix}\begin{matrix}\Delta X \\ \Delta Y\end{matrix}$$

The inverse transformation is

$$\begin{matrix}\Delta X \\ \Delta Y\end{matrix} = \begin{bmatrix}\cos\theta_p & -\sin\theta_p \\ \sin\theta_p & \cos\theta_p\end{bmatrix}\begin{matrix}\Delta X_p \\ \Delta Y_p\end{matrix} \tag{2}$$

Substituting equation (2) into position elements of equation (1) results in

$$\Delta R = \left(\frac{\partial R}{\partial X}\cos\theta_p + \frac{\partial R}{\partial Y}\sin\theta_p\right)\Delta X_p - \left(\frac{\partial R}{\partial X}\sin\theta_p - \frac{\partial R}{\partial Y}\cos\theta_p\right)\Delta Y_p \tag{3}$$

Defining the X_p axis as the principal position axis, the angle θ_p of this axis is determined by setting the component of ΔX_p to zero. Thus

$$\tan\theta_p = \frac{\partial R/\partial Y}{\partial R/\partial X}$$

and

$$\sin\theta_p = \frac{\partial R/\partial Y}{\sqrt{\left(\frac{\partial R}{\partial Y}\right)^2 + \left(\frac{\partial R}{\partial X}\right)^2}}, \quad \cos\theta_p = \frac{\partial R/\partial X}{\sqrt{\left(\frac{\partial R}{\partial Y}\right)^2 + \left(\frac{\partial R}{\partial X}\right)^2}} \tag{4}$$

Using these in equation (3) results in

$$\Delta R = \frac{\partial R}{\partial P}\Delta X_p$$

where

$$\frac{\partial R}{\partial P} = \sqrt{\left(\frac{\partial R}{\partial X}\right)^2 + \left(\frac{\partial R}{\partial Y}\right)^2} \text{ obtain by using (4) in (3).}$$

With the same approach, the principal velocity axis is developed.

(U) Summarizing the above, the necessary formulas are:

$$\theta_p = \tan^{-1} \frac{\partial R/\partial Y}{\partial R/\partial X}, \quad \theta_v = \tan^{-1} \frac{\partial R/\partial \dot{Y}}{\partial R/\partial \dot{X}}$$

$$\frac{\partial R}{\partial P} = \sqrt{\left(\frac{\partial R}{\partial X}\right)^2 + \left(\frac{\partial R}{\partial Y}\right)^2}, \quad \frac{\partial R}{\partial V} = \sqrt{\left(\frac{\partial R}{\partial \dot{X}}\right)^2 + \left(\frac{\partial R}{\partial \dot{Y}}\right)^2} \tag{5}$$

Using these in equation (1) results in the following equation for range error:

$$\Delta R = \frac{\partial R}{\partial P} \Delta P + \frac{\partial R}{\partial V} \Delta V \tag{6}$$

where the notation $\Delta P = \Delta X_p$ and $\Delta V = \Delta \dot{X}_v$ has been substituted.

APPENDIX III

CUTOFF LAW MECHANIZATIONS

Appendix II established the concept for mechanization of thrust termination. The desired condition for cutoff is when the range equation goes to zero. Setting equation (6), Appendix II, to zero results in

$$\Delta R = 0 = \frac{\partial R}{\partial P} \Delta P + \frac{\partial R}{\partial V} \Delta V \tag{1}$$

Then equation (1) can be solved for the required cutoff velocity as:

$$V_r = V_o - K_p (P - P_o)$$

where

$$K_p = \frac{\partial R / \partial P}{\partial R / \partial V} \tag{2}$$

Equation (2) forms the basis for range control by terminating thrust when the velocity along the principal axis equals the required velocity.

This concept for range control is simple, requiring only two scalar quantities (position and velocity along their respective principal axes) and equation (2) to be instrumented in order to satisfy the range control function. This concept relies on the control of the acceleration vector during powered flight to be of sufficient quality to satisfy conditions of linearity which was assumed in the development of equation (2).

To implement the concept, however, requires further approximations or more complex instrumentation than implied above. For example, the acceleration along a principal axis is

$$a_p = a_s \cos (\theta - \theta_p) - g_p \tag{3}$$

where

a_p = acceleration along the principal axis

a_s = total magnitude of sensed acceleration (sum of all nonconservative forces + mass)

$\theta - \theta_p$ = angle between a_s and principal axis

g_p = component of acceleration due to gravity along the principal axis

Integrating equation (3) results in

$$V_p = V_{sp} - V_{gp}$$

$$P_p = P_{sp} - P_{gp}$$

where

V_p = velocity along principal axis

V_{sp} = integral of sensed acceleration along principal axis

V_{gp} = integral of gravity along principal axis, with analogous definitions for the position (P) terms.

The sensed quantities can be measured by using inertial instruments. However, the gravity terms must be computed as a function of vehicle position which is not easily mechanized in principal axes.

If it is assumed that the flight trajectory is sufficiently close to the reference trajectory that the difference in the gravity integrals from nominal is small, then only the sensed quantities need be measured. The error which results from the assumption is termed "mechanization error." The magnitude of this contribution to the total mechanization error is a function of how well the boost system controls the flight trajectory. Therefore, to assess the magnitude requires a detailed analysis of major error sources contributing to trajectory deviations from the reference trajectory. Also, given one departure from the theoretical mechanization (i.e., neglecting explicit calculations of gravity), other than theoretical alignments or scaling constants (K_p) of equation (2) could result in smaller mechanization errors.

The position terms in equation (2) represent differences of large numbers which are hard to mechanize accurately without resorting to digital techniques. An alternate approach is to mechanize the difference between the reference (programmed) sensed velocity along the position axis and that measured by an integrating accelerometer. This difference is small (consistent with previous assumptions) and, therefore, can be integrated quite accurately without resorting to digital integration. With this consideration, equation (2) is modified to read

$$V_{rs} = V_{os} + K_p \Delta P \tag{4}$$

where

$$\Delta P = \int_0^t (V_p - V_s)\, dt$$

V_{rs} = required sensed velocity along velocity axis θ_v

V_{os} = reference sensed velocity along that axis

V_p = programmed sensed velocity as a function of time along position axis θ_p

V_s = measured sensed velocity along that axis.

APPENDIX V

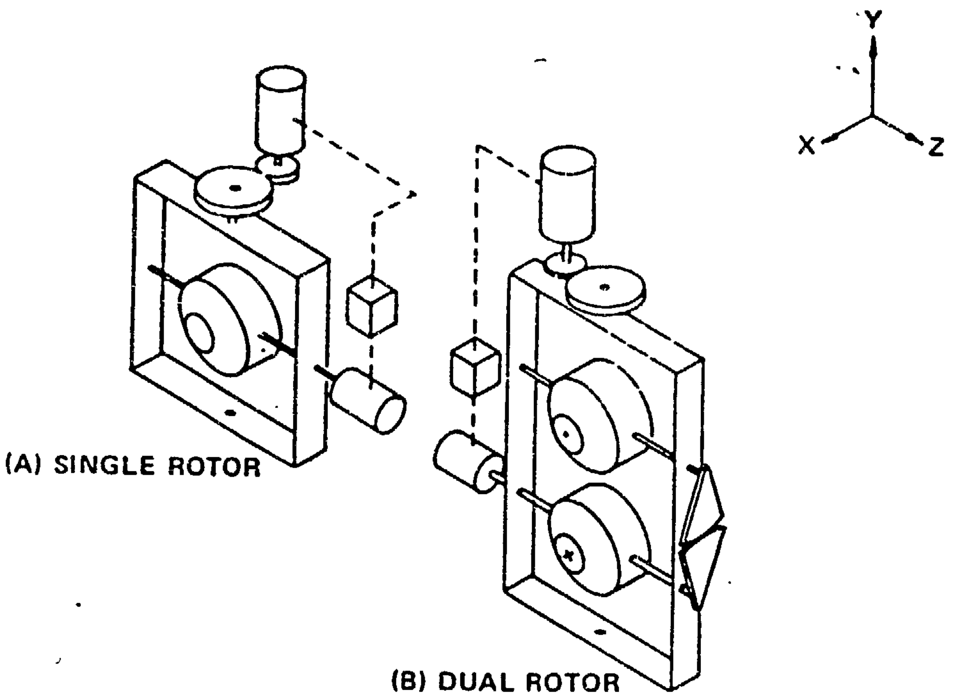

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Fig. 54 Soviet Directional Gyroscopes

APPENDIX VI

SOVIET NAVIGATIONAL AIDS (U)

(U) Reference to an inertial navigation system will occur throughout the following discussion. It is necessary at this point to provide a statement concerning the meaning of inertial navigation systems in order to clarify the context of the discussion. All inertial navigation systems are dead reckoning systems. A dead reckoning system depends upon continuous operation and periodic fixes inserted as updates to correct its inherent errors and reset the system. The periodic fixes must be supplied by systems which are independent of measureable drift, platform maneuvers, and other disturbing factors. No one fixing system meets all these criteria. Therefore, any inertial system must have several different fixing methods in order to function accurately.

LORAN positions are determined from the difference in the time of arrival of pulses transmitted simultaneously from a master and two slave stations. Greatest accuracy is achieved when the observer is on a line which bisects the angle formed by the two base lines from the master to the slave stations and decreases as the distance from the stations increases.

No special receiving equipment is required, but automated receivers are on the open market which would remove the basic operator errors involved in counting dots and dashes.

BIBLIOGRAPHY

1. Appazov, F. R., Lavrov, F. S., and Mishin, V. P., "Ballistics of Long-Range Rockets ," Science Publishing House, Moscow, USSR, April 1966.

2. DIA Study No. ST-CS-05-294-74, "Space Vehicle Guidance and Control - ECC ,"

3. FTD-MT-65-96, Rivkin, S. S., "Theory of Gyroscopic Devices, Part II ," Izdatel'stvo Sudostroyeniye, Leningrad, USSR, 1964.

4. Slomanskiy, G. A. "On Determination of the Drift of Float Integrating Gyroscopes Without the Use of a Dynamic Stand ," Izy, Vuz, Priborostrogeniye, Vol. No. 3, 1963.

5. Yagodkin, V. V., Khlebnikov, G. A., "Gyroscopic Devices of Ballistic Missiles " Voyennoye Izdvo Ministerstva Oborony, SSSR, Moscow, 1967.

6. Andreyev, U. D., "Theory of Inertial Navigation Correcting Systems " Moscow, USSR, 1967.

7. Ishlinskiy, A. Yu., "Inertial Guidance of Ballistic Missiles ," Published 1968, Translated 1971, FTD-MT-24-291-70.