MIDCOURSE GUIDANCE USING RADIO TECHNIQUES

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ABSTRACT

Lunar midcourse guidance using Earth-based radio tracking and computation is discussed. Primary emphasis is on engineering factors, including requirements placed on spacecraft, tracking stations, and computing facilities. Finally, performance is described, and maneuver size, number of maneuvers, tracking accuracy, and guidance accuracy are treated.

INTRODUCTION

Historical Background

In 1958, when intensive work at the Jet Propulsion Laboratory was begun on lunar and interplanetary guidance, early thoughts favored self-contained systems with heavy reliance on inertial and optical equipment. It quickly became apparent, however, that midcourse guidance utilizing Earth-based radio tracking, orbit determination, and command combined with self-contained terminal systems operating in the immediate vicinity of the target was adequate for most lunar and planetary missions. For unmanned missions, in which a radio link is essential for the transmission of data from the spacecraft to the Earth, the Earth-based system offered the attractive feature that both observations and computations were made on the ground rather than in the spacecraft. This technique forms the basis for the guidance systems of the Ranger, Mariner, and Surveyor spacecraft.

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Scope of Paper

This paper is concerned primarily with engineering experience and considerations rather than with mathematical aspects which have been presented fully elsewhere.[3] The paper attempts to relate the radio command midcourse system to the total mission, showing the constraints which it places on the spacecraft and on the ground system.

SYSTEM DESCRIPTION

Overall System

The radio command midcourse guidance system can be described in terms of a) spacecraft, b) tracking stations, and c) central computing facility.

Spacecraft

The spacecraft will contain, of course, a basic rf system, including receivers, transmitters, antennas, etc. There must be a transponder as a part of the doppler velocity or range rate system, and possibly a ranging transponder as a part of a range system. Since the tracking data which are obtained immediately after injection are of great power in the determination of the orbit, and since a typical spacecraft may not erect to its normal attitude immediately after injection, an omni-directional antenna system may have to be provided.

The spacecraft will contain an attitude reference and control system, very likely using visual references, such as the Sun, Earth, and stars, for attitude reference. For a typical lunar spacecraft on an impacting or direct landing mission, fuel may be carried for a maneuver as great as 50 m/sec (which corresponds to about 2% of the spacecraft weight in fuel); this maneuver may take several minutes to perform, and it may be required that the maneuver be executed in any direction in space. Thus, the spacecraft must be capable of orienting the midcourse rocket motor in any direction. Since the midcourse maneuver is a vernier type, in that, if injection were accomplished with no error, no midcourse maneuver would be required, it follows that errors committed in the execution of the midcourse maneuver are essentially second order. Thus, a simple autopilot in a spacecraft together with one body-fixed velocimeter are generally adequate. Accuracies of 1/2% in the magnitude of the midcourse maneuver velocity and 1/2° in its direction are typical.

[3] Noton, A.R.M., "Guidance of space vehicles by radio measurements and command," J. Brit. Interplanet. Soc. 18, 132-138 (July 1961).

Tracking Stations

A typical tracking station will contain, of course, a basic complement of rf gear, including transmitters, receivers and coders, antennas, antenna drive, and readout equipment, etc. Also, accurate frequency references and timing equipment will be needed for precision Doppler and range.

Acquisition aids may be required for errant spacecraft; such aids may be either rf in the form of beam broadening devices, or trajectory-related in the form of search patterns. It should be noted that the uncertainty in the location of the spacecraft, insofar as acquisition is concerned, is significant only immediately after injection. However, as stated previously, the data obtained immediately after injection are of particular significance.

In addition, the tracking station must contain terminal equipment for the communications link between the station and the Central Computing Facility. For unmanned spacecraft the stations of the DSIF (Deep Space Instrumentation Facility) are used.

Central Computing Facility

As pertinent to midcourse guidance, the Central Computing Facility processes tracking data, determines the orbit, and computes the midcourse command. Other important functions of the Central Computing Facility are processing of engineering and scientific data, determination of spacecraft performance, and overall command of the spacecraft and the tracking stations.

The central element in the Central Computing Facility is a large, high speed digital computer such as the IBM 7090. Other elements will include communications terminal equipment, special purpose computers for data processing and sorting, printers, plotters, special-purpose display equipment, and internal command and communications equipment to permit the flight operation to be carried out in a timely and orderly fashion.

The use of a large digital computer as an element of a control loop in a guidance system, involves new and significant problems, such as (a) computing the orbit and the command in a timely and reliable manner, (b) providing sufficient flexibility in the computer program so as to be able to accommodate nonstandard conditions, and (c) providing adequate and timely displays of information to enable command personnel to tell what is happening.

SYSTEM OPERATION

Pre-Injection

To an observer on Earth, the moon is a moving target, and hence the pre-injection trajectory will be a function of the launch time. For a typical lunar mission a "launch period" of approximately one week will be available during which launchings may occur; if this week is missed, the launchings will have to be delayed by one lunar month. Also, for a typical lunar mission, a "launch window" of approximately 2 hr per day will be available for launching. Prior to launch, data will have to be provided to tracking stations of the missile range as well as the DSIF in order to aid in their acquisition of the launch vehicle and the spacecraft.

Post-Injection

Following injection, data taken by tracking stations of the missile range will be processed quickly and transmitted to the appropriate DSIF station. As stated previously, the interval immediately following injection is particularly critical for three reasons: (a) the uncertainty in the angular position of the spacecraft is greatest at this time; (b) the angular rate of the spacecraft is greatest at this time; and (c) the tracking data taken during this interval has the greatest power in determining the orbit.

Following injection the spacecraft rises rapidly above the Earth's surface, such that, after about 1 hr, it usually will be visible to both the South Africa and Woomera stations of the DSIF. During this time the tracking stations will take angular information and doppler velocity and transmit these data back to the Central Computing Facility. At the Central Computing Facility the tracking data will first be edited, which involves sorting, editing, and coding of the tracking points into a form suitable for the orbit determination process, and also the rejection of obviously bad data points. Next, the orbit is determined, using a least-squares estimation procedure for the orbital elements. Finally, the midcourse maneuver is computed based on the orbital elements determined from the orbit determination procedure. In this computation one determines such things as (a) the impact point on the moon (or lack of impact if this be the case) if no maneuver is executed; (b) the locus of impact points at the moon which the midcourse fuel in the spacecraft will enable the spacecraft to attain,

(c) the maneuver required for the spacecraft to reach the desired target point without adjustment of the time of flight; (d) the maneuver required for the spacecraft to reach the desired impact point while adjusting the time of flight to a desired time; (e) both of the preceding maneuvers in terms of spacecraft coordinates; and (f) the extent to which any spacecraft constraints may be violated by the maneuvers which are desired. Finally, the maneuver is executed.

Post Maneuver

Following the maneuver, additional tracking is required in order to get an accurate determination of terminal conditions.

SYSTEM PERFORMANCE

Error Sources

In assessing the accuracy with which a given mission can be performed, it is convenient to divide errors into three categories: (a) tracking errors, (b) physical constant errors, and (c) maneuver execution errors.

Tracking errors include such things as receiver noise, antenna jitter, etc. It should be noted that two-way doppler velocity, using precise frequency references, is the data type with the overwhelming power on the orbit determination process. Angular information is of secondary importance and is useful only during the very early portion of a given flight.

Physical constant errors include geodetic errors such as uncertainties in station location, uncertainties in the mass and radius of Earth, etc. Also important are uncertainties in the velocity of propagation and uncertainties in the mass and ephemeris of the moon. Finally, execution errors are errors committed in the execution of the midcourse maneuver. In the current state of the art, errors from the above three sources are about comparable.

Mission Profile

In discussing the accuracy of radio command midcourse guidance, it is necessary to take into account the nature of the mission which is being performed. For example, the trajectory flown, including launch azimuth, flight time, and latitude and longitude of the injection point, strongly affects the geometry of tracking particularly in the early and important part of the trajectory and hence will influence strongly the guidance accuracy. In addition, since the maneuver corrupts the trajectory because of execution errors, one may exchange accuracy at the target for knowledge of conditions at the target. To be more specific, if a maneuver is performed relatively early, the orbit at that time will be less perfectly known. Hence the dispersion at the target will be somewhat greater than if the maneuver were performed later. However, a maneuver performed early permits more tracking following the maneuver, and hence conditions in the vicinity of the target will be known with greater accuracy.

Typical Performance

Maneuver magnitude

Some quantitative results of the guidance analysis for an unmanned lunar impact mission such as Ranger are now described. The standard trajectory has a flight time of 66 hr with a near vertical impact. A single midcourse correction is applied, and it is assumed to be an impulsive thrust. In computing the magnitude of the impulse, it is assumed that injection errors are the only perturbation and that the actual orbit can be determined perfectly.

For the lunar impact mission, it is desired to correct surface miss components. Since it is possible to correct three terminal errors with an impulsive maneuver, there is one degree of freedom which may be used to minimize the maneuver magnitude (option 1) or to correct another terminal variable (option 2). For option 2 it is convenient to correct impact time because this simplifies Earth-based operations. Of course, for some lunar missions it may be desirable to correct some other terminal quantity such as impact speed.

Fig. 1 shows the midcourse velocity increment v (using option 1) required to correct for a particular injection error as a function of the application time of the maneuver. Velocity increments are shown for 1σ injection errors in altitude (Δr), latitude ($\Delta \phi$), longitude ($\Delta \Theta$), speed (Δv), and path angle ($\Delta \gamma$, $\Delta \sigma$) for a typical injection guidance system. For large t, the maneuver magnitude varies approximately inversely as the time of flight remaining, but for small t's the variation is more complicated.

In order to compute the rms maneuver it is necessary to take into account the covariance matrix[4] of injection errors for a

[4]Loc. cit.

typical injection guidance system. For the injection guidance system considered here, the uncorrected rms miss at the moon is 1500 km. Fig. 2 shows the rms maneuver required for options 1 and 2 as a function of application time. Note that Δr and Δv are correlated in such a way that they compensate for each other when one computes the average maneuver.

The choice of the optimum maneuver time depends on, among other things, the weight of fuel which must be carried. Let $\overline{v^2}$ be the mean-squared magnitude of the midcourse velocity increment . It can be shown[5] that with a correction capability of $2.58\,(\overline{v^2})^{1/2}$, one can correct for at least 99% of all injection errors. Thus, the percentage of spacecraft weight required for fuel is $2.58\,(\overline{v^2})^{1/2}/gI_{sp}$. Assuming $I_{sp} = 230$ sec and an application time of 16 hr, 1.7% of the spacecraft weight is required for fuel.

For more exacting guidance requirements, it is necessary to use several midcourse corrections. With multiple corrections it is possible to correct for more terminal variables. Also it is possible to correct repeatedly for the same variables and thus to refine the guidance. In the latter scheme the orbit would be redetermined after each correction in order to determine the execution errors.

Accuracy of orbit determinations

The most obvious source of error in orbit determination is the noise in the radio observations. In order to obtain representative numbers, assume the same trajectory as before and two tracking stations, one at Goldstone Lake in California (Pioneer Station) and the other at Johannesburg in South Africa (Mobile Tracking Station or MTS). The latter assumption is quite conservative since the spacecraft normally would be tracked by a number of other stations. Also assume that the stations take a triad of observations (2 angles and range rate) at 1-min intervals whenever the spacecraft is above their horizon. Sampling at a greater rate does not markedly improve the accuracy of the orbit determination.

Table 1 shows the assumed noise characteristics of each station. In order to simplify the computing, a very simple noise model is used. Each noise source is assumed to be

[5] Loc. cit.

independent and is characterized by an rms noise amplitude σ and a correlation time T. A noise record from a particular noise source consists of a sequence of steps of random amplitude every T sec. Recent results indicate that the value shown in Table 1 for σ of the range rate data may be conservative by a factor of two.

By using statistical estimation theory, one may obtain the expected error in estimating the terminal conditions. Fig. 3 shows the semimajor axis λ_1 of the dispersion ellipse at the moon due to observation errors. First consider Curve A, which shows the case in which there is no midcourse maneuver. The MTS acquires the spacecraft a few minutes after injection and gradually decreases λ_1 until the spacecraft exceeds the assumed maximum range of the MTS (80,000 km) at 6 hr. The Pioneer Station acquires when the spacecraft rises above its horizon at 11 hr and begins to decrease λ_1 again. Impact occurs in the middle of the third pass over this station. In selecting the optimum maneuver time, one must take into account not only λ_1 at the maneuver time but also the accuracy with which the orbit can be redetermined for a terminal maneuver.

Curves B and C show λ_1 when there is a maneuver at 16 hr. There is an abrupt increase in λ_1 at this time because of the uncertainty in the execution of the maneuver. Curve B shows the case in which the orbit is redetermined assuming no prior tracking data. Curve C shows the case in which the orbit is redetermined taking into account the prior tracking data and the expected midcourse execution errors. Fig. 4 shows the rms error in predicting the impact time $(\overline{t_f^2})^{1/2}$. If range data are used in addition to the types in Table 1, λ_1 and $(\overline{t_f^2})^{1/2}$ decrease much more rapidly. Alternatively, with the use of range data, fewer samples are required to achieve a given accuracy.

Another source of orbit determination error is provided by errors in the mathematical model. This includes

(a) uncertainties in dynamical constants (e.g., the gravitational constant of Earth, mass of the moon)
(b) uncertainties in observational constants (e.g., station location, speed of light)
(c) ignored effects (e.g., higher harmonics in Earth's gravitational potential)

(d) computer errors (i.e., roundoff and truncation)

Note that some of these errors will be present regardless of the type of observations used to determine the orbit. Uncertainties in dynamical constants cause errors both in estimating the orbit from the tracking data and in computing the required correction. To a certain extent, these errors compensate for each other. The exact effect depends on the data-taking pattern during the flight. For the pattern assumed here, an uncertainty of one part in 10^5 in the gravitational constant of Earth will cause an uncertainty of 20 km at the moon. The values of the dynamical constants are gradually being improved by tracking satellites and space probes.

An uncertainty in station location of 100 m will cause an error of approximately 7 km at the moon. An uncertainty in station altitude of 50 m will cause an uncertainty of 10 km at the moon. Uncertainties in station coordinates are also being reduced by satellite tracking. An uncertainty in the vacuum speed of light of one part in 10^6 can cause an error of 10 km at the moon. In some cases it is possible to treat the uncertainty in a physical constant as an additional parameter to be estimated in the orbit determination process. It is assumed that type (c) and (d) errors are negligible. Another source of error is unpredictable disturbances (e.g., corpuscular pressure from a solar storm) which occur after the maneuver, but this is believed to be negligible for a lunar impact mission.

Execution errors

The statistical properties of the execution errors are difficult to compute because the error is a complicated function of the instrument errors and the desired maneuver, both of which are random variables. For exact answers, it is necessary to use Monte Carlo methods. However, reasonably accurate answers can be obtained by using a simplified model. The velocity error is divided into three orthogonal components: w_3 in the direction of the desired maneuver and w_1 and w_2 in the plane normal to w_3. The velocity error in each direction is composed of two parts: one which is proportional to the maneuver magnitude v, and the other which is fixed. Thus, the axial velocity error is $w_3 = \gamma v + \theta$ where γ is proportional to accelerometer scale factor and null offset errors and θ is a fixed velocity error due to accelerometer resolution and thrust tail-off errors. Similarly in the lateral directions, $w_1 = \alpha v + \zeta$ and $w_2 = \beta v + \eta$, where α and β are pointing errors caused by gyro and attitude sensor errors and ζ and η are fixed velocity errors which occur as a result of the midcourse autopilot correcting for thrust misalignments.

It is now assumed that all six component errors are independent of each other and of v. Analysis of a typical midcourse guidance system (Ranger) yields:

$$(\overline{\alpha^2})^{1/2} = (\overline{\beta^2})^{1/2} = 0.01 \text{ rad}, \; (\overline{\gamma^2})^{1/2} = 0.004 ,$$

$$(\overline{\zeta^2})^{1/2} = (\overline{\eta^2})^{1/2} = 0.06 \text{ m/sec, and } (\overline{\theta^2})^{1/2} = 0.03 \text{ m/sec.}$$

A spherical distribution for the components of v is now assumed and the rms values of w_1, w_2, and w_3 are computed. On transforming the velocity errors to the target, we find that the rms miss is about 48 km if the maneuver is executed at 16 hr. For the component errors listed in the foregoing, practically all of the miss is due to the pointing errors α and β. The miss due to proportional errors is relatively independent of application time whereas the miss due to fixed errors decreases with application time.

If option (1) is used, flight time variations occur not only because of injection errors but as a result of the midcourse maneuver itself. The rms variation in flight time due to this effect is 1500 sec. If option (2) is used, the rms flight time error is 23 sec due to execution errors. The error in other terminal coordinates is also of interest. If option (1) is used, the rms error in impact speed is 5.3 m/sec and if option (2) is used the error is 1.0 m/sec. The results are summarized in Table 2. Note that this table applies to a Ranger-type spacecraft; i.e., it assumes a single impulse maneuver and current values of observation, physical constant and execution errors. More advanced spacecraft using multiple maneuvers and improved observation and physical constant errors will achieve considerably better accuracies.

CONCLUSIONS

Analytical results indicate that radio command midcourse guidance is suitable for insuring impact on a small preselected area of the moon. Ranger 3 demonstrated the feasibility of the three major elements of the midcourse guidance system: (a) Earth-based tracking and orbit determination, (b) attitude control (in both the cruise mode and maneuver mode) and (c) the execution of the commanded maneuver. Earth-based radio command midcourse guidance is practical for midcourse guidance.

Table 1 Estimated noise characteristics

Station	Data type	Noise source	σ^{a}	T, sec
So. Calif.	range rate, m/sec	frequency stability and roundoff error on cycle count	0.06	60
	declination, deg	servo jitter	0.03	<10
		refraction	0.01	30
		antenna structure deflections	0.01	18,000
	hour angle, deg	same as declination		
So. Africa	range rate, m/sec	same as range rate for So. Calif.		
	elevation deg	servo jitter	0.1	<10
		antenna structure deflections	0.1	3,600
		refraction	0.01	30
	azimuth angle, deg	same as elevation angle		

[a]Units for σ are given in the column, Data Type.

Table 2 Summary of midcourse guidance characteristics (single impulse)

RMS maneuver	15 m/sec
Percentage of spacecraft weight for fuel	1.7
RMS execution errors	48 km
RMS observation errors	20 km
RMS physical constant errors	25 km
Total rms miss	58 km

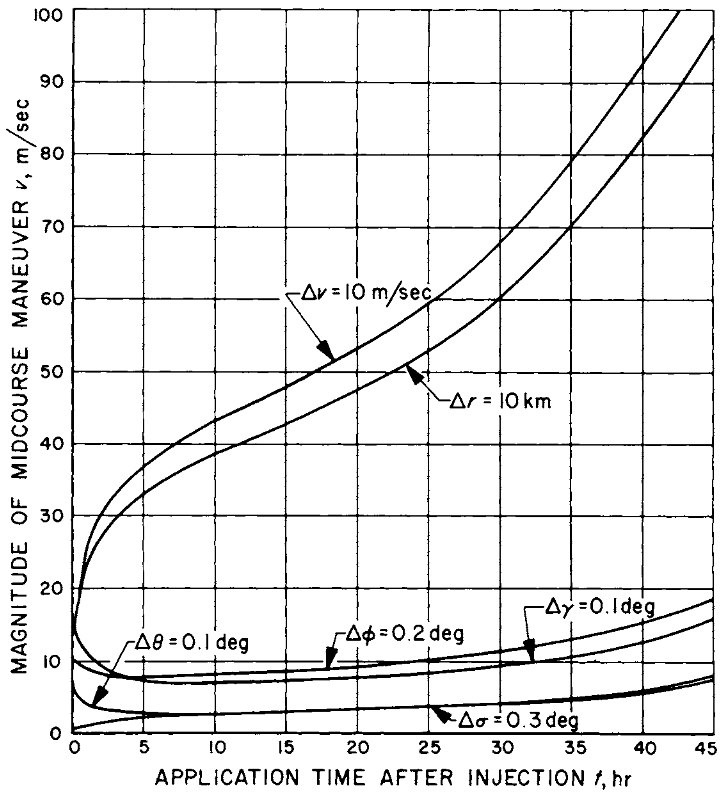

Fig. 1 Midcourse maneuver to correct injection errors

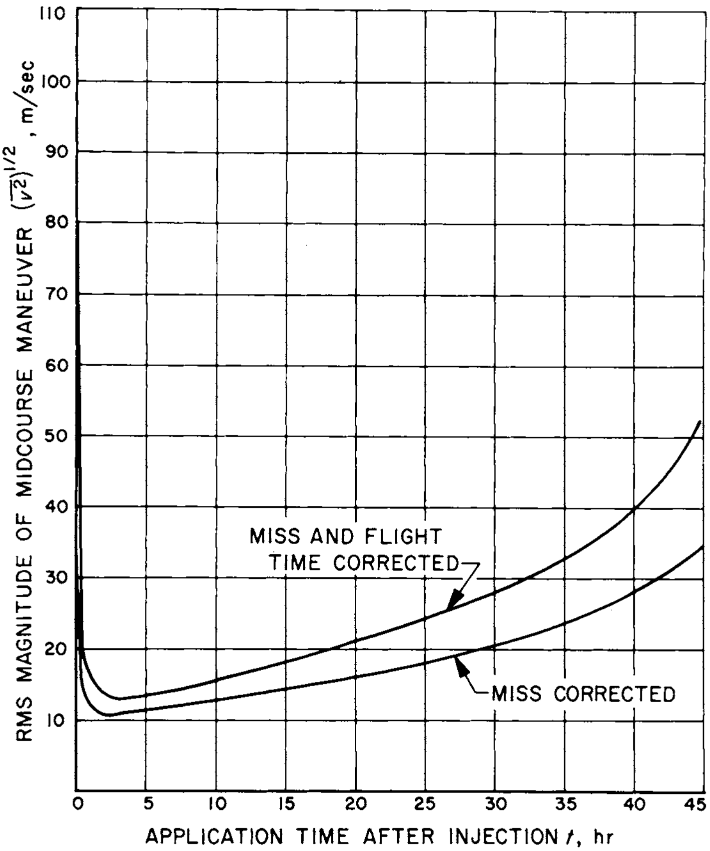

Fig. 2 Midcourse maneuver for a typical injection guidance system

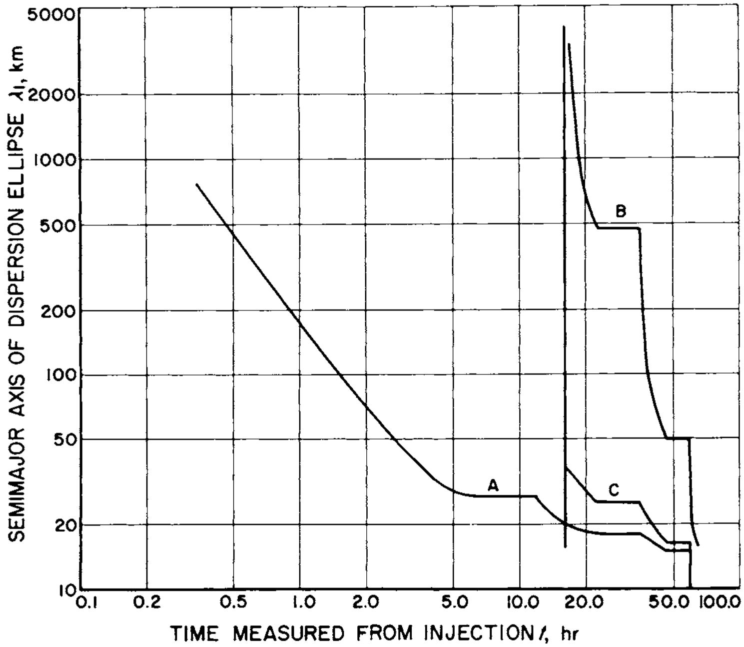

Fig. 3 Accuracy of predicting miss as a function of tracking time

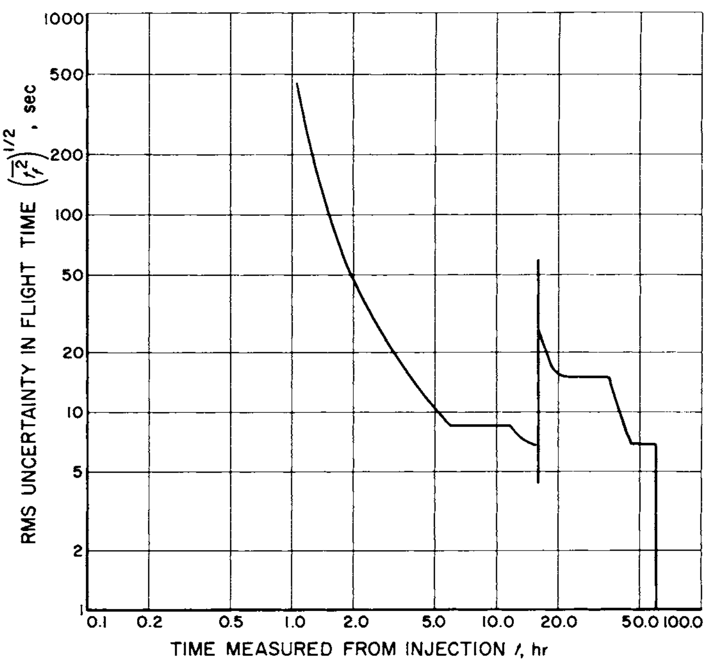

Fig. 4 Accuracy of predicting impact time as a function of tracking time