NAVIGATION, GUIDANCE, AND CONTROL PROBLEMS OF SPACECRAFT

David G. Hoag

Instrumentation Laboratory, Massachusetts Institute of Technology, Cambridge, Mass.

Introduction

As man in his never-ending curiosity has sent his instruments and himself on voyages into space he has had to recognize many difficulties in this new foreign and hostile environment. The impressive thing is the great number of technologies which had to be developed or expanded in concert to make this giant step up out of the earth's sensible atmosphere into space. This paper concerns one facet of the problem: that of measuring and controlling the spacecraft position, velocity, and orientation in support of mission objectives. But even this limitation of subject leaves a tremendous scope to cover in any sort of detail. We must look briefly, then, and admittedly perhaps superficially, at techniques and problems of particular interest in spacecraft navigation, guidance, and control.

To understand how this topic relates to actual spacecraft tasks, consider a fragment of a hypothetical space mission illustrated in FIGURE 1 which shows the midcourse, approach, and orbital phases of planetary exploration. In the sequence of phases we see operations concerning measurement and control of vehicle orientation and trajectory. The parameters of concern are the time history of the three degrees of freedom describing orientation and the time history of the three degrees of freedom describing position. This topic, then, is the rotational and translational management of spacecraft.

Missions like the hypothetical one illustrated in FIGURE 1 are operated in phases which alternate with a period of powered thrusting accelerated* flight followed by free-fall coasting. This is a consequence of the character of available propulsion typical in the chemical rocket engine. The nature of the rotational and translational management problems differ markedly between the free-fall and thrusting accelerated conditions. With paired combinations of "rotational" or "translational" and "free-fall" or "accelerated" we arrive at four separate aspects of the theme (see FIGURE 2). This figure shows these four with the assignment of names which have common usage: navigation, attitude control, thrust vector control, and guidance.

The fundamentals of the problem and the foundation of the solutions for each of the four aspects will be examined. Following this, several special problems of the subject will be analyzed.

In order to limit discussion to manageable proportions, this paper will be concerned only with spacecraft operation in the vacuum of space within the solar system. This excludes both the serious important considerations of the boost out of and entry into planetary atmosphere and the perhaps frivolous realm of flight among the stars.

NAVIGATION

(Translational Measurement in Free-Fall Coasting Flight)

Navigation is the process of measurement and computation to determine the existing present position and probable future position of a vehicle. As defined herein it is concerned only with the translational aspects of motion — i.e. position and velocity — and is particularly applicable to the free-fall coasting conditions of

*Accelerated in the context of this paper means acceleration with respect to the free-fall motion. It excludes, therefore, acceleration due to gravity forces.

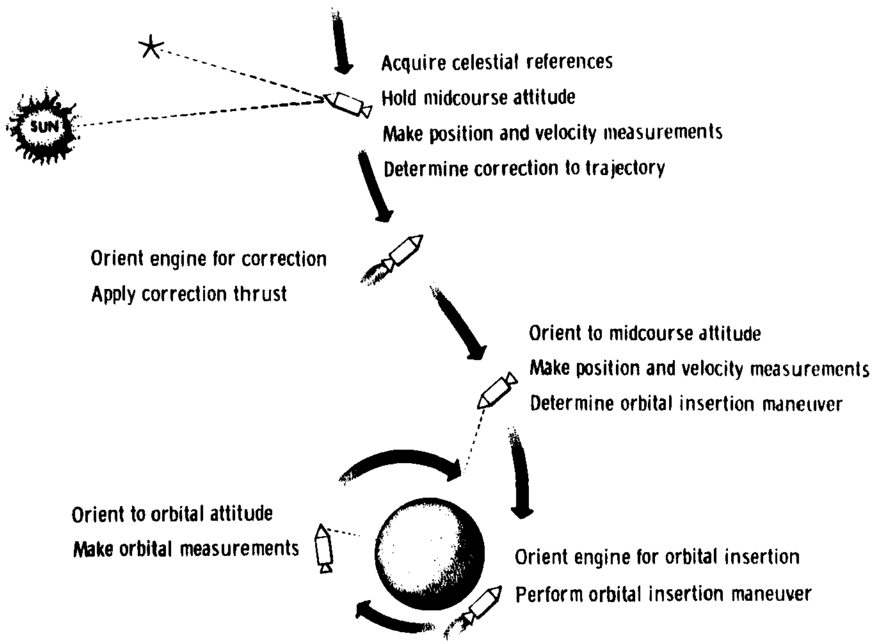


FIGURE 1. Phases of hypothetical mission.

spacecraft. It includes those processes necessary to determine needed trajectory corrections as well as to compute necessary initial conditions of major powered maneuvers.

In nonthrusting flight the forces on the spacecraft which determine its motion are dominated by the Newtonian gravitation attraction of the near bodies — the earth, moon, sun, and planets. Generally the vehicle is influenced primarily by only one body and follows nearly the classical Keplerian conic path. The effects of forces other than that of the point mass central body can usually be treated as deviations or perturbations to the simpler motion. A nonexhaustive listing of typical perturbing effects includes: (1) Mass distribution within central body, e.g. oblateness of earth, triaxiality of moon; (2) Attraction of more remote bodies; (3) Atmospheric drag; (4) Solar radiation pressure; (5) Meteoroid impact; and (6) Magnetic and electric field interactions with spacecraft.

In a given situation it is usually possible to ignore all but a few of the perturbing effects and predict the future trajectory of the vehicle with satisfactory accuracy many hours to many days into the future using knowledge of present position and velocity. However, for a given accuracy the prediction finally deteriorates due to ignored effects and the limitations on accuracy of the initial conditions and the extrapolation model.

Because of the relatively predictable nature of spacecraft trajectories in free coasting flight, continuous measurment of position and velocity is unnecessary. Measurements are needed periodically to correct the slow deviation of the actual spacecraft from the predicted path and to provide data to update the prediction.

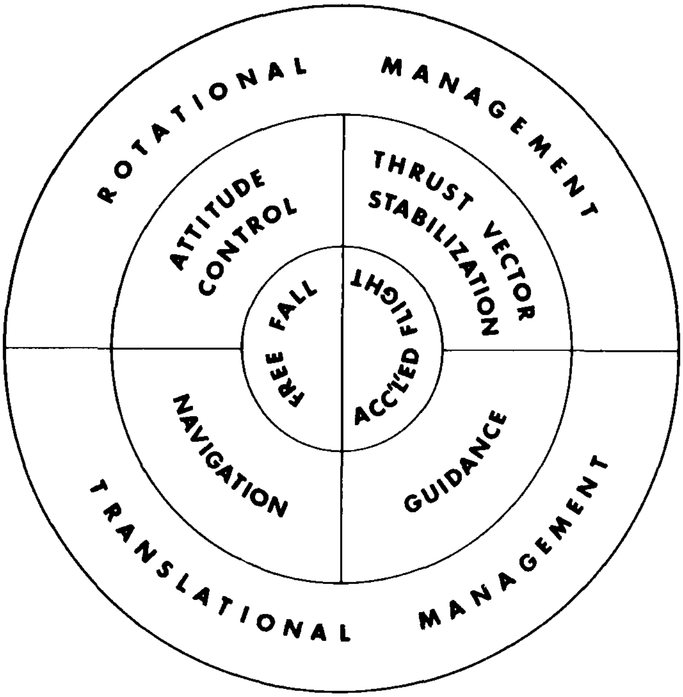


FIGURE 2. Spacecraft rotational and translational environment.

Practical navigation measurements in free coasting flight all utilize electromagnetic radiation at appropriate wavelengths to sense spacial relationships among the spacecraft and the near bodies of the solar system. These measurements can be categorized into two types: First, those made groundbased by remote tracking of the spacecraft from suitable stations on the earth, and second, those made from on-board using sensing devices on the craft itself. Only the first of these has yet been applied; all U.S. spacecraft and as far as we know all Soviet vehicles have been navigated using earth-based tracking measurements only.

Ground-based tracking for navigation usually uses radar frequencies with the cooperative use of transmitting beacons or transponders on the spacecraft being tracked. Optical wavelengths have seen use but suffer from the problem of blanketing cloud cover.

Radio tracking for navigation is founded upon: (1) The fixed and well-known speed of light in space; (2) the use of highly accurate time bases and stable frequency sources, and (3) the ingenuity and accuracy with which precise phase measurement can be made between two signals in the face of interfering noise.

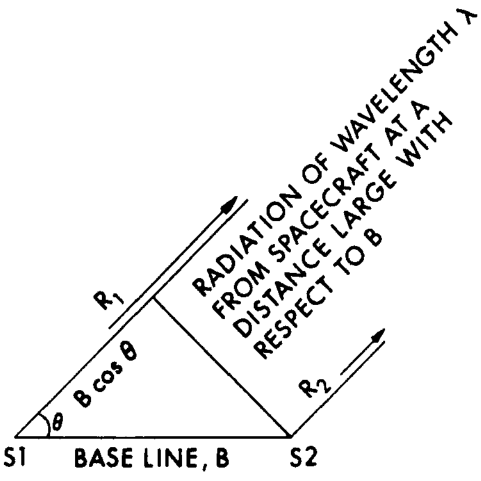
For measurement of range, the basic principle used is that the distance to and from the spacecraft is proportional to the phase lag of the waveform as received with respect to the transmitted waveform. Range resolution then is that fraction of a wavelength with which the phase can be measured. A 100 mc carrier, for instance, has a wavelength of 3 meters, and range resolution well inside this size is straightforward. Lower frequency modulation tones with longer wavelengths are used to resolve the ambiguities and thereby determine the more significant figures of the number representing measured range. For spacecraft this technique depends upon a transponder in the spacecraft which will amplify the received transmission and reradiate with controlled phaseshift an appropriate signal to the ground station.

Although range tracking, as defined, has almost micrometer resolution capability, several limitations on the total overall accuracy exist. The most apparent, of course, is our knowledge of the exact speed of light. This is currently known to about 1 part in 10^6. Without calibration correction as discussed below, this means a range error of 150 kilometers in a spacecraft distance of one astronomical unit.

Range rate information is measured by the rate of change of phase shift of the received signal or more familiarly by the equivalent Doppler frequency shift.

For spacecraft position across the line of sight or directional data, two direct radio frequency measurement techniques are possible. The first, a scanning or lobing process similar to radar tracking with directional antennas, has many limitations. Particularly, the problem of relating the actual direction of received radiation with respect to a suitable coordinate frame is constrained by limits in structual accuracy and stability of the gimbaled high gain antennas and presents a challenge to design and measurement.

The second direction measurement technique is a sort of inverted triangulation using multiple receivers on accurately known baselines. If this baseline array is suitably short, the received signals can be simultaneously processed in the same equipment in an interferometric measure of the differences in range of the spacecraft from the various receivers (see FIGURE 3). The Minitrack network which first tracked Sputnik in 1957 falls in this class. It is a technique which still offers



$$B \cos\theta = R_1 - R_2 \qquad \Delta\phi = \text{PHASE DIFFERENCE (S1, S2)}$$

$$\cos\theta = \frac{R_1 - R_2}{B} \qquad R_1 - R_2 = \frac{\lambda}{2\pi}\Delta\phi$$

$$\cos\theta = \frac{\lambda}{B}\,\frac{\Delta\phi}{2\pi}$$

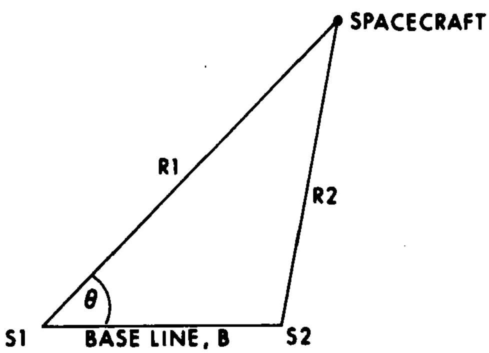
FIGURE 3. Short baseline interferometric direction measurement.

many advantages, particularly the fact that the spacecraft need carry only a radio beacon which does not need to be interrogated from the ground.

For these short baselength systems the differences in phases of the various received waveforms can be measured with extreme precision. A 3-meter signal wavelength (100 mc) can be resolved by phase measurement to 3 millimeters utilizing techniques such as heterodyning to a lower frequency and precision timing. On a 150-meter baselength this corresponds to 20 microradians (4 seconds of arc) of angular resolution of spacecraft directions which lie near normal to the baseline. At a one astronomical unit spacecraft distance this is resolution of spacecraft position across the line of sight of 3000 kilometers.

For the usual horizontal array of receivers, it is seen that best directional accuracy is obtained for conditions with the spacecraft direction near perpendicular to the baseline. As the vehicle gets near in line with the baseline the angular resolution degenerates inversely as the sine of the angle from the baseline. Moreover, near the horizon, earth atmospheric refraction uncertainty degenerates the total indicated direction accuracy.

For greater accuracy in direction measurement, the baseline can be increased. However, several problems interfere with proportional accuracy improvement of longer baselines in comparison with the short baseline interferometric systems. First, wide separation of the receiving stations prevents accurate, simultaneous, direct phase comparison of the received signals. Also the direction is not a direct function of the range difference anymore (see FIGURE 4). So rather than using range differences obtained directly by phase comparison, the total range values



$$R_2^2 = R_1^2 + B^2 - 2R_1 B \cos\theta$$

$$\cos\theta = \frac{R_1^2 - R_2^2 + B^2}{2R_1 B} = \frac{R_1^2 - R_2^2}{2R_1 B} + \frac{B}{2R_1}$$

$$\cos\theta = \frac{(R_1 - R_2)}{B}\left(1 - \frac{R_1 - R_2}{2R_1}\right) + \frac{B}{2R_1}$$

$$\cong \frac{R_1 - R_2}{B} \text{ when } B << R_1, R_2$$

FIGURE 4. Long baseline inverse triangulation direction measurement.

FOR SHORT OR LONG BASELINES:

$$\cos\theta \cong \frac{R - R}{B} \approx \frac{\Delta R}{B}$$

WITH ERRORS IN ΔR and B:

$$(E)\theta \cong \frac{(E)\Delta R}{B}\,\frac{1}{\sin\theta} - \frac{(E)B}{B}\,\frac{1}{\operatorname{Tan}\theta}$$

	SHORT BASELINE 150 METERS	LONG BASELINE 1500 kilometers	
ERROR CONTRIBUTIONS	30 microradians	30 microradians	3 microradians
RANGE DIFFERENCE (E)ΔR =	3.2 mm	32 meters	3.2 meters
BASELINE LENGTH (E)B =	4.5 mm	45 meters	4.5 meters
BASELINE ORIENTATION (E)B =	4.5 mm	45 meters	4.5 meters

NOTE: ERROR AT $\theta = 45°$

FIGURE 5. Error in ground tracking direction measurement.

from the several stations must be individually collected and then processed for determination of direction.

Realization of directional measurement accuracy improvement by increasing baselength requires that the range measurement and baseline errors accumulate less rapidly then does the baseline increase. The practicability of this can be seen by examining FIGURE 5. This shows the range and baseline errors needed to maintain 30 microradian direction error near 45 degrees for baselines of 150 meters and 150 kilometers.

Improvement of the 150 kilometer long baseline system by a factor of ten to 3 microradians seems feasible, as shown in the figure. However, at this precision and better, a serious question arises, particularly with the necessary accuracy of station location. This is clearly a problem of survey and geodesy. The precise knowledge of the size and the shape of the earth is a question actively being pursued and about which agreement does not exist.

Ground tracking ranging and directional measurements described above provide the basis for determining directly all components of the position and velocity of a spacecraft. The error values of our hypothetical model above by no means provide the accuracy limit from ground tracking. Considerable improvement for a given station array can be demonstrated by calibration techniques in tracking targets and applying corrections to fit the known target motions.

In any event, range and direction error in ground-based tracking navigation of spacecraft is roughly proportional to the distance of the vehicle from earth. Future missions will require more and more accuracy of measurement with respect to the moon and planets as maneuvers and mission objectives related to these bodies become more complex. This leads to the second category of navigation measurements: those using sensing devices on-board the spacecraft itself.

Spacecraft-borne navigation measurement tends more to optical frequency and direction measurement rather than the radio frequency direct ranging that is so accurate for ground tracking. For relatively close work, direct ranging with radio frequencies with rendezvous or landing radars becomes possible, albeit necessary. But further from the planets and other targets, direct measurement of range or range rate, or the use of radio frequencies has not appeared attractive to the designers due to the weight and power penalties.

Spacecraft on-board directional measurements are those made to the near bodies — the sun, moon, earth, and other planets. The stars provide no position data because of their extreme distances. But because of this distance they are still, as they have always been, most excellent direction references against which to measure the directions to the nearer bodies, when suitably corrected for aberration due to spacecraft motion.

In a sense, then, on-board navigation is performed by observing the near bodies relative to the background stars. This can be done indirectly by measuring the angles sequentially from a gyro stabilized base to the stars and the near body. Alternately a direct and simultaneous measurement of the angle between a reference star and the near body with a suitable sextant-like instrument avoids an accumulation of errors with which the former sequential technique must cope.

The ancient sextant, updated and refined with a suitable telescope for image resolution and with a precision angle readout of the deflecting mirror, can provide in a reasonable size the angle between a feature of a near body and a star superimposed upon that feature in the field of view to accuracies of the order of 50 microradians or better.

The "feature" alluded to above is some distinct point on the planet to which the direction is being measured. The center of the disk naturally comes to mind, but identifiable surface landmark features and horizons that can be related to planet coordinates are easier and more accurate for visual use, particularly under crescent illumination.

Use of the planet horizon brings up the question of the accuracy of the altitude knowledge of the detected horizon. For the moon, for instance, the effect of mountains on the limb add uncertainties. For larger planets holding significant atmosphere, the true surface horizon at visual frequencies cannot be seen from space due to scattered sunlight from higher altitudes. For the earth, at least, it appears that the systematic brightness variation with altitude of this scattered light can be utilized with photometric measurement to fix the altitude of the line of sight within about 1000 meters (Seward, 1962). This accuracy figure applies roughly to landmark feature location also.

Thus from sextant and sextant-like measurements, directions can be determined with accuracies, for instance, of the order of 50 microradians to targets with an additional target positional accuracy of the order of 1,000 meters. For distances greater than 20,000 kilometers the 50 microradians dominates. However, closer navigation is limited by the location knowledge of the target features being used on the planet.

Each such direction measurement from the spacecraft provides a locus surface of spacecraft position. Several together define position uniquely at the common loci intersection. Here we see that range information is determined indirectly from the combination of direction data in a fashion not unlike triangulation, where the baseline is the known distance and direction between the target features of the planets. This in effect includes the stadiometric ranging made by measuring the apparent diameter of a planet disk.

Measurements separated in time provide the basis for velocity determination.

To obtain three components of position in the presence of spacecraft motion, one would desire the simultaneous measurement of at least three directional components. Practical considerations make time sequential directional measurements easier, and no direct computation of position or velocity is possible by purely geometric calculations.

Current schemes (Battin, 1964) depend upon the use of an on-board computer, programmed to accept the sequence of single coordinate navigation data and the precise time each measurement occurred. Each datum point is received and used to update and improve in an optimum fashion the six dimensional state vector of the spacecraft recognizing the expected error in each measurement and the motion constraints of the spacecraft in free fall.

In summary, we can compare the similarities and differences between navigation of a spacecraft in free-fall using earth-based tracking measurements and using vehicle-borne direction measurements as follows:

1. The two categories of navigation measurement complement each other in that earth-based tracking gives strong results along the line of sight from earth, while on-board measurement adds strength across the line of sight. The latter is particularly accurate at distances far from earth and near a target planet.

2. Both categories depend upon optimum processing of data points taken over a period of time, recognizing known measurement uncertainties and spacecraft motion as constrained by orbital mechanics. Both categories use the past history of data to determine present position and velocity as limited by data uncertainty and can predict future motion further limited by the lack of knowledge of the space environment.

3. Availability of earth-based navigation data from a given station is dependent upon the spacecraft being sufficiently above the horizon for that station. Spacecraft-based navigation measurements must compete for control availability with other operations of the spacecraft.

4. Earth-based navigation stations can support simultaneously only a limited number of missions. Spacecraft-based equipment, of course, is solely available for use of that mission.

5. Earth-based navigation tracking facilities are most limited by economic factors in the attempt to gain more capability by the use of many large radio tracking installations with complex communication networks and data processing centers. Spacecraft-based navigation is limited more by the weight that can be carried in the sensors and data-processing computers on board.

6. Earth-based navigation tracking facilities have the strong advantage of multiple use and re-use in sequential support of many types of missions. Spacecraft-based navigation equipment is, in a sense, consumed, and even if the equipment is recovered it is doubtful if it would be re-used.

7. Earth-based navigation measurements fail while the spacecraft is passing in back of its target planet. This is unfortunate since efficient orbital insertion and transearth escape maneuvers always occur in back of the moon and have a strong probability of being out of sight for other planets.

8. Earth-based navigation is vulnerable to enemy action against military spacecraft. Spacecraft-based navigation measurement can be strictly passive for military use and is invulnerable to jamming or sabotage activity.

Attitude Control

(Rotational Control in Free-Fall Flight)

Attitude control is the process of aligning the spacecraft to a desired orientation in response to commands and in the face of disturbing torques. As defined here the operation of attitude control applies to free-fall coasting flight only. The diverse nature of the problem is seen in terms of: (a) the orientation requirements; (b) the attitude-sensing techniques; (c) the nature of the disturbing torques; and (d) the technique of applying the control torques. These will be discussed briefly to show the wide spectrum of problems and solutions that appear in designing attitude-control systems for spacecraft.

The orientation requirements are naturally a function of the vehicle's mission and the associated operating constraints:

(a) Scientific payloads of a radiation or field-sensing nature generally have pointing requirements for the sensitive axis of the instrument. Often these aiming requirements are not particularly stringent, but again others such as astronomical telescopes can require the utmost in accuracy and stability of aiming.

(b) Spacecraft management orientation constraints generally are of a low order of accuracy. These include (1) aiming of solar cells for power gathering; (2) aiming of communication, telemetry, transponder, and beacon antennas toward earth; and (3) the maintenance of thermal balance by controlling attitude with respect to the sun.

(c) Navigation and guidance functions require attitude control arising from (1) the need to point the operating field of the navigation sensors towards the desired portion of the sky, and (2) the need for initial pointing of the rocket engines just prior to ignition for a trajectory correction or major maneuver.

This multitude of possible requirements can lead to impossible conflicting situations which are relieved only by mounting the lightweight instruments on articulating gimbals to make them at least partially independent of spacecraft attitude.

The attitude-sensing function is also performed in a number of ways:

(a) In some cases radiation-sensing instruments requiring pointing can be made to track the sensed flux themselves by providing error signals to the control system.

(b) For earth-orbital spacecraft the most common attitude sensing uses infrared horizon detectors to indicate spacecraft orientation deviations from local vertical. These, used in conjunction with a gyroscope reference, can also provide the attitude about the local vertical with respect to the orbital plane. This process is similar to the earthbound gyrocompass in that the pendulum is replaced by the horizon detectors and the earth's rotation is replaced by the rotation in orbit.

(c) Basic attitude sensing for small cislunar and interplanetary vehicles most often depends upon a sun seeker/tracker to set up a vehicle axis with respect to the sun, combined with a star tracker offset by an adjustable angle to acquire and track a star to provide attitude sensing about that sun line.

(d) Once an orientation reference is established this can be maintained by the use of gyroscopes to detect deviations from the reference. Gyroscopes also provide capability to meter orientation changes accurately from the attitude established by other means.

The disturbance torques upset spacecraft orientation and cause the need for correction from the control system:

(a) Lightweight vehicles can be affected by the relatively weak forces associated with the space environment. For spacecraft with large unsymmetric surfaces with respect to the center of mass, radiation pressure from the sun is a significant torque disturbance. Lightweight vehicles also may be affected by interaction of electrical current loops or other spacecraft magnetic sources with the earth's field. Electrostatic forces, unsymmetric atmosphere drag, and the integrated effect of micrometeoroids have also been suggested as a source of disturbance torques.

(b) Vehicles having one long dimension resulting in a wide difference in the principal moments of inertia can be strongly affected by differential gravity forces near a massive planet.

(c) Spacecraft will experience disturbance torques any time mass is thrown off. This can occur for instance, by the venting of cryogenic fuel boiloff or the offloading of other wastes.

(d) Relative acceleration of masses within the vehicle cause a redistribution of angular momentum with associated torques. Speed changes of on-board rotating machinery, the pumping or sloshing of fluids, or the process of erection of solar panels or antennas are examples. On manned craft the movements of the crew cause significant disturbance.

Control torques to counteract these disturbances or to reorient the vehicle can utilize one of three phenomena.

(a) The weak forces associated with the space environment can be utilized in a passive or semipassive attitude control. Self-aligning mechanisms based upon solar radiation pressure, magnetic field torques, or gravity gradient unbalances can provide weak but often adequate restoring torques to a stable orientation satisfactory for some missions. Some form of energy dissipation for damping oscillations must be provided.

(b) Small reaction rocket engines arrayed to provide suitable torque couples depend upon angular momentum transfer to the exhausted gas. These are usually chemical or cold gas low thrust engines designed for as many on-off cycles as demanded by the control loop. Control is characterized by pulsed operation of the jets and limit cycle oscillation about the desired attitude.

(c) Flywheel or gyroscope momentum-exchange systems achieve control torque by either accelerating a heavy flywheel or precessing a spinning gyro. Unlike the jet or rocket systems above, only power is consumed and operation is not limited by the amount of working fluid carried. However, there is a capacity limit in the sense that there is a maximum momentum that can be stored by practical speeds of heavy flywheels or gyrowheels. Thus, in application, these momentum exchange systems are used in conjunction with periodic use of a jet or other type of external torques to "desaturate" the system back to its control range. Finally, a simple spin of the whole spacecraft itself can often provide adequate simple means of stabilization.

The design of attitude control systems is complicated by a number of factors. The classical equations of motion under assumptions of spacecraft rigidity are straightforward. But even though it is theoretically possible to predict the rotational motions of the vehicle, a simple control system cannot make large rotational changes directly when the desired axis of rotation does not coincide with one of the principal axes of inertia. Usually the torquing axes are close to these principal axes and large rotational maneuvers are made sequentially axis by axis. This may be less efficient than a hypothetical control system that would control to the shortest path achieved by applying a single torque impulse. This impulse would create that angular momentum vector which will carry the vehicle into the desired orientation by free-tumbling rotation where a second impulse would stop it.

The energy used in a rotational maneuver is directly a function of the speed with which the maneuver must be accomplished. Rather than build up kinetic energy in a fast turn only to cancel it again at the destination orientation with an opposite impulse, the designers tend towards very slow rotation rates for the large turns when mission requirements permit.

Another complicating factor occurs when the spacecraft carries a significant mass of fluid fuel. A practical attitude control cannot measure the angular momentum contribution of this fluid since its loose coupling to spacecraft allows it independent motion. Again the theoretically most efficient application of control torque cannot be achieved by simple attitude control systems.

Reaction jet control engines are characterized by fixed torque levels during firing and a wearout limit on the number of individual firings. Since disturbance torques are generally less than the available control torque, the attitude control system provides on-off cycles of firing, resulting in a limit cycle oscillation about the desired orientation. The total jet fuel used and the number of on-off cycles should be minimized by optimization of the control system.

Thrust Vector Stabilization and Control

(Rotational Control During Accelerated Flight)

Thrust vector stabilization and control is the closed-loop process which (1) keeps the vehicle attitude from tumbling under the high forces of engine firing, and (2) accepts turning or guidance steering commands to change the direction of engine-caused acceleration.

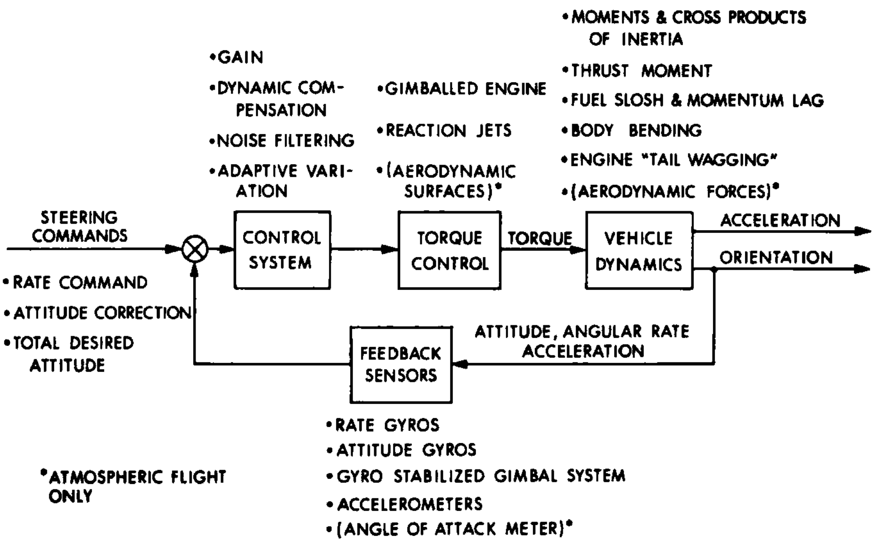


FIGURE 6. Thrust vector stabilization and control.

FIGURE 6 illustrates a generalization of thrust vector stabilization and control. In order to illustrate the variations possible, the boxes in the figure may contain one or more of the aspects listed with "dot" prefixes adjacent to the boxes. These systems are characterized by appropriate feedback to provide a stable control of angle or angular velocity of the thrusting vehicle. The loop also accepts input steering commands from guidance to achieve a particular desired thrust direction.

The design constraints on thrust vector stabilization and control systems represented by FIGURE 6 vary considerably. The figure lists typical variations possible in the spacecraft body dynamics and the torque-producing control devices. Most of these characteristics are not only gross nonlinearities but are time variant and interacting as well. The design is further complicated by necessary constraints on dynamic response to inputs and disturbances. It is usually restricted by allowable limits on angular acceleration, angle of attack, and other variables depending on

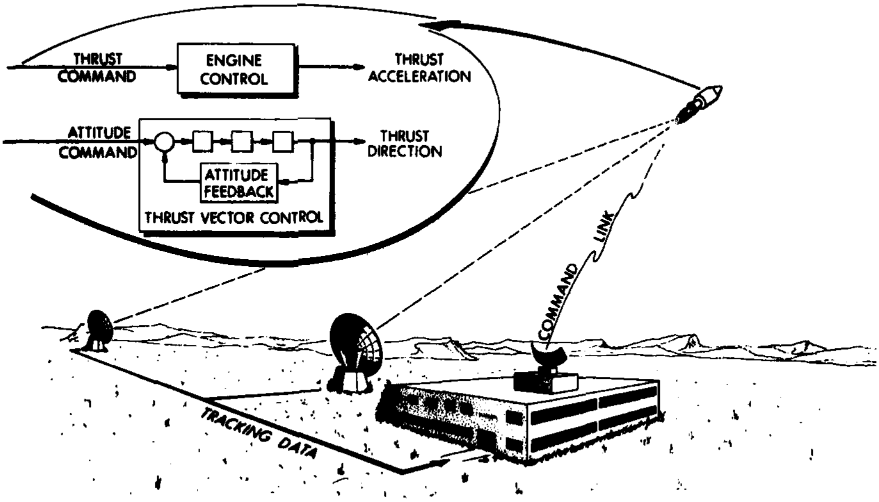


FIGURE 7. Radio command guidance.

structural and controllability considerations. All this and the usual concern about reliability, weight, cost, etc. makes design particularly difficult.

GUIDANCE

(Translation Control During Accelerated Flight)

Guidance is the process of measurement and computation necessary to provide steering signals to the thrust vector control system and signals to modulate engine thrust level in order to achieve vehicle acceleration to a desired trajectory. Modulation of engine thrust level in the more common case of a nonthrottleable engine consists only of turn-on and cut-off commands.

Powered steering of some of the early ballistic missiles and workhorse spacecraft launch vehicles used ground tracking data in a radio command guidance (see FIGURE 7). This type of guidance is characterized by a continuous ground tracking monitor of position and velocity changes during the powered phases and a radio command to the vehicle to change the direction of thrust appropriately — and finally to signal thrust termination. A basic requirement is an attitude reference system carried aboard the vehicle. This is illustrated in FIGURE 7 as the attitude feedback, implemented with gyros for instance, as part of the thrust vector control system.

Far from the earth, delays occur associated with necessary longer smoothing of the noisier tracking signals and delays associated with the finite speed of electromagnetic propagation. For deep-space spacecraft requiring short burn trajectory corrections of moderate accuracy these delays are not significant since the ground command need only specify the direction and length of burn required. However, for precise, long duration maneuvers the thrust vector control alone cannot assure accuracy in metering the direction or magnitude of the specified velocity change. And the mentioned delays in the receipt of the steering commands make loop closure corrections of questionable effectiveness. Here inertial guidance is the only practical method of powered steering control.

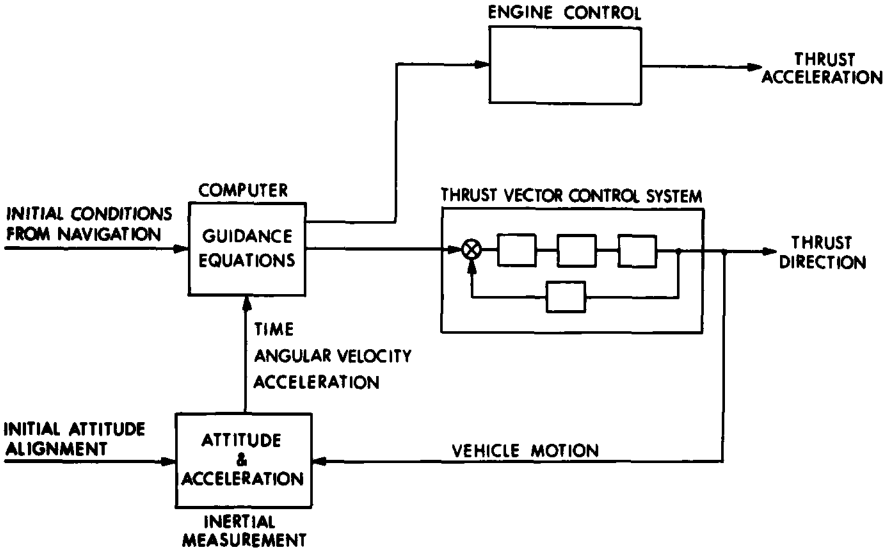


FIGURE 8. Inertial guidance.

Inertial guidance (see FIGURE 8) is based upon measurements of vehicle motion using self-contained instruments which do not depend upon radiation sensing. In every inertial guidance system three types of measurements are made involving distinctive instruments: (1) angular rate or direction using gyroscopic devices; (2) linear acceleration using restrained test masses in accelerometers; and (3) time using precision reference frequency sources. The integration with time of the sensed acceleration in the indicated direction with proper recognition of known gravity forces is the essence of inertial guidance. The implied processes are accomplished in a computer with the result of converting the input data into steering commands to the thrust vector control system. Since inertial guidance of the type described can only integrate vehicle motion into changes in position, velocity, and orientation, accurate initial conditions are required in these parameters before the accelerated guidance phase is initiated. Initial conditions in position and velocity are provided by navigation. Initial conditions in orientation come from the attitude control systems or directly from stellar references.

One well-debated problem with inertial guidance is the presence of an increasing error with time. When an error in the gyro data, commonly called gyro drift, is processed in the computer the direction of the controlled acceleration is in error. When an error in the acceleration sensing exists, again the direction of acceleration and the controlled length of motor burning are affected undesirably. However, due to the motivation to perfect inertial instruments for their well-adapted use in military guidance and navigation, the technology is advanced to the point that inertial guidance performance can be kept always abreast of needs for spacecraft missions in controlling accelerated flight. Furthermore, spacecraft inertial guidance can be tolerant of error, in the sense that errors in the resulting trajectory usually can be measured later by navigation and corrected with a short burn of the propulsion.

It is perhaps pertinent to examine these last statements with respect to two propulsion situations, which undoubtedly will exist in the future. The first is that of the high specific impulse, low thrust electric engines. Here the very low thrust to mass ratio requires long periods of controlled engine operation — measured in weeks. In such long periods, inertial guidance measurements alone without recourse to periodic external navigation measurements would be unacceptable, even if the inertial sensing were perfect. The inertial system cannot sense the perturbations in trajectory caused by the imperfectly known gravitation forces. Such systems then require periodic navigation by onboard or ground-tracking measurements. It is doubtful whether these navigation checks would be needed any more often than during the coasting free-fall phases with the more conventional chemical engine mission.

The second future propulsion situation is that which will exist with high thrust nuclear rockets providing more abundant total impulse. In this realm, mission times will be shortened by longer burning to higher interplanetary velocities than permitted with current chemical engines. In spite of the larger velocity changes to be measured during thrust by the inertial sensing, the dramatic shortening of the subsequent time of flight is enough to decrease required measurement precision for the same accuracy in arrival at the destination planet.

Problem of Preflight Verification

This is the first of several typical problems which are of concern in the navigation, guidance, and control of spacecraft.

These systems are characterized by complicated dynamics resulting from hardware imposed nonlinearities typical of those mentioned earlier. Because of the tremendous waste in resources — and the important loss of prestige — flight failures due to lack of understanding of vehicle control characteristics are particularly repugnant. Extensive experimentation on the ground using digital and analog simulation and involving flight hardware where realistically possible is mandatory. But many important parts cannot be tested with real hardware and must be simulated. The simulation models must depend upon accurate characterization of all important features of the equipment represented. Yet important nonideal characteristics which may not be suspected, let alone be accurately measured, can be lurking. Devices may have deadbands, saturations, noise sources, hystereses, and a myriad of other things which can show up unanticipated for the first time in flight. The designer, once satisfied that he has made the best effort to uncover all the important idiosyncrasies, attempts to design his control loops with dynamic margins which will absorb harmlessly the unknown. Fortunately, the record is good.

Allied to this subject is another aspect which has on several occasions caused unforgivable embarrassment. No excuse for polarity error can be given even though magnitude is accurate. Reversed wires and reversed signals can somehow be overlooked through several cycles of checking. Simulations are helpful only when the error also appears in the simulation. No relaxation of discipline or checking can be justified.

Problem of Comparative Evaluation of Systems

What makes one system better than another? When the judgment is on a single parameter the answer may be clear. But many parameters seem to be involved for relative evaluation of spacecraft equipment: performance, reliability, weight, power drain, and cost are familiar. Other aspects such as testability, history of successes, simplicity, familiarity with the user and delivery schedule also are judgment values. No satisfactory way for scoring when all these are involved can be claimed to be universally acceptable. Each situation differs and overall excellence is a subjective evaluation.

1. FOR IMPROVED ACCURACY IN GUIDANCE OF A MAJOR MANEUVER

   a.) INCREASE GUIDANCE MASS BY ΔM_g

   b.) OFF-LOAD THE SAME MASS OF FUEL $\Delta M_{f1} = \Delta M_g$

2. WITH IMPROVED ACCURACY, LATER TRAJECTORY CORRECTION

   a.) IS REDUCED BY ΔV_c

   b.) WHICH SAVED MASS OF MANEUVER FUEL $\Delta M_{f2} = \dfrac{M_s}{I_{sp}\, g_e} \Delta V_c$

   WHERE M_s = MASS OF SPACECRAFT AT CORRECTION MANEUVER
   AND $I_{sp}\, g_e$ = ROCKET EXHAUST VELOCITY

3. ADDED GUIDANCE WEIGHT WAS ADVANTAGEOUS IF $\Delta M_{f2} > \Delta M_{f1}$

   OR $\dfrac{\Delta V_c}{\Delta M_g} > \dfrac{I_{sp}\, g_e}{M_s}$

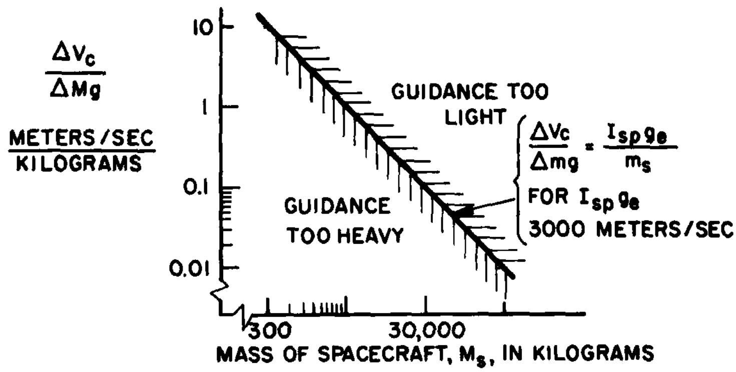


FIGURE 9. Weight performance comparison.

However, a reduction in the number of judgment variables can be made for specific use of equipment for a specific mission plan. For instance, several of these variables can be reduced rather explicitly to equivalent weight. FIGURE 9 shows the relationship between performance and weight (expressed as fuel mass) for a simple mission maneuver. In principle this relationship can be expanded to cover a complete mission.

Likewise equipment power drain can be related to the weight of the power source and associated thermal control system to remove dissipated power.

The investment of design time and material expense can reduce the weight of equipment without adversely penalizing other factors. When the reduction in cost propulsion for accelerating less mass exceeds the increase in cost needed to reduce the equipment mass, then the greater equipment expense is desirable.

A similar argument can be made for the parameter reliability. If one accepts heavy conservative design as being more reliable the relationship to weight is achieved in principle. If an equipment weight savings is made at the expense of reliability, this saved weight might be applied to other spacecraft equipment of marginal reliability such as to increase overall spacecraft reliability.

In a sense these relationships could be considered less than satisfactory in leading to a single scoring parameter. However, they do suggest an approach to making a more objective overall appraisal of spacecraft systems.

Problem of Astronaut Involvement

There is no question about the desirability of involving man in space flight, even if it is only justified for its own sake. The proper question here is the degree to which man's talents should be used in rotational and translation spacecraft management once he is on board. Many tasks are best left to the machine: those that are too tedious or require too much energy, speed of response, or accuracy outside man's limits. His assets, however, are many. His judgment and adaptability, his decision-making in the face of the unanticipated, and his unique ability to recognize and evaluate patterns are valuable characteristics. Of this latter, consider his unsurpassed faculty to pick out a particular navigation star from the heavens or to evaluate a suitable touchdown spot on the moon. But providing for man's involvement is costly. To supply acceptable displays and controls with proper characteristics for smooth emergency transition from automatic to manual operation may require unfortunate design compromises.

The resolution of the dilemma must certainly recognize that man shouldn't be relegated to the status of a mere passenger or, on the other hand, be over-burdened with work. We should recognize that eventually man ought to be provided with a space vehicle of which he is the master.

Problem of Special Military Needs

Until mankind can find the solution to enmity, the use of space for military advantage is inevitable. Avoiding the use of weapons in space as a national policy is based upon ethical considerations, but should be abandoned unless universally honored.

The requirements on the military spacecraft are similar to those of any weapon. For navigation, guidance, and control the requirement for quick action and reaction on command is indisputable. These military space systems must be designed for flexilibity to respond to changing situations or provide evasive tactics rather than follow a rigid preprogrammed mission. For strategic weapons, performance must be adequate and demonstrable to be credible. Reliability, as always, must take prime concern.

Of particular consequence for military spacecraft is the need to provide complete on-board capability. Ground tracking and control stations are vulnerable to sabotage and jamming. Exploitation of and experience with on-board navigation measurements in peacetime spacecraft and the continued perfection of inertial guidance can be justified for their manifest need for future military use. It is more than a question of national pride; it may be a question of national survival.

References

Battin, Richard H. 1964. Astronautical Guidance. McGraw-Hill Book Co. New York, N.Y.

Seward, H. H. 1962. The blue-white boundary horizon sensor, MIT Instrumentation Laboratory Report E-1206.