AAS/AIAA
Astrodynamics Specialist Conference

Westin Alyeska Prince Hotel
Girdwood, Alaska

August 16-19, 1999

PROGRAM

	General Chairs	Technical Chairs
AAS	**Bernard Kaufman** **ASTRO DYN** **6408 Glen Oak Drive** **Temple Hills, MD 20748** **301-449-7498 Voice** **301-449-2986 Fax** **email:** bkaufman@erols.com	**Kathleen Howell** **School of Aeronautics & Astronautics** **GRIS 1282, Purdue University** **West Lafayette, IN 47907** **765-494-5786 Voice** **765-494-0307 Fax** **email:** howell@ecn.purdue.edu
AIAA	**Terry Alfriend, Professor & Head** **Aerospace Engineering Dept.** **H.R. Bright Bldg., Room 701** **Texas A&M University** **College Station, TX 77843-3141** **409-845-5920 Voice** **409-845-6051 Fax** **email:** alfriend@aero.tamu.edu	**Felix Hoots** **GRC International, Inc.** **985 Space Center Drive, Suite 310** **Colorado Springs, CO 80915** **719-596-5395 Ext. 216 Voice** **719-596-6139 Fax** **email:** fhoots@grci.com

TABLE OF CONTENTS

MEETING INFORMATION

REGISTRATION: The following registration fees will be in effect for the meeting:

AAS and AIAA Members	$190
Non-members	$250
Students	$ 25
Non-members	$290

(and includes 1 year new membership in AAS, does not apply to membership renewal)

The registration desk will be open:

Sunday	03:00 PM - 05:00 PM
Monday, Tuesday, Wednesday	08:00 AM - 05:00 PM
Thursday	08:00 AM - 10:00 AM

Please note that credit cards cannot be accepted for payment of any fees.
Checks payable to the American Astronautical Society are the preferred form of payment.

SOCIAL EVENTS:

Sunday Evening	Cocktail party for early registrants	06:00 PM - 08:00 PM	Prince Court
Monday Evening	Reception	07:00 PM - 09:00 PM	Columbia Ballroom
Tuesday Evening	Dinner	07:00 PM - 10:00 PM	Glacier Express
Wednesday Evening	Nothing currently planned		

Please address questions or comments to one of the General Chairs:

Bernard Kaufman
AAS General Chair
ASTRO DYN
6408 Glen Oak Drive
Temple Hills, MD 20748
301-449-7498 Voice
301-449-2986 Fax
email: bkaufman@erols.com

Terry Alfriend
AIAA General Chair
Aerospace Engineering Dept.
H.R. Bright Bldg., Room 701
Texas A&M University
College Station, TX 77843-3141
409-845-5920 Voice
409-845-6051 Fax
email: alfriend@aero.tamu.edu

TECHNICAL PROGRAM

SPEAKERS BRIEFING: Authors who are presenting papers, and session chairs will meet for a short briefing each morning from 07:00 - 08:00 a.m. in the Prince Court (Monday, Tuesday, Wednesday) or the Bering Room (Thursday). Please attend only on the day of your session. A continental breakfast will be provided.

PRESENTATIONS: Each paper will be allotted 20 minutes (including introduction and question and answer period). Please note that a "no paper-no podium" rule will be strictly enforced for all presentations. An author will not be permitted to give his or her paper if a written paper has not been prepared and made available at the conference. Also, papers will be automatically withdrawn from the meeting and will not be eligible for inclusion in the proceedings if one of the stated authors is not in attendance to present the paper.

PAPER SALES: Authors have been requested to bring 50 copies of their paper to the meeting. These reprints will be on sale for $1 per paper in the Harding Room. Bound copies of the proceedings may be ordered at the registration desk.

COMMITTEE MEETINGS: Committee meetings will be held in the Portage Salon according to the following schedule:

AIAA Astrodynamics TC Meeting	Monday, 8/16	12:00 PM - 02:30 PM
AAS Spaceflight Mechanics TC Meeting	Tuesday, 8/17	12:00 PM - 02:30 PM
Astrodynamics Committee on Standards	Wednesday, 8/18	12:00 PM - 02:00 PM
Coffee Breaks - Columbia Ballroom Foyer	Monday-Thursday	09:50 AM - 10:10 AM
	Monday-Wednesday	02:50 PM - 03:10 PM

Please address questions or comments on the Technical Program to one of the Technical Chairs:

Kathleen Howell
AAS Technical Chair
School of Aeronautics & Astronautics
GRIS 1282, Purdue University
West Lafayette, IN 47907
765-494-5786 Voice
765-494-0307 Fax
email: howell@ecn.purdue.edu

Felix Hoots
AIAA Technical Chair
GRC International, Inc.
985 Space Center Drive, Suite 310
Colorado Springs, CO 80915
719-596-5395 Ext. 216 Voice
719-596-6139 Fax
email: fhoots@grci.com

GIRDWOOD - SOUTH CENTRAL ALASKA

(Some of the following information can be found at http://www.alaskaonline.org/travelplanner/southcentral/Girdwood.htm)

Location: Located on the northern coastline of the Turnagain Arm, 40 miles south of Anchorage International Airport via the Seward Highway. Girdwood is part of the Municipality of Anchorage. Only 11 miles from Portage Glacier, Alaska's most visited attraction, often referred to as the "Gateway to the Kenai Peninsula."

Westin Alyeska Prince Hotel is located at 1000 Arlberg Road, Girdwood, AK 99587. Phone: 907-754-1111 or 1-800-880-3880 Fax: 907-754-2200. Check-in is 3:00 p.m. and check-out is noon. Modernistic, chateau-style facility accented by natural bridges/patios. Nestled among majestic mountains and the Chugach Forest. There are 307 rooms, self parking, 24-hour security coverage, service desk, photocopying, translating audio equipment, audio headsets for the hearing impaired, video teleconferencing capabilities, secretarial support, and babysitting services. Rooms have mini refrigerators, safes, data ports, hair dryer, non-smoking and special needs accommodations. The fitness center includes stationary bicycles, treadmills, stair-steppers, rowing machines, free weights and weight machines, a 16-person mountain view window-side whirlpool, massage therapy, lap pool, and sauna.

Climate: Temperatures range from 50 to 70 degrees F. in summer.

History: The Town was originally named Glacier City and acted as a gold mining and railroad supply camp in the early 1900s. Later it was renamed "Girdwood" in honor of James Girdwood, an Irish entrepreneur who came in 1896 and bought 4 mining claims along Crow Creek. In 1906, the tiny city boasted fifteen buildings including a bathhouse, stable, blacksmiths, commissary, and 5 saloons. On weekends, the tiny city was flooded with over 300 railroad workers, miners and visitors from Anchorage. In 1923, the construction of the Crow Creek Highway provided easy transportation for miners and their equipment. In 1924, the town became a movie set for the silent screen frontier saga: "Cheechakos" produced by Austin Lathrop (a copy remains in the archives of the Anchorage Museum of History of Art). Gold operation ceased before WWII by presidential order. In 1960, the first chairlift was built. In 1961, the town became incorporated. After the good Friday Earthquake of 1964 which destroyed the townsite and the airstrip, the town was relocated at the present location in early 1965. Construction of the Alyeska Highway was terminated in late 1964. In 1977, the post office enters their own building. Today, the town economy is based on the ski resort and tourism. There are accommodations of all kinds, restaurants, coffee shops, grocery stores and general stores, a post office, a gas station, a school, rafting organizations, guides and airflight operators at the state-owned airstrip.

How to get to Girdwood: Off the Seward Highway, 39 miles south of Anchorage, turn east on the Alyeska Highway. Buses, shuttles, limousine and taxi are available to drive you there from Anchorage. No scheduled flights available. Local charters may provide quick commuting to Anchorage, Kenai Peninsula, and elsewhere. Check at the municipal airport. No harbor exist at Girdwood, and no facility for boat launch is available.

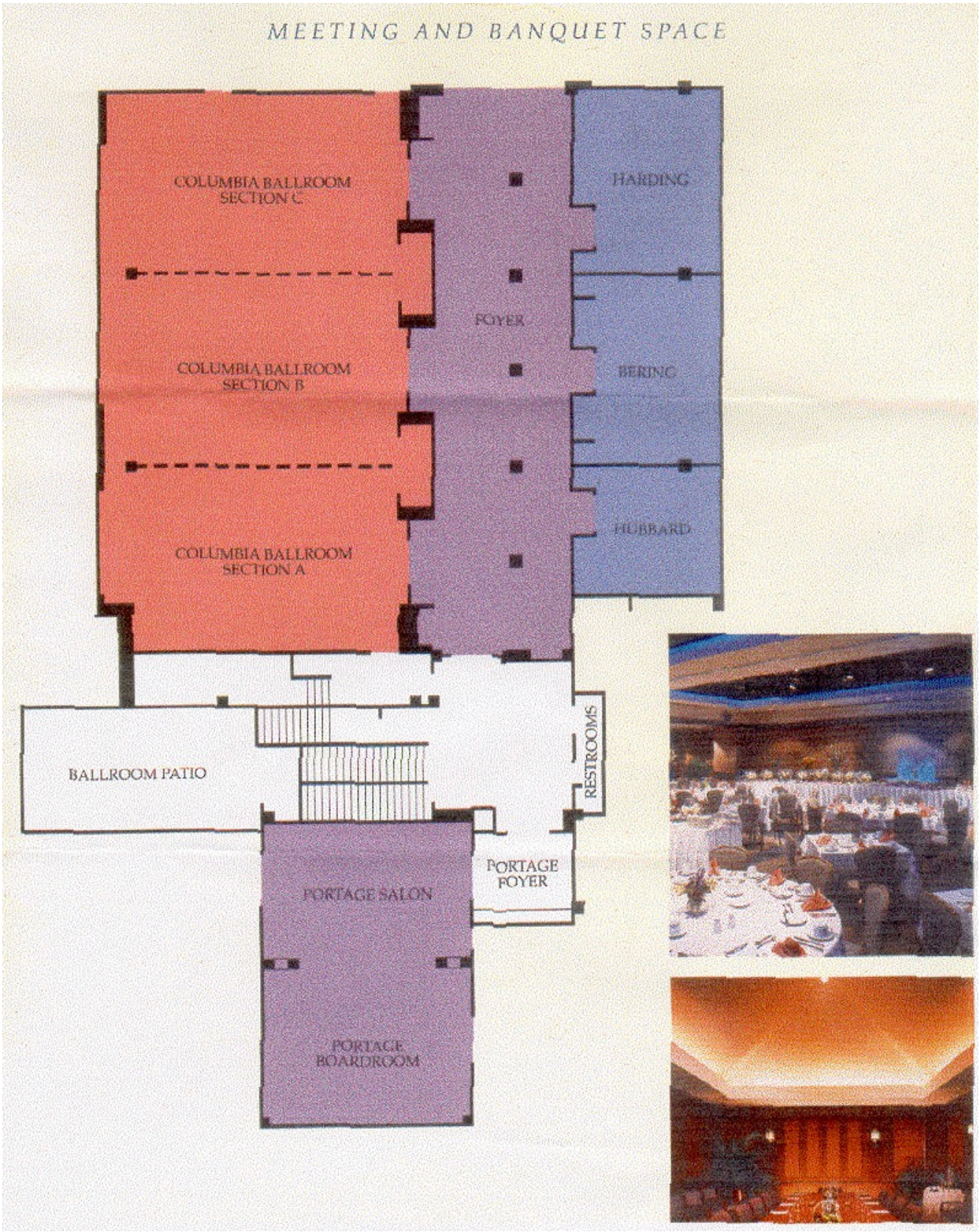
Outdoor Activities: A nearby 18 hole golf course with tee times as late as 10 p.m. Nearby historic Crow Creek Mine for gold panning. Flightseeing tours of surrounding valley and nearby Prince William Sound departing only 5-minutes from the hotel. Fishing, hunting (Girdwood is in unit 14C), hiking, mountain biking, rafting, kayaking, canoeing, rock climbing, paragliding, berry picking, sightseeing and more.

Movies: Regal Valley River Cinemas 6, 11801 Business Blvd. in Eagle River.

Bars: Max's Bar & Grill, Crow Creek Highway in Girdwood.

Restaurants: Alpine Diner & Bakery, Bake Shop, Chair 5, Double Musky Inn, Girdwood Griddle, Mezz, Tidewater Café and Yang's all located in Girdwood. Also available are The Pond Café Katsura Teppanyaki (Japanese steakhouse), Seven Glaciers Restaurant and Lounge (2,300' above the valley), Glacier Express, and The Kiosk (pre-made sandwiches, baked goods, and gourmet coffee).

Westin Alyeska Prince Hotel Floor Plan 1

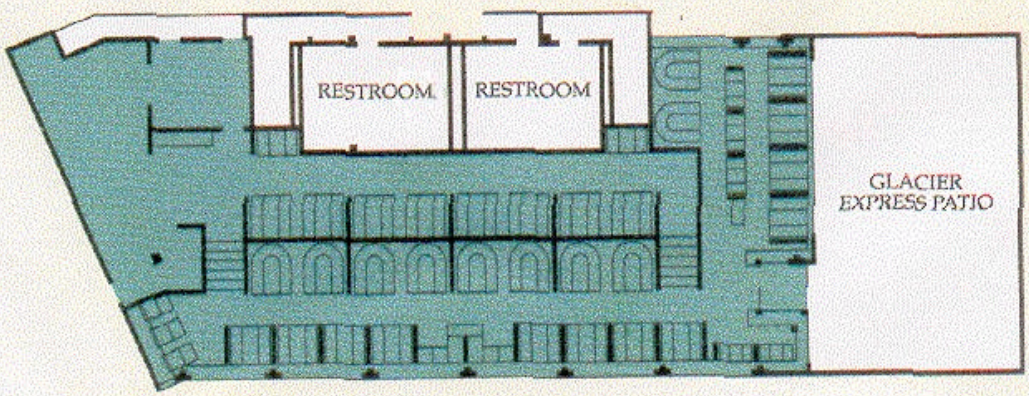


Westin Alyeska Prince Hotel Floor Plan 2

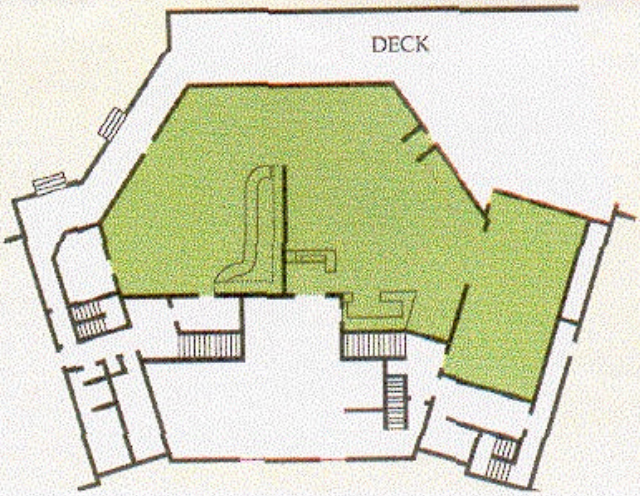
CAPACITIES FOR SUCCESSFUL MEETINGS

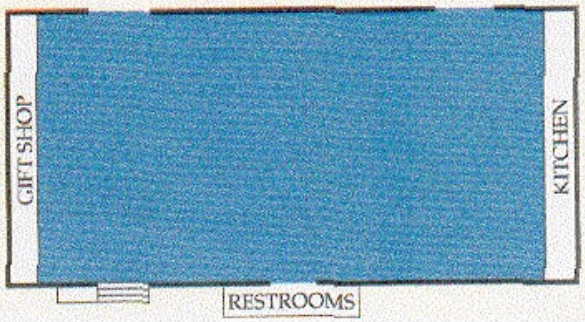
	Dimensions	Square Feet	Ceiling Height	Theater	Classroom	Conference	Banquet	Reception
Columbia Ballroom	41' x 74'	3,034'	9' - 17'	340	200	—	250	450
Section A	41' x 24'	984'	9' - 14'	100	60	34	60	100
Section B	41' x 26'	1,066'	9' - 17'	120	60	34	70	100
Section C	41' x 24'	984'	9' - 14'	100	60	34	60	100
Portage Boardroom	26' x 20'	520'	9' - 15'	NA	NA	16	—	—
Portage Salon	26' x 21'	546'	9' - 15'	40	24	24	30	40
Foyer	21' x 76'	1,596'	9' - 10'	—	—	—	—	150
Bering	22' x 25'	550'	9' - 10'	40	24	24	32	40
Harding	22' x 25'	550'	9' - 10'	40	24	24	32	40
Hubbard	22' x 16'	352'	9' - 10'	20	16	16	20	20
Daylodge	59' x 122'	7,198'	30'	450	340	—	340	500
Glacier Express	32' x 100'	4,000'	12' - 27'	NA	NA	NA	200	200
Sitzmark	76' x 48'	3,860'	12' - 30'	NA	NA	NA	200	300

Room diagrams are not equal to scale.



GLACIER EXPRESS





PROGRAM SUMMARY

DATE/TIME	EVENT	LOCATION
Sunday, August 15		
03:00 PM - 05:00 PM	Registration	Ballroom Foyer
06:00 PM - 08:00 PM	Cocktail party for early registrants	Prince Court
Monday, August 16		
08:00 AM - 05:00 PM	Registration	Bering Room
07:00 AM - 08:00 AM	Speakers Breakfast	Prince Court
08:30 AM - 11:50 AM	Morning Papers	Columbia Ballroom Section **A, B, or C**
09:00 AM - 05:00 PM	Paper Drop Off	Harding Room
09:00 AM - 05:00 PM	Paper Sales	Harding Room
09:50 AM - 10:10 AM	Coffee Break	Columbia Ballroom Foyer
12:00 PM - 02:30 PM	AIAA Astrodynamics TC Meeting	Portage Saloon
01:30 PM - 04:50 PM	Afternoon Papers	Columbia Ballroom Section **A, B, or C**
02:50 PM - 03:10 PM	Coffee Break	Columbia Ballroom Foyer
07:00 PM - 09:00 PM	Reception	Columbia Ballroom
Tuesday, August 17		
08:00 AM - 05:00 PM	Registration	Ballroom Foyer
07:00 AM - 08:00 AM	Speakers Breakfast	Prince Court
08:30 AM - 11:50 PM	Morning Papers	Columbia Ballroom Section **A, B, or C**
09:00 AM - 05:00 PM	Paper Sales	Harding Room
09:50 AM - 10:10 AM	Coffee Break	Columbia Ballroom Foyer
12:00 PM - 02:30 PM	AAS Spaceflight Mechanics TC Meeting	Portage Saloon
01:30 PM - 04:30 PM	Afternoon Papers	Columbia Ballroom Section **A, B, or C**
02:50 PM - 03:10 PM	Coffee Break	Columbia Ballroom Foyer
07:00 PM - 10:00 PM	Dinner	Glacier Express
Wednesday, August 18		
08:00 AM - 05:00 PM	Registration	Ballroom Foyer
07:00 AM - 08:00 AM	Speakers Breakfast	Prince Court
08:30 AM - 11:50 PM	Morning Papers	Columbia Ballroom Section **A, B, or C**
09:00 AM - 05:00 PM	Paper Sales	Harding Room
09:50 AM - 10:10 AM	Coffee Break	Columbia Ballroom Foyer
12:00 PM - 02:00 PM	Astrodynamics Committee on Standards	Portage Saloon
01:30 PM - 04:50 PM	Afternoon Papers	Columbia Ballroom Section **A, B, or C**
02:50 PM - 03:10 PM	Coffee Break	Columbia Ballroom Foyer
Thursday, August 19		
08:00 AM - 10:00 AM	Registration	Ballroom Foyer
07:00 AM - 08:00 AM	Speakers Breakfast	Bering Room
08:30 AM - 11:50 PM	Morning Papers	Columbia Ballroom Section **A, B, or C**
09:00 AM - 12:00 PM	Paper Sales	Harding Room
09:50 AM - 10:10 AM	Coffee Break	Columbia Ballroom Foyer

SESSION 1 *Monday, August 16, 1999*
8:30 AM – 11:30 AM

Session Room: Columbia Ballroom
Session Title: Mission Design: Mars
Session Chair: Robert G. Melton
233 Hammond Boulevard
University Park, PA 16802
Tel: (814) 865-1185
Fax: (814) 865-7092
Email: rgmelton@psu.edu

99-300 **Architectural Design for a Mars Communications & Navigation Orbital Infrastructure** -
0830am R.J. Cesarone, R.C. Hastrup, D.J. Bell, D.T. Lyons, K.G. Nelson

Mars has become the focus of an unprecedented series of missions spanning many years, numerous nations and an evolution from robotics to humans. Operations of this exploratory fleet will require implementation of a new communications and navigation architecture, satisfying the needs of robotic landers, rovers, ascent vehicles, sample canisters, balloons and airplanes, as well as eventual human explorers. NASAs Jet Propulsion Laboratory has begun development of this architecture, comprising Mars communications and navigation satellites, along with linkage to traditional Earth-based assets, such as the Deep-Space Network. The total system will effectively extend Earth-based nodes to Mars, initiating an interplanetary Internet that will bring planetary exploration right into our homes. The baseline architectural system design is presented, as derived from evolving mission and program requirements. Focus is on the orbital infrastructure, considering effects of Mars-orbit design trades on telecommunications and navigation performance. Launch, near-Earth, interplanetary and Mars orbit insertion phases are briefly treated.

99-301 **Mars Network Constellation Design Drivers and Strategies: A Navigation Perspective** –
0850am P. Kallemeyn, T. Ely, J. Guinn, M. Jah, Y.E. Bar-Sever, S. Lichten, S-C. Wu, L. Romans, D.J. Bell, E. Levene

NASA has asked the Jet Propulsion Laboratory to design a spacecraft constellation that will provide communication relay, and navigation support for a variety of future Mars missions. The objective of this constellation is to provide increased data return, enable autonomous onboard navigation with reduced reliance upon Earth-based tracking data, and substantially lower the combined operations costs of anticipated missions for Mars explorations. This studys efforts are guided by anticipated user requirements, candidate Mars scenarios, and the desire to supply an evolving, enabling navigation/communication infrastructure for future unspecified missions. Navigation and communication requirements, drivers, and metrics are presented and discussed.

99-302 **Mars Polar Lander Entry Descent & Landing Design** –W.H. Willcockson
0910am

This paper addresses the overall entry descent and landing (EDL) design of the Mars '98 Lander vehicle (now Mars Polar Lander) with emphasis on the hypersonic and parachute phases. This includes the analysis basis for the trajectory, aeroshell aerodynamics, aerothermal, TPS, and parachute. A more limited treatment of the terminal descent design is included. The end-to-end EDL performance, including the terminal descent phase, is described.

99-303 **The Strategy for the Second Phase of Aerobraking Mars Global Surveyor -**
0930am M.D. Johnston, P.B. Esposito, V. Alwar, S.W. Demcak, E.J. Graat,. P.D. Burkhart, B.M. Portock

On February 19, 1999, the Mars Global Surveyor (MGS) spacecraft was able to propulsively establish its mapping orbit. This event followed the completion of an ‘extended’ orbit insertion phase that was characterized by two distinct periods of aerobraking. During the second period of aerobraking, called “Aerobraking Phase 2,” the orbit period of the spacecraft was reduced from 11.6 hours to 2 hours in just over four months. This paper focuses on and describes the strategy developed for the second phase of aerobraking MGS. This description includes the baseline aerobraking flight profile, the trajectory control methodology, and the key trajectory metrics that were monitored in order to successfully ‘guide’ the spacecraft to its desired mapping orbit. Additionally, the planned aerobraking flight profile is compared to the actual aerobraking (trajectory) results.

0950am -1010am Coffee Break

99-304 **Aladdin's Phasing and Mission Orbit Design** –Y. Guo
1010am

As one of the finalists proposed under NASA’s 1998 Discovery Program, the Aladdin mission’s objective is to explore and return samples from the two Martian moons, Phobos and Deimos. The innovative phasing orbit transfers the Mars arrival state to the departure state with great Δv savings and makes it possible to use the Delta II type launch vehicle. Nineteen mission orbits provide ample opportunities for flybys at both moons for remote sensing, navigation, and sampling. The phasing and mission orbit design is presented in the paper.

99-305 **Mission Design Overview for the 2003/2005 Mars Sample Return Mission** –W.J. Lee,
1030am L.A. D'Amario, R.B. Roncoli, J.C. Smith

In May 2003, a new and exciting chapter in Mars exploration will begin with the launch of the first of three spacecraft that will collectively achieve the objective of delivering samples from the Red Planet to Earth. This project is called Mars Sample Return (MSR) and will utilize launches in both 2003 and 2005 with a sample return in October 2008. The baseline mission mode selected for MSR is Mars orbit rendezvous, analogous in concept to the lunar orbit rendezvous mode used for Apollo. Specifically, MSR will employ two landers of nearly identical design and one orbiter carrying a payload of rendezvous sensors, orbital capture mechanisms, and an Earth return capsule. The high-level concept is that the landers will launch surface samples into Mars

orbit, and the orbiter will retrieve the samples in orbit and then carry them back to Earth. This paper will provide an overview of the preliminary mission design for MSR.

99-306 **Mars Orbit Rendezvous Strategy for the Mars 2003/2005 Sample Return Mission** –L.A.
1050am D'Amario, W.E. Bollman, W.J. Lee, R.B. Roncoli, J.C. Smith

The baseline scenario for the Mars 2003/2005 Sample Return Mission utilizes Mars orbit rendezvous to retrieve two orbiting sample canisters containing Martian surface material. The two canisters, placed into Mars orbit by Landers launched in 2003 and 2005, will be retrieved by an Orbiter launched in 2005 and returned to Earth in 2008. Rendezvous operations last for approximately one year and are divided into three phases: the preliminary (search) phase, the intermediate (orbit matching) phase, and the terminal (proximity operations and capture) phase. During intermediate rendezvous, nodal phasing orbits are used to minimize the Δv required to align the orbit planes of the Orbiter and the sample canister. Preliminary analyses have shown that the total Δv required for intermediate rendezvous (to retrieve two sample canisters) is about 480 m/s (99% probability).

99-307 **Terminal Rendezvous Analysis and Design for the Mars Sample Return Mission** –
1110am P.S. Kachmar, C.N. D'Souza, T.J. Brand

The guidance, navigation and control design for the terminal phase of the Mars 2003/2005 Sample Return Mission will be discussed. The terminal phase is defined to be when the range between the Mars Sample Return Orbiter and the Orbiting Sample Canister is less than 2 km. The sensors will include a Radio Direction Finder, a laser radar and an IMU. The present rendezvous strategy of a co-elliptic approach to the leading V-bar, with a stationkeeping period at 80 m from the sample canister, will be presented.

SESSION 2 *8:30 AM – 11:30 AM*

Session Room: Columbia Ballroom
Session Title: Formation Flying/Constellations
Session Chair: Jay W. Middour
Naval Research Laboratory
Code 8103
4555 Overlook Avenue, S.W.
Washington, DC 26375
Tel: (202) 767-6528
Fax: (202) 404-7516
Email: middour@nrl.navy.mil

99-308 **Guidance Strategies for Satellite Formations** - G.B. Palmerini
0830am

The common aspect of all the multi-platform missions is the need for a correct relative positioning of the satellites. Performing guidance strategies are required to identify a configuration which is suitable for the mission concept. Once this target configuration has been chosen, the problem to identify capable control laws arises. An autonomous on-board control has to be pursued, to reduce the need for a heavy ground-segment. Moreover, the different nature and amount of perturbations to be counter-acted at the relevant altitudes is a point to be focused. A simple model to deal with this problem is presented.

99-309 **Dynamics of Clustered Satellites via Orbital Elements** –D.F. Chichka
0850am

This paper considers the problem of creating a cluster of satellites that maintains a constant apparent distribution as seen from the surface of the planet. Using the linearized equations of relative motion about a central point of the cluster, and assuming that the bulk motion of the cluster lies on a circular orbit, the possibility of creating such a cluster is easily shown. These equations are subject to inaccuracies due to nonlinear terms, however. This paper presents estimates of these errors in terms of the nominal eccentricity of the orbits of the clustered satellites.

99-310 **Spacecraft Formation Flying Control Using Mean Orbit Elements** –H. Schaub,
0910am S.R. Vadali, K.T. Alfriend

To establish J2 invariant relative orbits, it is convenient to describe the relative orbit geometry in terms of specific differences in mean orbit elements. A control law is presented which feeds back errors in mean orbit elements, instead of the traditional position and velocity error vectors. Gauss' variational equations are used to model the relationship between the control and the orbit element rates. The advantage here is that essentially only the orbit elements are corrected which differ from the desired values. This makes it easier to predict the transient flight path and avoid collisions.

99-311 **Elliptical Sun-Synchronous Orbits with Line of Apsides Lying In or Near the Equatorial**
0930am **Plane** –J.E. Draim, P.J. Cefola, R.J. Proulx, D.R. Larsen, G.R. Granholm

This paper explores the characteristics of retrograde, sun-synchronous elliptic orbits with line of apsides lying in or near the equatorial plane. Coverage plots for a five-satellite ring showing the number of satellites in view and elevation angle data versus latitude and local time are presented. Stability of the orbit is discussed. Also analyzed is the effect on these orbits of the trapped radiation field environment (Van Allen Belts), as well as the exposure to damage by natural and man-made debris. A major advantage seen for these orbits is that they can be used to provide augmented daytime coverage for virtually any location on earth.

0950am -1010am Coffee Break

99-312 **Characteristics of Crosslinks Between Satellites in Large Symmetric Constellations** –
1010am T.J. Lang

In large constellations of satellites, it is often important to provide communication links between the satellites. These crosslinks require that there be a continuous unobstructed line of sight between the linked satellites. Furthermore, it is desired to select crosslink partners such that gimbal angles and rates on the crosslink antennas are kept within reasonable bounds. It may also be attractive to select crosslink partners so that all satellites in the constellation are connected in a single communication loop. This paper examines the availability and characteristics of crosslinks in large symmetric (Walker-type) constellations. Equations are developed which allow the determination of acceptable crosslink partners and describe the gimbal angle requirements to complete the link.

99-313 **Design and Analysis of Space Station-Based Micro-Satellite Networks** –J.Yuan, Q. Fang,
1030am J. Wei

Space station as one of the permanent manned space platforms, can implement many types of space missions. Some of which have been proved by MIR and former other space stations, some are going to be carried out by International Space Station (ISS) and others are to be developed by scientists. Micro-Satellite network, due to its global coverage, can be used for navigation, communications, surveillance, observation, etc. On the other hand, because of its simple construction and low cost of the satellite, are fit for batch processing and netting. This will lead to a limitation to the satellite: limited on board power supplies, space-to-ground transmission abilities and store abilities. The combination of space station with satellite network will open a new field of space applications. The mission of the space station is analyzed when put into a network of micro-satellite, including data receiving from satellite and pre-processing, data relay, satellite management and control, protection, etc. Some design and analysis have been implemented in this paper, including: Relative motion description of space station and sat-net, Visibility and data receiving window analysis of space station to sat-net, data transmission design and analysis. Iridium System and ISS are taken as examples, analysis and simulation are implemented.

99-314 **Using Tessellations for Large Constellation Design** –B. Kantsiper, H. Drake
1050am

Designing large constellations of satellites is a time-consuming process, typically requiring slow numerical analysis in order to ensure coverage requirements are met. This paper presents an alternative geometric approach based on tessellations of spherical polygons. This approach allows the analyst to determine much more rapidly which constellations bear further examination for a given satellite altitude. It may be applied both for global coverage and when a restricted latitude band is the region of interest. The method is evaluated by comparing its results to the correct answers determining from searching a database of global coverage constellations. The method yields optimal results for large constellations, but fails for small numbers of satellites due to distortions in the polygon geometry.

99-315 **The Multi-Synchronous Technique for Environment Monitoring by a Satellite**
1110am **Constellation** –C. Ulivieri

Standard remote sensing satellite are rarely able to provide timely and update information for the environment monitoring; in fact, most of the existing platforms are optimized for specific applications influencing temporal and spatial requirements and sensor characteristics. In this paper, the selection criteria for optimum orbital characteristics are presented, taking into account constraints and requirements in disaster and environment monitoring. In particular, the paper investigates the nature of the dependence of the satellite trace repetition factor (number of orbits per day) upon the shape of the orbit. The analysis is extended to the study of constellations consisting of satellites deployed on similar multi-synchronous orbits over the same, or equally spaced, planes.

SESSION 3 *8:30 AM – 11:50 AM*

Session Room: Columbia Ballroom
Session Title: Attitude Control: Theory
Session Chair: Peter Bainum
Howard University
Department of Mechanical Engineering
2300 6th Street, N.W.
Washington, DC 20059
Tel: (202) 806-6612
Fax: (202) 806-5258
Email: pbainum@fac.howard.edu

99-316 **Vibration Reduction for Flexible Spacecraft Following Momentum Dumping**
0830am **with/without Slewing -** A.K. Banerjee, N. Pedreiro, W.E. Singhose

Vibration reduction following momentum dumping by on-off thrusters on a flexible spacecraft has been demonstrated by a method that solves for the thruster switching times in an optimization problem. When slewing is done together with momentum dumping, it is shown that advantage can be taken of the latter to reduce slewing time while ensuring vibration suppression.

99-317 **Satellite Attitude Control and Power Tracking with Momentum Wheels** –H. Shen,
0850am P. Tsiotras

In a previous paper, control laws were developed for a rigid spacecraft to track desired maneuvers using both thrusters and momentum wheels. The model studied comprised of a rigid body with external thrusters and N rigid axisymmetric momentum wheels controlled by axial torques. The Modified Rodrigues parameters (MRPs) were used to describe the kinematics. The thruster torques and the axial motor torques were the controls used to track given attitude motions, using the angular velocity and MRP errors to develop linear and nonlinear control laws. Since three momentum wheels suffice to provide attitude torques to the spacecraft, in this work, we consider four non-colinear momentum wheels, and present a control law that uses the extra degree of freedom to track a desired power profile. This control law can be incorporated in an Integrated Power/Attitude Control System (IPACS) on-board a satellite. The possibility of occurrence of singularities, where no arbitrary energy profile can be tracked, is studied with a specific configuration of a four-momentum-wheel cluster. The ensuing analysis provides conditions for singularity avoidance.

99-318 **Mass Expulsion Control for Precision Pointing Spacecraft** - H.D. Stevens, J.J. Rodden,
0910am S. Carrou

The Gravity Recovery and Climate Experiment (GRACE) mission will produce a new model of the Earth's gravity field with unprecedented accuracy every 15-30 days for a period of 5 years. The GRACE mission involves flying two satellites in a tandem formation spaced 100 to 500 km apart. The mission uses two co-orbiting satellites which are themselves the instrument. The Attitude and Orbit Control System design is constrained by both the mission cost cap and the novel requirements imposed by using the satellites as the science instrument. The AOCS uses star tracker attitude measurements to precisely point the two spacecraft at each other using cold gas thrusters and magnetic torque rods. The attitude and noise spectra of the attitude variations are tightly controlled. This paper discusses the inter-relationship between the science objectives and the requirements on the AOCS. The AOCS design will be discussed, highlighting the innovations required to meet the science objectives.

99-319 **Controller Design for Space Platform-Based Variable Geometry Manipulators** - Y. Cao,
0930am V.J. Modi, C.W. deSilva, A.K. Misra

Control of an elastic space platform-based flexible manipulator with four links, two free to slew while the other two permitted to deploy, is studied using two procedures: (1) nonlinear Feedback Linearization Technique (FLT) applied to rigid degrees of freedom with flexible generalized coordinates passively regulated through coupling; () rigid as well as flexible degrees of freedom controlled through FLL and Linear Quadratic Regulator (LQR), respectively. Results suggest the FLT control to be quite effective even for flexible degrees of freedom. The combination of FLT and LQR further improves the controller performance.

0950am - 1010am Coffee Break

99-320 **Initial Guess Schemes for Generating Deflection-Limiting Commands** –M. Robertson,
1010am W.E. Singhose, A.K. Banerjee

This paper describes a command generation algorithm for flexible spacecraft with both elimination of residual oscillations and suppression of the deflection during the move. A numerical optimization is used to obtain the profile. This method produces initial guesses that insure convergence to the time-optimal, or nearly time-optimal, command. Simulation results are presented for a benchmark system.

99-321 **Sensitivity of Spin-Axis Reorientation and Delta-V Maneuvers** –S. Tanygin,
1030am J.W. Woodburn

A method for evaluating spin-axis deviations during finite burns is presented. It can be applied repeatedly when studying rhumb-line and other multiple burn maneuvers. The method generates estimates of both angular rate and angular displacement of the spin-axis. A technique for minimizing such deviations is then derived using jet start time adjustment in particular.

99-322 **Thermally-Induced Attitude Vibration Control of a Spacecraft with a Flexible Boom**
1050am **Using Smart Structures** - S-Y. Park

This paper presents an advanced method of attitude dynamics and control for a spacecraft with a flexible boom suffering from thermally-induced vibrations. The analysis for spacecraft model is conducted by a coupled system composed of a flexible appendage and a rigid main body. The active control system design for vibration suppression uses smart structures and closed-loop control methods. With an embedded piezoelectric device, a modified technique of Positive Position Feedback makes it attractive for thermally-induced attitude vibration control of space structures.

99-323 **Feasible Trajectory Generation for Underactuated Spacecraft Using Differential Flatness**
1110am –P. Tsiotras

We consider the problem of feasible trajectory generation of an underactuated axisymmetric spacecraft subject to two external torques acting on the plane normal to the symmetry axis. We derive the conditions that must be satisfied by an attitude history in order to be a feasible trajectory. We then propose a methodology to generate trajectories satisfying these conditions. Our approach makes use of the well-known flatness of the corresponding differential equations. We emphasize the importance of being able to generate these trajectories on-line and with minimal off-line intervention. Feasible trajectories can later be used as reference trajectories for tracking problems for an underactuated spacecraft.

99-324 **Exponential Stabilization of Spacecraft's Angular Velocity with Underactuated**
1130am **Configuration of One-Sided Thrusters** –B. Zhang, H. Wu, H. Zhang

This paper discusses the problem of detumbling a rigid spacecraft with underactuated configuration of one-sided thruster. First, some new definitions on the configuration of one-sided thruster are introduced, such as completeness and incompleteness, and the incomplete configuration is divided into two cases: disturbance-free two-dimensional complete configuration for three-dimensional control and two-dimensional complete configuration with disturbance for three-dimensional control. For the latter one a special case as disturbance-coplaner two-dimensional complete configuration is considered to stabilize the angular velocities of a rigid spacecraft, in which a Sliding Mode Control (SMC) law is used. The simulations show the effectiveness of the approach.

SESSION 4 *1:30 PM – 4:10 PM*

Session Room: Columbia Ballroom
Session Title: Orbit Determination: Missions
Session Chair: Paul Cefola
Charles Stark Draper Laboratory
Mail Stop 86
555 Technology Square
Cambridge, MA 02139
Tel: (617) 258-1787
Fax: (617) 258-1880
Email: cefola@draper.com

99-325 0130pm **Lunar Prospector Orbit Determination Uncertainties Using the High-Resolution Lunar Gravity Models** – E. Carranza, A. Konopliv, M. Ryne

Orbit determination of the Lunar Prospector spacecraft performed at the Jet Propulsion Laboratory was conducted as part of the principal science investigation of the lunar gravity field. The orbit determination was performed using S-band two-way Doppler and range data, collected from Deep-Space Network tracking stations at a near continuous high-rate. A description of the mission and its trajectory will be provided, followed by a discussion of the orbit determination estimation procedure and models. Accuracies will be examined in terms of orbit-to-orbit solution differences, as a function of gravity model truncation and inclination in the plane-of-sky; the orbit determination accuracies are determined for the latest 100th degree gravity model (LP100J) for both the nominal and extended mission. Long-term predictions for several gravity fields will be compared to the reconstructed orbits to demonstrate the accuracy of the orbit determination and gravity models.

99-326 0150pm **Orbit Determination for Medium Altitude Orbits Using GPS Receivers and Ground -Based Tracking** – G.R. Granholm, R.J. Proulx, P.J. Cefola

For certain types of medium-altitude or highly-eccentric orbits, it is not always clear if measurements should be taken from the ground or in space. This paper analyzes orbit determination for the Ellipso™. Borealis sub-constellation using high-precision computer simulations of conventional ground-based tracking and on-board GPSR tracking. OD is performed in the presence of various errors, such as atmospheric and gravitational mismodeling, observation noise and biases, ionospheric distortion, and other GPS-associated errors. The sensitivity of solution accuracy to atmospheric conditions is investigated. Numerical results in the form of fit and prediction errors are tabulated and presented graphically.

99-327 **Orbit Determination Using Space-to-Ground Differential GPS in NRL's OCEAN**
0210pm **Package** –P.W. Binning, J. Middour, M. Soyka

This paper will detail results of the Differential GPS (DGPS) development in NRL's orbit determination package, OCEAN. The algorithms and methodology will be detailed as will the actual operations of the software modules. Results will be presented from a number of test cases using data from the TOPEX satellite and ORPHEUS SPAS free flyer. Results are going to be shown from the OCEAN filter/smoother as well as the OCEAN Batch processor. Comparisons between ephemerides from both will be made.

99-328 **Orbit Determination for Mars Global Surveyor during Mapping** –F.G. Lemoine,
0230pm D.D. Rowlands, D.E. Smith, D.E. Pavlia, D.S. Chinn, S.G. Luthcke, G.A. Neumann, M.T. Zuber

The Mars Global Surveyor (MGS) spacecraft reached a low-altitude circular orbit on February 4, 1999, after the termination of the second phase of aerobraking. The MGS spacecraft carries the Mars Orbiter Laser Altimeter (MOLA) whose primary goal is to derive a global, geodetically referenced 0.2° x 0.2° topographic grid of Mars with a vertical accuracy of better than 30 meters. During the interim science orbits in the Hiatus mission phase (October - November 1997) and the Science Phasing Orbits (March - April 19998, and June - July 1998), 208 passes of altimeter data were collected by the MOLA instrument. On March 1, 1999, the first ten orbits of MOLA altimeter data from the near-circular orbit were successfully returned from MGS by the Deep-Space Network (DSN). Data will be collected from MOLA throughout the Mapping phase of the MGS mission, or for at least one Mars year (687 days). Whereas the interim orbits of Hiatus and SPO were highly eccentric, and altimeter data were only collected near periapsis when the spacecraft was below 785 km, the Mapping orbit of MGS is near circular, and altimeter data will be collected continuously at a rate of 10 Hz. The proper analysis of the altimeter data requires that the orbit of the MGS spacecraft be known to an accuracy comparable to that of the quality of the altimeter data. The altimeter has an ultimate precision of 30 cm on mostly flat surfaces, so ideally the orbits of the MGS spacecraft should be known to this level. This is a stringent requirement, and more realistic goals of orbit error for MGS are ten to thirty meters. In this paper, we will discuss the force and measurement modeling required to achieve this objective. Issues in force modeling include the proper modeling of the gravity field of Mars, and the modeling of non-conservatives forces, including the development of a "macro-model," in a similar fashion to TOPEX/POSEIDON and TDRSS. During Cruise and Aerobraking, the high gain antenna (HGA) was stowed on the +X face of the spacecraft. On March 29, 1999, the HGS will be deployed on a meter long boom that will remain Earth-pointing while the instrument panel (including the MOLA instrument) remains pointed at nadir. The tracking data must be corrected for the regular motion of the high gain antenna with respect to the center of mass, and the success of the MGS determination during Mapping will depend on correctly accounting for this offset in the measurement model.

0250pm - 0310pm Coffee Break

99-329 **Nozomi Cis-Lunar Phase Orbit Determination** –M. Ryne, K. Criddle
0310pm

The Nozomi spacecraft, also known as Planet-B, is the first to achieve an Earth-Mars transfer orbit without a direct hyperbolic insertion maneuver. The trajectory is accomplished by use of multiple lunar gravity assists and a powered Earth flyby. This paper describes the Jet Propulsion Laboratory's orbit determination effort in support of the mission. Stringent swingby delivery requirements were achieved despite poor tracking data information content, due to factors such as large spacecraft range, small absolute velocity and low declination. Also described is the near real time assessment of the post Earth flyby orbit that enabled the design and successful execution of a critical trajectory correction maneuver.

99-330 **A Comprehensive Orbit Reconstruction for the Galileo Prime Mission in the J2000**
0330pm **System** –R.A. Jacobson, R.J. Haw, T.P. McElrath, P.G. Antreasian

The Galileo spacecraft began its orbital tour of the Jovian system in December 1995 and completed its 11 orbit prime mission in November 1997 having had 16 successful close encounters with the Galilean satellites. Earlier papers discussed the determination of the spacecraft orbit in support of mission operations from arrival at Jupiter through the first 9 orbits. In this paper, we re-examine those earlier orbits and extend the analysis through orbit 12, the first orbit of the Galileo Europa Mission. The objective of our work is the reconstruction of the spacecraft trajectory together with the development of a consistent set of ephemerides for the Galilean satellites.

99-331 **Adaptation of a Spaceborne Geolocation System to an Airborne Experiment** –H. Pickard,
0350pm A. Hope

The Naval Research Laboratory (NRL) conducted an experiment on an airship with a TV Camera and the previously space flown HERCULES geolocation system. The HERCULES system contains a ring laser gyro (RLG) which provides accurate attitude determination of the camera platform. The adaptation of the initialization of the gyro is discussed. The position of the airship was determined using Global Positioning System (GPS) data. A comparison of the real-time versus the post-processed high accuracy differential positions is presented. The position and attitude data, combined with an Earth model, provides a geolocation solution. Earth model selection and its effects are discussed. Finally, several still frame images from the HDTV are geolocated and compared to truth reference points that are located in the frame.

SESSION 5 *1:30 PM – 4:50 PM*

Session Room: Columbia Ballroom
Session Title: Solar Sails and Low Thrust
Session Chair: Craig Kluever
University of Missouri-Columbia/Kansas City Campus
Mechanical and Aerospace Engineering
5605 Troost Avenue
Kansas City, MO 64110
Tel: (816) 235-1278
Fax: (816) 235-1260
Email: klueverc@umkc.edu

99-332 0130pm **Near-Optimal Solar Sail Trajectories Generated by a Genetic Algorithm** –G. Rauwolf, A. Friedlander

This paper describes a method for efficiently generating near-optimal Solar Sail trajectories using a Genetic Algorithm. This method identifies near-optimal solutions for several challenging missions and is capable of identifying Pareto solution fronts that allow the mission designer to examine the potential solution space clearly and quickly. Results will be presented for a range of missions and at least one optimal trajectory will be determined, by traditional methods, from the near-optimal solution.

99-333 0150pm **A Technique for Earth Escape Using a Solar Sail** –V. Coverstone-Carroll, J.E. Prussing

The feasibility of escaping Earth orbit using a solar sail is investigated. Starting in geosynchronous transfer orbit, a small solar sail force per unit mass (less than 1 mm/s/s) can be used to gradually accelerate the spacecraft over several months until escape energy is attained. Optimal orientation of the sail at each point in the orbit is required to achieve the desired performance for this three-dimensional problem.

99-334 0210pm **Solar Sail Optimal Orbit Transfers to Synchronous Orbits** –R.B. Powers,V. Coverstone-Carroll, J.E. Prussing

A constant outward radial thrust acceleration can be used to reduce the radius of a circular orbit of specified period. Heliocentric circular orbits are designed to match the orbital period of Earth or Mars for various radial thrust accelerations and are defined as synchronous orbits. Minimum-time solar sail orbit transfers to these synchronous heliocentric orbits are presented.

99-335 **Stability of Levitated Cylindrical Orbits by Using Solar Sails** – C. van de Kolk
0230pm

This paper discusses the feasibility of putting satellites in levitated orbits by using solar sails. In order to stay in such an orbit, a force must be generated which cancels out the vertical component of the gravitational force. It has been suggested to use a solar sail to generate this force, and over the past years several papers dealing with this concept have been published, some stating it is possible, others stating it is not possible. In this paper, it will be shown that it is indeed not possible to use solar sails for light-levitated cylindrical orbits, due to neglect of certain terms in the force balance.

0250pm - 0310pm Coffee Break

99-336 **Solar Sail Trajectories for Solar Polar and Interstellar Probe Missions** – C.G. Sauer, Jr.
0310pm

Solar Sail trajectories for two high energy *Space Physics* missions, Solar Polar and Interstellar Probe, are examined in this paper. An ideal, flat-perfectly reflecting sail together with optimal thrust steering is assumed in the trajectory synthesis. Parametric data is presented for both missions as a function of both sail characteristic acceleration and minimum solar distance. The purpose of the Solar Polar Mission is to place a payload into a short period, polar orbit about the Sun. Final circular solar orbits, ranging from 0.3 AU to 1 AU, were investigated for a range of sail characteristic accelerations from .4 to 1.5 mm/s^2. The Solar Sail Interstellar Probe Mission is designed to place the spacecraft on a heliocentric escape trajectory that will reach 100 to 200 AU in 10 to 20 years in the direction of the solar apex. The above requirement dictates trajectories that are much more energetic than any that have previously been flown including the Solar Polar Mission above. As a consequence, higher sail characteristic accelerations in the range of 1 to 10 mm/s^2, are required to satisfy the mission requirements.

99-337 **Analytic Representations of Low-Thrust Trajectories for Gravity-Assist Applications** –
0330pm A.E. Petropoulos, J.M. Longuski, N.X. Vinh

Given the benefits of coupling low-thrust propulsion with gravity assists, techniques for easily identifying candidate trajectories would be extremely useful to mission designers. Rather than directly addressing the optimization problem, we develop analytic and semi-analytic techniques for identifying candidate low-thrust, gravity-assist trajectories. We assume either a trajectory shape (and derive associated thrust profiles and velocities) or a thrust profile (and derive the associated trajectory shape and velocity). Application of these techniques provide similar results to optimal trajectories reported in the literature. Such easily computed trajectories serve as initial guesses in optimization and for quick overviews of a broad launch-date space.

99-338 **Preliminary Design of Low-Thrust Interplanetary Missions** –J.A. Sims, S.N. Flanagan
0350pm

A direct optimization method intended to be used primarily for preliminary design of low-thrust interplanetary trajectories, including those with multiple gravity assists, is presented. Results from several different types of trajectories are compared to those from a low-thrust trajectory optimization program using an indirect method. The results from the two programs agree very closely. The new method has shown less convergence sensitivity and the ability to handle more intermediate flybys than the indirect method.

99-339 **Trajectories towards Near-Earth-Objects Using Solar Electric Propulsion** –G. Colasurdo,
0410pm L. Casalino

Electric propulsion (EP) will probably be the preferred option for space transportation in the near future, due to the mass and coast savings that can be provided by higher specific impulse compared to chemical propulsion. The authors use an indirect optimization method which has been developed at the ``Politecnico di Torino'' for the optimization of finite-thrust trajectories. It is based on the assumption of the switching structure; a procedure based on Newton's method solves the boundary value problem that arises from the application of the optimal control theory. Pontryagin's Maximum Principle is used during the analysis to assess the optimal switching structure of the trajectory, which is composed of a succession of propelled and coast arcs. The possibility of obtaining gravity assist from a planet is also considered; the necessary optimum conditions are presented. One of the most interesting aspects of the analysis is the use of a 1:1 DV-EGA trajectory to reach a near Earth object.

99-340 **Solar Electric Propulsion Earth to Mercury Optimal Trajectories with Variable**
0430pm **Maximum Power** –M.Guelman, D. Tahan

Optimal trajectories are developed for the case of variable maximum power. An analysis of the relation between the optimal cost and the terminal time is performed. Cost is shown to be a monotonous decreasing function of transfer time and for specific terminal times cost has a jump reduction, together with a jump increase of the transfer angle. Based on these results, the case of full state constrained Earth to Mercury transfer for fixed terminal time when the anomaly transfer angle is constrained to discrete values was solved. It was found that there is a specific value of the anomaly angle which leads to a minimum cost.

SESSION 6 ***1:30 PM – 3:50 PM***

Session Room: Columbia Ballroom
Session Title: Topics in Control Theory
Session Chair: Thomas E. Carter
Math/CSC Department
Eastern Connecticut State University
83 Windham Street
Willimantic, CT 06226
Tel: (860) 465-5257
Fax: (860)
Email: cartert@ecsuc.ctstateu.edu

99-341 **Reducing Conservatism of Analytic Transient Response Bounds via Shaping Filters** –
0130pm J. Watson, K. Grigoriadis, J-W. Jang, N. Bedrossian

Recent results show that the peak transient response of a linear system to bounded energy inputs can be computed using the energy to peak gain of the system. However, analytically-computed peak response bound can be conservative for a class of bounded energy signals, specifically pulse trains generated from jet firings encountered in space vehicles. In this paper, shaping filters are proposed as a methodology to reduce the conservatism of peak response analytic bounds.

99-342 **Disturbance Identification and Rejection Experiments on an Ultra Quiet Platform** -
0150pm S.G. Edwards, B.N. Agrawal, M.Q. Phan, R.W. Longman

Vibration isolation on spacecraft is needed for imaging sensors, microgravity experiments, and other sensitive payloads. Thus far, passive isolation methods are preferred for simplicity and low cost. Active vibration isolation and disturbance rejection will soon be more common as space qualified sensors, actuators and processors become more capable and affordable, and performance requirements increase. Spacecraft disturbances are typically periodic vibrations that are most effectively controlled through feedforward techniques. A popular choice of feedforward control for disturbance rejection is the Filtered-x LMS algorithm that requires a measured disturbance-correlated signal in its implementation. A new technique called "Clear Box" takes a system identification approach to control. It allows operation in an information-rich environment with built-in fault tolerance, the ability to control harmonics as easily as fundamental modes, and the ability to select which modes to when actuator saturation is an issue, all without the need for a measured disturbance-correlated signal. Experiments are performed on a testbed with a Stewart platform employing ultra sensitive geophone sensors, and designed for testing of vibration isolation techniques. These experiments use both the Filtered-x LMS and Clear Box techniques, and provide an effective demonstration of the advantages of the Clear Box method.

99-343 **The Optimized Interaction Forces in Minimum Time Control of Elbow Robots** –G. Giese,
0210pm R.W. Longman, H.G. Bock

In research reported at the recent AAS/AIAA Spacecraft Mechanics Conference, time optimal control was studied for polar coordinate robots. The present paper is a companion work that develops understanding of the nature of time optimal maneuvers for the much more common elbow robot configuration. Physical interpretations of the time optimal maneuvers are given. Typically one axis of a robot has the most difficult task to perform in getting from its starting point to its desired endpoint. In so far as other axes have extra time to get to their endpoints, they can use this freedom to help the first axis by using interaction forces such as gravity, centrifugal force, Coriolis and angular acceleration related forces, and equal action and reaction effects. The overall objective of the research effort is leading to speeding up industrial robots, allowing one to operate assembly lines faster, result in increased productivity.

99-344 **Subtleties in the Use of Zero-Phase Low-Pass Filtering and Cliff Filtering in Learning**
0230pm **Control** –A.M. Plotnik, R.W. Longman

Learning control laws adjust the command given to a feedback control system in order to converge on a command that approaches zero tracking error without redesigning the feedback controller itself. Successful learning laws that use low-pass filtering to eliminate frequencies that destabilize the learning process have been developed and tested. Since the learning process is performed off-line between repetitions, it can utilize zero-phase filtering techniques, which are not practical in real-time applications. In this paper, we use simulations to explore the use of two such methods of low-pass filtering, zero-phase IIR filtering and what we choose to call cliff filtering. Cliff filtering is an approach in which undesirable high-frequency components of a signal are entirely eliminated in the discrete frequency domain. With either technique, important subtleties are found that affect the stability and effectiveness of the learning process.

0250pm - 0310pm Coffee Break

99-345 **Spaceflight Dynamics Support for the MightySat II.1 Hyperspectral Payload Operations**
0310pm –S. Carter, Dr. C.A. McLaughlin, Dr. C. Sabol, M. Bir

The MightySat II.1 spacecraft is a low-cost technology testbed for space-based experiments sponsored by the Space Vehicles Directorate of the Air Force Research Laboratory. Potentially, the first hyperspectral instrument in space, the Fourier Transform Hyperspectral Imager (FTHSI) is one of seven experiments manifested on MightySat II.1 and will provide AFRL with an opportunity to collect and exploit high spectral resolution information for the primary purpose of terrain categorization. This paper discusses the operations architecture developed for MightySat II.1 FTHSI flight dynamics support, specifically focusing on the unique software contributions in the areas of orbit determination and payload mission scheduling. Integration of the Draper Semianalytical Satellite Theory (DSST) and MightySat II.1 spacecraft dynamical models into Satellite Toolkit (STK) are specifically addressed.

99-346 **Development of a Bapta Mechanism for Small Satellites** –M.C. Ricci, S.C. Varotto
0330pm

This work intends to show some aspects about the mechanical layout and modeling of a mechanism in development involving a government institution (INPE) and private companies. Some kinds of artificial satellites must be appointed toward the Earth while the solar panels must be appointed toward the Sun in order to maximize the solar energy that reaches the solar panels. These kinds of satellites shall incorporate a BAPTA (Bearing and Power Transfer Assembly) mechanism. The BAPTA is a Power Subsystem mechanism that performs the tasks of maintaining the one degree-of-freedom of the output axis (panel axis) while transmitting the electrical and power signals from the panel to the satellite body. The BAPTA consists basically of three main units: (1) the Bearing Unit, (2) the Drive Unit and (3) the Slip Rings Unit. The design here described is being evolved for application in small satellites (~0.5kW with rigid solar panels of ~1kgm2). In this work, we concentrate the efforts in the modeling of the Drive Unit. Two solutions were considered for the Drive Unit: (1) closed-loop synchronous system and (2) open-loop incremental motion control system. Since the orbital period is too large, (on the order of 100 min) the nominal speed and its variation are very low and very difficult to measure. Therefore, the second solution is to be the better choice. The selected concept uses a conventional 1.8 degree stepper motor and a 100:1 harmonic drive reduction gearing to achieve a theoretical output step size of 0.018 degrees. Simulations in computer are showing that significant interaction with the solar array and spacecraft are avoided (spacecraft pitch disturbing velocity is lower than 0.001 deg/sec) with power consumption as low as 3 to 5 W. The adequate determination in the power consumption is possible due to an adequate representation of magnetic non-linearity in the motor, including its effect on electromagnetic torque production. The reliable prediction of the important dynamic characteristics of the motor and general system necessitates a precise representation of the flux-linkage data. A good example is the single-step damping, which can be predicted satisfactorily only if the current disturbance and associated power loss in each stator circuit are calculated correctly. This requires that the rate of change of flux-linkage be defined accurately for any combination of the system variables.

SESSION 7 *Tuesday, August 17, 1999*
8:30 AM – 10:50 AM

Session Room: Columbia Ballroom
Session Title: Launch, Mission Support Tools, and Standards
Session Chair: Lester L. Sackett
Charles Stark Draper Laboratory
555 Technology Square
Cambridge, MA 02139
Tel: (617) 258-2283
Fax: (617) 258-2555
Email: lsackett@draper.com

99-347 **Trajectory Development and Optimization of an RBCC-based Launch Vehicle** –
0830am K.J. Hack, J.P. Riehl

The development and optimization of the trajectory for a Bantam-class (300-pound payload) launch vehicle using rocket-based combined cycle (RBCC) engines is presented. In addition to a brief explanation of the operational characteristics of RBCC engines and a launch vehicle using such propulsion, this paper documents the trajectory design and performance optimization analysis approach, assumptions, constraints, and results. A discussion of the selection of controls, reference frames, and constraints is also included. In specific cases, a trade study was performed to examine how results changed as one or a combination of constraints were varied. For these trade studies, the effect of extended atmospheric flight is identified. Additionally, several simplifying assumptions are discussed which have been made to more easily show the capability of this propulsion technology. The impact of these assumptions on overall performance is discussed as well as methodology on their removal. The amount of mass delivered to specific orbit from a fixed initial mass will be the performance figure of merit for this examination.

99-348 **Aquarius, a Low-Cost, Low-Reliability Launch Vehicle to Carry Consumables** –
0850am A.E. Turner

This paper discusses a concept for a low-cost, low-reliability launch vehicle to ship consumables such as water, food, oxygen, nitrogen, propellants, as well as other nonperishable items of low intrinsic value such as spare parts to low-earth orbit. The goal is to reduce launch costs below $1,000 per kilogram using existing technology by taking advantage of the savings inherent in a launch vehicle overall reliability of 0.7 or less, and in the economy of scale provided by mass production of more than 1,000 units annually. Infrastructure needed to support Aquarius both at the launch site and in orbit are also discussed. Launch costs for spacecraft intended for geosynchronous orbit will be shown to be reducible by a factor of two or three depending upon the concepts and cost models used.

99-349 **Orbit Analysis Tools Software (Version 1 for Windows™)** –J. Cox, J. Middour,
0910am R. Llewellyn

The Orbit Analysis Tools Software (OATS) is a mission planning and analysis tool for Earth-orbiting satellites. OATS evolved from a collection of software tools developed by the Astrodynamics and Space Applications Office of the Naval Center for Space Technology located at the Naval Research Laboratory in Washington, DC. There have been three previous versions of OATS that are available to the public for use on a Macintosh computer. This release of the OATS program is a revised and expanded version written using Visual C++ for use on the Windows 95, 98, or Windows NT operating systems. The program's function is to perform satellite mission and coverage analysis using numerical and graphical techniques to analyze and display Earth coverage data and ground-to-satellite geometrical parameters.

99-350 **A Graphical Method for the Computation of Antenna Obscuration for the International**
0930am **Space Station** –J.W. Woodburn, D. Ohlarik

One of the operational challenges for the International Space Station (ISS) will be to determine the effect of self obscuration and obscuration by other orbiting objects on communication links between the ISS and the ground or to other spacecraft. A graphical solution to the problem of proximity obscuration has been developed using the OpenGL™ library of three dimensional graphics functions. The algorithm involves the construction of a full three dimensional scene from the perspective of the antenna and allows for modeling of time varying locations, orientations and articulations. Extensions of the current method to yield higher fidelity simulations are also discussed.

0950am - 1010am Coffee Break

99-351 **Collision Vision: Covariance Modeling and Intersection Detection for Spacecraft**
1010am **Situational Awareness** –R.G. Gist, D. Oltrogge

There is an accelerating need to achieve high levels of operability in space while at the same time minimizing the risk of collision between space vehicles and trackable resident space objects (RSOs), with special emphasis for the safety of manned orbiters. Previous efforts to ensure mission safety by establishing a conservative keep-out region around the satellite(s) of interest have the effect of either (1) needless expenditure of fuel for an on-orbit avoidance maneuver, (2) unacceptable reduction of launch availability, or (3) implicit acceptance of unknown collision risk due to lack of analysis capability or data. This conflict of needs has driven an effort to increase the fidelity of modeling of object position uncertainty for both on-orbit (or launching) spacecraft and the resident space object population, and using this improved knowledge to reasonably estimate the potential of the objects to collide. To achieve this fidelity, a suite of programs called "CollisionVision" has been developed and is currently being used to provide operational situational awareness. Covariance models are used which are generated by statistical perturbation of initial conditions. A dynamic covariance reconstruction method rapidly re-creates the maneuvering spacecraft's position dispersions at each point in the trajectory. For the resident space objects, CollisionVision makes use of dynamic ellipsoidal

position uncertainty caused by propagator error and error growth derived statistically by direct observations of RSOs. Error and error growth rate may be provided either by orbit class (derived from apogee/perigee height) or by individual RSO. The position uncertainties of the launcher and RSO are represented by ellipsoidal regions of 3-sigma probability uncertainty. Conjunction assessment software is then used to determine if these ellipsoids intersect using a novel mathematical approach. Once an intersection is detected, probability algorithms are used to statistically estimate the probability that the objects themselves actually will collide. A variety of models is available to compute the probability, including a NASA formulation, as well as internally derived methods. The results of a study will be provided in which booster/spacecraft re-contact risk was mitigated. A Monte Carlo method was used which sampled deviations in orbit perturbations, thrust uncertainty and attitude dispersions and provided maneuver information that essentially eliminated the collision threat. This paper will also examine the impact of position knowledge errors in RSO orbit determination and propagation on the ability to realistically predict collisions. A sample probability analysis will be included for a simulated geosynchronous orbit conjunction in which various orbit determination methods and accuracies are combined. This analysis demonstrates the value of having the best available data for all conjuncting vehicles, including active satellite ranging, when available. Dependence of computed conjunction probability on orbit determination accuracies of affected objects. This paper will contain details of covariance modeling and propagation of spacecraft and RSO, 3-σ probability bound overlap detection between spacecraft, and results of probability estimations based on such conjunctions. Also discussed will be the applicability of the system to a parallel-processing environment to facilitate other relative-motion studies, including monitoring of on-board sensor intrusion, radio frequency interference and launch radio frequency impingement.

99-352 **The Importance of Space Standards: An Astrodynamics Example** –F.A. Slane
1030am

The ascendancy of the commercial space market over the military market in 1996 marked the normalization of spaceflight; economic forces will quickly surpass the forces of military utility in shaping future space capability. This marks the achievement of a level of maturity of the global space industry. Streamlining will occur in all facets of the industry to maximize return on investment. Efforts of the United Nations and the AIAA described in this paper are aligned to help in the development of voluntary standards. Arguably, this is a linchpin activity to achieve the full potential of international space efforts. As an example, the potential impact of Astrodynamics Standards is reviewed in the areas of military and commercial operations and technical leadership. In this paper, the evolution of Astrodynamic Standards is discussed as Space has matured over the last 40 years. The challenge ahead is the development of broadly accepted standards with simple rules for application and the greatest available technical accuracy.

SESSION 8 *8:30 AM – 11:30 AM*

Session Room: Columbia Ballroom
Session Title: Interplanetary Missions
Session Chair: Louis A. D'Amario
MS 301-276
Jet Propulsion Laboratory
4800 Oak Grove Drive
Pasadena, CA 91109
Tel: (818) 354-3209
Fax: (818) 393-5214
Email: louis.damario@jpl.nasa.gov

99-354 **SpaceTime - A MIDEX Proposal to Test Einstein's Equivalence Principle** –S. Matousek
0830am

SpaceTime is a MIDEX class proposal. This paper describes the mission proposed for MIDEX 98. SpaceTime uses a Jupiter gravity-assist trajectory to send a spacecraft past the Sun at approximately 4 solar radii. This allows for the search of a violation of Einstein's Equivalence Principle by studying the differential redshift of 3 atomic clocks. This low-cost mission is made possible by the new application of technology (i.e., carbon-carbon heat shield) in conjunction with a simple spacecraft design utilizing a high degree of inheritance.

99-356 **Venus Surface Sample Return: A Weighty High-Pressure Challenge** –T. Sweetser,
0850am J. Cameron, G-S. Chen, J. Cuts, B. Gershman, M.S. Gilmore, J.L. Hall, V. Kerzhanovich, A.D. McRonald, E. Nilsen, W. Petrick, D. Rodgers, C.G. Sauer, B. Wilcox, A. Yavrouian, W. Zimmerman, and the Advanced Projects Design Team of the Jet Propulsion Laboratory.

A mission to return a sample to Earth from the surface of Venus faces a multitude of challenges. Venus has a deep gravity well essentially equivalent to Earth's and a hot-house atmosphere which generates extremes of high temperature, density, and pressure unmatched at any other known surface in the solar system. The final design of such a mission is years away but the study results presented here show our current mission architecture as it applies to a particular mission opportunity, give a summary of the engineering and science trades which were made in the process of developing it, and identify the main technology development efforts needed.

99-355 **A Light-Weight Inflatable Hypersonic Drag Device for Venus Entry** –A.D. McRonald
0910am

The author analyzes three ballutes (inflatable decelerators) for a Venus sample return mission. The benefits of a ballute over a conventional rigid entry body are less entry system mass, reduced heating rate and g-load, easier and better control of aerocapture, and freedom from fitting the orbiter or lander into a body with a substantial heat shield and a limited range of center of gravity. The ballute also gives stable attitude where a rigid body usually has trouble: at low atmospheric density and in transonic flight. The author will discuss wind tunnel and flight tests to validate the ballute entry concept.

99-357 **Venus and Beyond Using the Ariane ASAP Launch Capability** –P.A. Penzo
0930am

A new trajectory technique allows planetary missions to be carried as secondary payloads on GEO launches. This method, called Moon-Earth Gravity Assist (MEGA), requires 3 or more major maneuvers, together with close flybys of the earth and moon. This process is immediately available on the Ariane 5 and will be used, as currently planned, to carry an aircraft to Mars in 2003. The method is generally applicable to escape missions, but differs for missions to the inner planets, Venus and Mercury. This paper discusses the MEGA process as applied to Venus, and includes specific mission possibilities to other targets using the effectiveness of Venus gravity assist.

0950am - 1010am Coffee Break

99-358 **Aerobraking at Venus and Mars: A Comparison of the Magellan and Mars Global**
1010am **Surveyor Aerobraking Phases** –D.T. Lyons

On February 4, 1999 the Mars Global Surveyor spacecraft became the second spacecraft to successfully aerobrake into a nearly circular orbit about another planet. This paper will highlight some of the similarities and differences between the aerobraking phases of this mission and the first mission to use aerobraking, the Magellan mission to Venus. Although the Mars Global Surveyor (MGS) spacecraft was designed for aerobraking and the Magellan spacecraft was not, aerobraking MGS was a much more challenging task than aerobraking Magellan, primarily because the spacecraft was damaged during the initial deployment of the solar panels. The MGS aerobraking phase had to be completely redesigned to minimize the bending moment acting on a broken yoke connecting one of the solar panels to the spacecraft.

99-359 **GEM-2000 Satellite Tour Design** - R.J. Haw, J.R. Johannesen, C.A. Halsell, M.G. Wilson,
1030am J.L. Pojman

A study of a tour extension to augment the current Galileo Europa Mission satellite tour around Jupiter was initiated in 1998. Remote sensing observations will continue to be important in this extension, but unique in-situ fields and particles measurements will have a high priority in any post-GEM tour. A significant feature of any possible post-GEM tour is the possibility of simultaneous fields and particles experiments with the Cassini spacecraft as it swings by the Jupiter system in December 2000.

99-360 **Europa Orbiter Mission Trajectory Design** –J.R. Johannesen, L.A. D'Amario
1050am

The Europa Orbiter mission will place a spacecraft in orbit about Europa to investigate whether a subsurface ocean exists on Europa. The initial approach phase at Jupiter utilizes a Ganymede flyby and an orbit insertion maneuver performed at about 12.5 RJ perijove range, followed by a nearly ballistic orbital tour lasting about a year which reduces the orbital period to about 10 days (3 times Europa's period). In the "endgame" phase, a series of nearly resonant Europa encounters and apojove maneuvers further reduce the period about Jupiter to be nearly commensurate with that of Europa, leading to insertion about Europa and a 30-day orbital mission. Third-body perturbations are used to reduce orbital insertion ΔV at Europa.

99-361 **Automated Design of Aerogravity-Assist Trajectories** –E.P. Bonfiglio, J.M. Longuski,
1110am N.X. Vinh

An aerogravity assist (AGA) can reduce energy and propellant costs in space missions. The spacecraft consists of a lifting body that flies through the atmosphere of another planet, tremendously amplifying the gravity assist. A program is introduced which automatically finds AGA trajectories over a wide range of launch dates and energies. An analytical solution for the flythrough trajectory is developed and verified by numerical simulation. The automated design program is used to find energy-efficient AGA trajectories for various planetary missions. For example, with AGA, a spacecraft can reach Pluto in 8 years for a C3 of 49 and a Lift/Drag ratio of 7.

SESSION 9 *8:30 AM – 11:50 AM*

Session Room: Columbia Ballroom
Session Title: Orbit Determination
Session Chair: David Cicci

Department of Aerospace Engineering
211 Aerospace Engineering Building
Auburn University
Auburn, AL 36849-5338
Tel: (334) 844-6820
Fax: (334) 844-6803
Email: dcicci@eng.auburn.edu

99-362 0830am **Parametric Analysis for the Selection of Data Source in the Orbit Determination Processes** – M. Bir, S. Carter, Dr. C. Sabol

Many factors affect the accuracy achievable in the orbit determination and prediction process, including the available tracking information, the type of propagator that is used, and the orbital parameters of the satellite. This paper discusses the development of a matrix that cross-references typical tracking data sources and propagators with the resulting orbit determination and prediction accuracy for standard orbit classes. For each orbit class, tracking information sources were evaluated by comparing orbits obtained from fits to this data with an appropriate ‘truth’ orbit. The fits were performed using the Draper version of the Goddard Trajectory Determination System (DGTDS) which supported the evaluation of Cowell numerical, Draper semi-analytic, and simplified general perturbations analytic propagators. The design process for the MightySat II.1 spacecraft is used as an example.

99-363 0850am **A Fresh Look at Angles-Only Orbit Determination** – S. Carter, Dr. C. Sabol, R. Burns, M. Bir

Although the roots of angles-only orbit determination extend back to the times of Laplace in the late 1700s, today’s astrodynamics community has at best been uncertain as to the applicability of Gaussian techniques for accurate satellite orbit determination. This approach provides a modestly robust method of processing angles-only information, and combined with improvements in the accuracy of optical observations in recent years, could provide a reasonable method for orbit determination. This paper addresses the applicability of angles-only information in the orbit determination processes by utilizing real-world test cases from Air Force Research Laboratory (AFRL) assets. The results demonstrate the utility of precise angles information in the context of high-accuracy orbit determination.

99-364 **Low-Earth Orbit Estimation from Single Ground Station** - F. Curti, M. Parisse
0910am

There is a straightforward relationship between the orbital parameters and the antenna angles with the respective rates, ranging and ranging rate measurements. In fact, as soon as the antenna autotracking starts, the antenna angles are strictly related to the satellite motion such that the angle dynamics is driven by the satellite dynamics. In addition, by using the on-board transponder and the ground station range system, the range and range rate can be evaluated. In some LEO small satellite missions, to reduce the operation and manufacturing costs, it is convenient to have not a range measurement system; as a consequence, range and range rate are not available. The present work investigates the possibility of estimating the orbit by using only the azimuth and elevation angles and rates data of a single ground station, in the autotracking phase, processed with the Earth magnetic field measurements. In fact, the Earth magnetic field vector is a function of the orbit position and therefore contains the information on the range parameter. The magnetic field data can be carried out from the telemetry frames, in which are packed the along track field measurements.

99-365 **Performance of a Dynamic Algorithm for Processing Uncorrelated Tracks** –
0930am K.T. Alfriend, J-I. Lim

The processing of uncorrelated tracks (UCTs) is currently performed using a static algorithm, that is, the correlation criteria are the same for all tracks in LEO. In this paper, the performance of a previously proposed dynamic algorithm that uses the track covariance as the correlation or association volume is evaluated. The theoretical performance is established and then through simulations it is validated. Then the performance of the algorithm using actual space track data is evaluated.

0950am - 1010am Coffee Break

99-366 **The Naval Space Command Automatic Differential Correction Process** –D.A. Danielson,
1010am D. Canright, D.N. Perini, P.W. Schumacher

Naval Space Command maintains a database of element sets for roughly 9,000 Earth-orbiting objects, which is essentially the U.S. Space Command satellite catalog. NAVSPACECOM receives about 270,000 observations per day and performs an average of 18,000 element set updates per day. About 98.5% of the element sets are updated, without human intervention, by computer software called AUTODC. The purpose of our report is to elucidate the technical aspects of AUTODC for the astrodynamics community. Topics covered include: mathematics of batch least squares differential correction process, calculation of residuals and partials, inclusion of historical data, solution to normal equations, iterations and tolerances.

99-367 **GPS Carrier Phase Ambiguity Resolution Using Satellite-Satellite Single Differences** –
1030am M.J. Gabor, R.S. Nerem

Aliased in GPS carrier phase ambiguities are initialization constants. Instead of double differencing to remove these non-integer phase contributions, single differences remove the receiver effects. The satellite effects are estimated from a globally distributed network of receivers used for GPS precise orbit computation. Once satellite pair biases are calibrated, this paper's method allows single site ambiguity resolution that takes advantage of the bootstrapping of the network. Time series of the widelane carrier phase calibrations show definite trends. Because of the higher relative noise, it is more difficult to see trends in the L_1 bias calibration.

99-368 **Automated Tuning of an Extended Kalman Filter using the Downhill Sismplex Algorithm**
1050am T.D. Powell

The paper outlines the application of the 'Downhill Simplex' numerical optimization algorithm to the problem of tuning an Extended Kalman Filter. The tuning problem is defined, as well as the application of the simplex algorithm. A number of examples of increasing complexity are presented to illustrate the utility of the method.

99-369 **Altitude Effects on Autonomous Orbit Determination** –C.A. McLaughlin, K. Gold,
1110am G.H. Born

The reduced dynamic technique (RDT) has become popular for removing unmodeled dynamic error. The technique is particularly useful for removing errors caused by poor atmospheric modeling. This paper examines the ability of RDT to remove drag error at low altitudes. The error was simulated by perturbing the drag coefficient. The orbit determination was performed using simulated GPS data with the GIPSY-OASIS II software package. The RDT sigma's for minimizing the orbit error are presented for low altitude, sun synchronous orbits. In an autonomous application, RDT removes drag as a significant error source at altitudes above 350 kilometers.

99-370 **Geo-Stationary Orbit Determination by Means of a Parabolic Sinusoidal Fit for**
1130pm **Satellite Longitude and a Sinusoidal Fit for the Sine of Satellite Declination** - B. Srinivasan

A new orbit determination method, conceived, worked out and applied is presented here for application to the domain of near geo-stationary orbits. A parabola + sinusoidal combine has been chosen to fit the data equivalent of the satellite longitude and a sinusoidal function to fit the data equivalent of the sine of the satellite declination derived from tracking data. With reduced complexity and run time, it is good for bulk simulations. The orbit determination accuracy is quite better than that from preliminary orbit determination methods. The ground trace generated compares well over a day, making it useful for monitoring collocation operations.

SESSION 10 ***1:30 PM – 4:30 PM***

Session Room: Columbia Ballroom
Session Title: Attitude Dynamics and Control: Applications
Session Chair: Christopher Hall
Aerospace and Ocean Engineering
Virginia Tech
Blacksburg, VA 24061-0203
Tel: (540) 231-2314
Fax: (540) 231-9632
Email: chall@aoe.vt.edu

99-371 0130pm **Flight Control Overview of STS-88, the First Space Station Assembly Flight** – R.A. Hall, K. Kirchwey, M. Martin, G. Rosch, D. Zimpfer

When the Space Shuttle Endeavor undocked from the Zarya/Unity configuration, it marked the completion of the most challenging shuttle mission to date and the beginning of an enormous task of assembling the International Space Station. The first Station flight offered an array of complex dynamics and control challenges in order to mate the American module 'Unity' to the Russian module 'Zarya'. An overview of the shuttle flight control system design and performance is presented, including the issue of control/structure dynamic interaction and the design of a mated vehicle reboost capability.

99-372 0150pm **Space Station Attitude Control During Payload Operations** – N. Bedrossian

Evaluating the feasibility of planned robotic operations requires an analysis methodology and tools that can quickly assess proposed attitude control strategies. In this paper, an efficient approach to model the attitude dynamics of the Space Station during payload motion is presented. This approach is used to formulate an alternative numerical solution for torque equilibrium attitudes including the effect of payload velocity. It is also shown that a judicious choice of attitude command during robotic operations can substantially reduce peak attitude control momentum use.

99-373 0210pm **GOES On-Orbit Storage Mode Attitude Dynamics and Control** – E. Harvie, A. Dress, M. Phenneger

A method of spin-stabilization for on-orbit storage of the normally 3-axis stabilized GOES I/M spacecraft is described. The major axis spin angular momentum is processed using magnetic dipole coils to track the one degree-per-day motion of the Sun. Acquisition of the eigenaxis spin and techniques for spin-axis precession control exploit spacecraft design features not originally intended for such a mode. Long-term attitude dynamics are particularly influenced by solar radiation pressure torque. The method has been proven in-flight for two GOES spacecraft.

99-374 **ICESat Attitude Algorithm for Maintained Reference Groundtrack Pointing** –
0230pm D.G. Kubitschek, K. Gold, M. Ondrey, P. Axelrad, G.H. Born, D. Hill

The Geodetic Laser Altimeter System (GLAS) mission is designed to measure changes in the elevations of the polar ice sheets. The ICESat spacecraft will carry the GLAS altimeter, and science requirements for the GLAS mission demand that the laser altimeter be pointed to within 50 m of a predetermined reference groundtrack. This paper discusses the development of the attitude pointing algorithm, inclusion of topography, and an overview of the operational system that will be used to generate the command attitude quaternions for upload to the on-board attitude control system.

0250pm - 0310pm Coffee Break

99-375 **Autonomous Control of the Proba Spacecraft** –J. de Lafontaine, P. Vuilleumier,
0310pm P.Van den Braembussche

The European Space Agency is currently developing the PROBA spacecraft, an Earth-observation mini-satellite whose mission objective is to demonstrate the feasibility and benefits of autonomy in space. Potential benefits include the reduction of ground-station operating costs and a better management of limited on-board resources. This paper presents the operation and predicted performance of the PROBA Attitude Control and Navigation System which enables the acquisition of orbit and attitude knowledge, the planning of spacecraft operations and the execution of attitude maneuvers required by the imaging spectrometer. The critical imaging maneuvers will be analyzed with emphasis on the autonomous attitude control functions.

99-376 **Integrated Structural and Control Optimization of a Large Space Structure Using**
0330pm **Analytical Sensitivity Analysis** –I.M. Fonseca, P.M. Bainum

This work deals with the analytical sensitivity analysis procedure to implement the integrated structural and control optimization of a large low-Earth orbit space structure. The objective is to show that significant computation time can be saved when the analytical procedure is used instead of numerical calculation of the sensitivity. Despite the fact that the numerical approach is attractive because it does not require the hard algebraic task to develop the sensitivity derivative expressions, the approach is often responsible for the major computational cost of an optimization process. The analytical approach represents an attractive option when the computational effort becomes prohibitive under cost considerations.

99-377 0350pm **Comparison of Melnikov's Method and Numerical Simulation for Predicting Nonlinear Dynamics in the Pitch Motion of Actively Controlled Satellites in a Gravity Gradient Field** –M.D. Toniolo, M. Lee, G.L.Gray

We compare the predictions of Melnikov's method, which is applied to study the effect of feedback control on the existence of chaos in the pitch dynamics of a satellite in a gravity gradient field, to a numerical investigation of the same system. An analytical method allows the global investigation of the system parameter space, as opposed to using a numerical method, in which specific system parameters must be chosen for each simulation. On the other hand, the range of validity of the analytical method must be verified. These results shed light on the interaction of pitch dynamics and control in satellites.

99-378 0410pm **ATTDES - Generalized Equations of Motion** –D.L. Mackison

Attitude Determination and Control System (ATTDES) is a collection of Matlab programs for attitude studies. The object is the rapid development of equations of motion, selection of sensors, and the design of control and estimator gains. The equations of motion for ATTDES are generated using Matlab's symbolic mathematics capability in a Lagrange formulation, where the respective energy terms and their generalized coordinates can be added to the Lagrangian by selecting rigid, flexible, gimbaled, and momentum components for the dynamics, and arbitrary sensor models - star, sun, earth, magnetic, and gyros, for example- for the measurement schemes.

SESSION 11 ***1:30 PM – 4:30 PM***

Session Room: Columbia Ballroom
Session Title: Special Session: Atmospheric Modeling
Session Chair: Frank Marcos
Air Force Research Laboratory/VSBP
29 Randolph Road
Hanscom AFB, MA 01731-3010
Tel: (781) 377-3037
Fax: (781) 377-9950
Email: marcos@plh.af.mil

99-379 0130pm **The Solar Cycle and its Effect on Neutral Density and Satellite Drag** –R. Viereck, J. Joselyn

The neutral density of the upper atmosphere changes by more than an order of magnitude over the course of the 11 year solar cycle. This produces large changes in satellite drag. Changes in solar EUV radiation and in geomagnetic activity produce most of the fluctuations in neutral density. Because geomagnetic activity is correlated with solar activity, during solar maximum, the combined effects of increased EUV flux and increased geomagnetic activity can produce drastic increases in atmospheric drag. In this presentation, I will review the cause and effects of increased solar and geomagnetic activity on neutral density and atmospheric drag. Predictions for the present Solar Cycle will be presented and compared with previous cycles. The use of proxies to estimate the solar contribution satellite drag will be examined and compared with actual measurements. Improvements in modeling and measurements will be discussed.

99-380 0150pm **Atmospheric Density, Solar Activity, and Spacecraft/Satellite Characteristics: An Integrated Relationship for Orbital Lifetime Prediction Assessment** - J. Owens, W.W. Vaughan, K. Niehuss

The ability to accurately predict a spacecraft/satellite orbital lifetime, insertion altitude, re-boost requirements, and mission performance is mainly the result of the integrated effect from knowledge of the atmospheric density, solar activity, and timeline of vehicle characteristics. Each of these elements is dependent upon a model developed to provide the inputs necessary for the use of an orbital lifetime prediction program. This paper will address relative influences of these elements based on uncertainties associated with the model products used to predict orbital lifetime and related spacecraft/satellite design and operational conditions. Issues associated with the potential for improvement of the lifetime prediction model input elements will be discussed with regard to their relative contributions to improving orbital lifetime and performance predictions.

99-381 **Satellite Drag Accuracy Improvements from Neutral Density Model Calibration -**
0210pm F.A. Marcos, J. O. Wise, M. J. Kendra, J. N. Bass, D.R. Larson, J.J. Liu

The major limitation for low-altitude satellite precision orbit determination is knowledge of atmospheric drag. Operational satellite drag deficiencies are dominated by the inability to accurately specify neutral density variations. Time-dependent neutral density corrections to operational drag models have been derived based on assimilation of satellite tracking data. The approach uses a special perturbations orbit propagator for orbit determination. In the orbit determination process, the model density errors are absorbed into the satellite's ballistic coefficient to obtain an appropriate value for the true drag acting on the satellite. The ballistic coefficient is then an additional scalar parameter that is adjusted so as to be as consistent as possible with the observed satellite trajectory. If the satellite's true ballistic coefficient is known, however, the drag adjustments can be interpreted as errors in the neutral density model. These corrections to the model, obtained from one satellite, are then extrapolated globally based on empirical expressions within the model. This atmospheric calibration technique provides dramatic improvements in providing an accurate global depiction of neutral density fields. The typical 15% error of current operational models is reduced to 5% or less.

99-382 **A Methodology for Using Optimal MSIS Parameters Retrieved from SSULI Data to**
0230pm **Compute Satellite Drag on LEO Satellites** - A.C. Nicholas, J.M. Picone, S.E. Thonnard

The Naval Research Laboratory (NRL) has developed five ultraviolet remote sensing instruments for the Air Force Defense Meteorological Satellite Program (DMSP). These instruments, known as Special Sensor Ultraviolet Limb Imager (SSULI), will launch aboard the DMSP Block 5D3 satellites starting in 2001. In conjunction with the hardware program, an extensive operational data processing system has been developed. This system, known as Ground Data Analysis Software (GDAS) includes data reduction software using NRL's advanced physics-based inversion algorithms, a customized graphical data interface, and comprehensive validation techniques. The purpose of the SSULI program is to provide detailed measurements of the composition of the upper atmosphere and ionosphere. Every 90 seconds, the SSULI sensor observes limb airglow intensities from 110 to 127 degrees from the satellite zenith. From these intensities, the operational analysis programs will determine the atomic oxygen ion density and neutral density composition from 120 to 600 km. These composition soundings from the SSULI sensors will be combined with data from other sensors and with atmospheric models to provide a global specification of the near-Earth space environment. The goal of global specification is to generate warnings, forecasts, and corrections for DoD and civilian resources that are affected by the variability of the upper atmosphere. Scientifically, this information will be used to gain a better understanding of the physical processes and dynamics that govern the upper atmosphere. In addition to the development, testing and validation of the SSULI hardware and software, NRL maintains a strong commitment to the transition of new technology to improve the performance, effectiveness and reliability of DoD assets. For the SSULI program, a key technology transition area is the improvement of satellite tracking and orbit prediction. For low-Earth orbit (LEO) satellites, solar variability correlates to variability in satellite drag. On several occasions, large solar bursts have caused such severe changes in satellite drag that an accurate position of the satellite is lost until the trajectories of all the

objects in the catalogue can be reanalyzed. NRL is currently developing techniques to apply the neutral density information from the SSULI sensors to optimize the MSIS model for superior satellite tracking capabilities. To demonstrate this process, we will ultimately analyze several objects in low LEO, for which accurate ground truths are available, using a high precision orbit propagator and our data inversion software. The propagator integrates the equations of motion using Runge-Kutta-Fehilberg methods of order 7-8. The MSIS, Jacchia 1971, and the Harris-Priester models at the time of the ground truth measurement will provide alternative representations of the atmospheric density. These models will also be used to simulate atmospheric densities for solar minimum and solar maximum conditions. Using these simulated conditions, we will provide detailed examples of the baseline methodology, in which optimal MSIS parameters are retrieved from either drag or SSULI UV data to provide an accurate global three-dimensional density field. The resulting improvement on orbit determination will then be quantified. By the time of this meeting, a copy of the SSULI instrument, the Low Resolution Airglow and Auroral Spectrograph (LORAAS), will have been launched on the Advanced Research and Global Observation Satellite (ARGOS). It is expected that preliminary results from analysis of early LORAAS data will be presented. This will serve as a proof of concept of the operational methodology proposed for the SSULI data.

0250pm - 0310pm Coffee Break

99-383 0310pm **Neutral Atmosphere Density Monitoring Based on Space Surveillance System Orbital Data** –P.J. Cefola, A. Nazarenko, R.J. Proulx, V. Yurasov

One approach for increasing the accuracy of satellite orbit determination and prediction for LEO satellites is the organization of an upper atmosphere monitoring function. This would be the analog of a weather service in the lower atmosphere. Monitoring of the upper atmosphere based on the use of the available satellite atmospheric drag data (ballistic factors) on all catalogued LEO satellites offers a low-cost approach to this capability. These data are operationally updated in the Space Surveillance System (SSS) as a result of regular satellite observations. It is concluded that there are actual possibilities for operational monitoring of the global atmospheric density variations at altitudes ranging from 200 up to 600 km. The elaboration of a plan for a real data test of the upper atmospheric monitoring concept is discussed.

99-384 **Calibration of Semi-Empirical Atmosphere Models through the Orbital Decay of**
0330pm **Spherical and Cylindrical Satellites** - C. Pardini, L. Anselmo

We performed a calibration of the semi-empirical air density models (JR71, MSIS86, MSIS90, TD88) included in the trajectory predictor SATRAP, specifically developed and implemented at CNUCE for the study of decaying space objects. The analysis was based on the decay histories of spherical and cylindrical satellites, whose orbital state evolution and physical characteristics (area and mass) were known. The results obtained suggest that no model is able to correctly compute the atmospheric density at all altitudes. While the MSIS86 and the MSIS90 models give practically the same results above 200 km, and can be considered the best models to compute air density below 400-500 km, the JR71 model seems to be more precise for higher altitudes, mostly in conditions of high solar activity.

99-385 **Satellite Drag Accuracy Improvements Estimated from Orbital Energy Dissipation Rates**
0350pm –M.F. Storz

Satellite drag is the dominant perturbation force effecting the prediction of low-altitude satellite trajectories. The satellite drag force is proportional to the atmospheric density that is highly variable and difficult to model. Errors in current thermospheric density models (e.g., the Jacchia or MSIS models) can cause significant errors in predicted satellite positions. The highly variable orbital energy dissipation rates observed for low-altitude satellites may be exploited to solve near real-time for a more accurate representation of thermospheric density. A theoretical algorithm for solving for the neutral density is described. This algorithm estimates the neutral density indirectly through exospheric temperature parameters. It uses observed energy dissipation rates from a variety of satellites as input. Orbits with different altitudes, inclinations, eccentricities and orbital planes can be exploited simultaneously to generate a time-varying global density field. The density field can be used operationally to improve the accuracy of predicted trajectories for all low-altitude satellites.

99-386 **Utilization of Mars Global Surveyor Accelerometer Data for Atmospheric Modeling** -
0410pm R.H. Tolson, S.N. Noll, G.M. Keating, D.T. Baird, T.I. Shellenberg

Accelerometer data, taken during the aerobraking phases of the MGS mission to support operations, will be used to develop models of the Martian thermosphere. Methods are developed to address the numerous issues relative to such utilization. Issues peculiar to MGS and Mars include accounting for the multi-body dynamics due to the broken solar array, thruster activity, high-frequency signals of varying frequency due to solar panel vibration, and nearly instantaneous changes in dynamic pressure which may be due to atmospheric shock waves.

SESSION 12 ***1:30 PM – 4:30 PM***

Session Room: Columbia Ballroom
Session Title: Stationkeeping and Proximity Operations
Session Chair: Todd Ely
MS 301-125L
Jet Propulsion Laboratory
4800 Oak Grove Drive
Pasadena, CA 91109-8099
Tel: (818) 393-1744
Fax: (818) 393-6388
Email: tely@pop.jpl.nasa.gov

99-387 0130pm **An Operational Approach for Generating Near-Optimal Station Keeping Strategies via Parallel Genetic Algorithms** - J.E. Smith, R.J. Proulx, P.J. Cefola, J.E. Draim

Extending upon the results of the authors' previous parallel genetic algorithm optimization approach, this study investigates ways in which the parallel genetic algorithm can be used as the basis for an operational stationkeeping system. Specifically, an orbit is defined and parallel genetic algorithms are applied in such a manner that the orbit is maintained within a given set of tolerances. However, unlike the previous study that focused only on maintaining the orbit within the state constraints, this study focuses on ways to maintain near-optimality in the stationkeeping maneuvers, while also maintaining the operational characteristics of repeatability, speed of convergence and ease of implementation. Finally, the use of this operational stationkeeping algorithm as a planning tool is discussed.

99-388 0150pm **A Practical Stationkeeping Method for Modular Geosynchronous Satellites with a Xenon Propulsion System** –A.A. Kamel, W. Gelon, K. Reckdahl

A new method is described using xenon propulsion system for stationkeeping of modular satellites. In these satellites, the xenon plasma thrusters are mounted at a large cant angle to the solar arrays and, therefore, produce a large radial delta V when performing North/South stationkeeping. To minimize the impact of this radial delta V, two plasma thrusters are used to simultaneously correct orbital inclination and eccentricity. Less efficient (bipropellant) thrusters are used to correct the orbital semimajor axis which only requires about 5% of the total delta V. This stationkeeping method remains the same in the presence of thruster failure.

99-389 **Impact of Eccentricity on East-West Stationkeeping for the GPS Class of Orbits –**
0210pm T.A. Ely

There exists a strong relationship between eccentricity and the potential for a repeating groundtrack orbit to exhibit chaotic motion. These complex motions can significantly impact the east-west stationkeeping process for maintaining the repeating groundtrack property of a commensurate orbit. The focus of the current study is to investigate orbits with characteristics that are similar to GPS satellites except with modestly larger eccentricities. It will be shown that at eccentricities larger than ~ .01 the chaotic regions become significant, and the need arises for a robust stable stationkeeping approach. Furthermore, the investigation will develop an analytical model for eccentricity (the dominate effect being the luni-solar perturbations) and show the factors that contribute to its growth, thus increasing the probability of encountering chaotic motion during a typical satellite lifetime.

99-390 **Earth-Centered Angle Stationkeeping in Low-Earth Orbits with Yaw-Steering**
0230pm **Constraints** –V. Martinot, D.Vignaux

The Earth-centered angle between the reference and actual positions of a satellite is the natural stationkeeping criterion for telecommunication applications. This paper presents the related stationkeeping strategy in low-Earth orbits for a satellite in yaw-steering mode in case the correcting maneuvers can only be done when the actuators - fixed with respect to the satellite - are sufficiently close to the optimal thrust direction. For illustration, a case study is selected for a commonly-used orbit for emerging telecommunication systems (around 1500 km). The robustness of the strategy to various error sources is addressed.

0250pm - 0310pm Coffee Break

99-391 **Maintenance of the ICESat Exact Repeat Ground Track** –P. Demarest, B.E. Schutz
0310pm

The Ice, Cloud, and land Elevation Satellite will measure ice-sheet topography using the Geoscience Laser Altimetry System. A simulation was conducted to determine the maneuvers required to maintain the ground track, at all latitudes, within 800-m of a repeating reference. Longitude targeting was used to maintain the ground track at the equator. A latitude targeting scheme has been developed to maintain the ground track at high latitudes. A maneuver simulation for the entire 3-year mission of ICESat resulted in 156 tangential maneuvers totaling 17.861 m/s of ΔV and 7 normal maneuvers, totaling 11.37 m/s of ΔV.

99-392 **Orbit Analysis of the Adjoint Satellite to Space Station** –J. Luo, X. Dou, Y. Huang
0330pm

The Space Station Adjoint Satellite (SSAS) is a special spacecraft servicing space station. Its main functions are to improve the in-orbit servicing condition of space station, guarantee the space station flying in safety, decrease the times and period of the EVA of astronauts and extricate the astronauts from the time consuming, heavy and risky activities. The concept and the missions of SSAS are briefly introduced in this paper. The model of adjoint kinematics is given. In order for passive orbit keeping of SSAS, release or launch of SSAS should meet the adjoint conditions. These conditions are analyzed and presented. The adjoint orbit under some major orbital disturbance is analyzed and the calculation results are given. Finally, some useful conclusions are reached by comparing the calculation results of different disturbance effect.

99-393 **Close Approach Spacecraft Maneuver Criterion** –J.R. Wright
0350pm

Space assets are frequently faced with the threat of collision with other space objects. Space asset managers make spacecraft collision avoidance maneuver decisions based on ad-hoc criteria derived from incomplete available information. The purpose of this paper is to define a rigorous asset maneuver criterion derived from complete available information, and to describe its implementation.

99-394 **Contingency ISS Rendezvous Recovery Planning by Houston and Moscow Control**
0410pm **Centers** –D.R. Adamo

During launch to International Space Station (ISS), contingencies compromising Space Shuttle Orbiter (SSO) propulsive capability may preclude nominal rendezvous. In some of these contingency scenarios, ISS translation maneuvers to a lower orbit height, planned in real time, can resurrect SSO rendezvous capability. Expending some ISS propellant in such scenarios can thereby avoid a costly and disruptive delay to SSO cargo delivery. A cooperative response to these contingencies, called Joint Underspeed Recovery (JURe), has been negotiated by flight mechanics specialists from Houston and Moscow Control Centers. Constraints, strategies, procedures, and a dedicated spreadsheet associated with JURe operations are presented. An example "paper simulation" is also provided.

SESSION 13 *Wednesday, August 18, 1999*
8:30 AM – 11:30 AM

Session Room: Columbia Ballroom
Session Title: Mission Design: Near Earth, Earth/Moon, and Libration Points
Session Chair: David Folta
Code 571.0
NASA Goddard Space Flight Center
Greenbelt, MD 20771
Tel: (818) 354-3209
Fax: (818) 393-5214
Email: david.folta@gsfc.nasa.gov

99-395 **Mission Planning for the CHANDRA X-ray Observatory** –L.D. Mullins, R.L. Stone,
0830am S.W. Evans

The scientific purpose of the Chandra observatory is to do astronomical research in the x-ray portion of the electromagnetic spectrum (0.1 - 10.0 keV), both high-resolution spatial imaging (0.5") and moderate- to high-resolution spectroscopy. This paper describes the mission planning that was conducted at MSFC to design the orbit and launch window that would permit the observatory to function properly within its constraints and resources for at least 5 years and maybe 10 years without any reservicing (which is impossible for the orbit that it is in). This mission planning also addressed the orbital transfer sequence required to take the observatory from its initial parking orbit to the final operating orbit.

99-396 **System Design Considerations for a Formosa-X Microsatellite** - J-S. Chern
0850am

In summer, it happens very often that typhoons attack Taiwan. The strong and variable dynamic pressure of the wind usually causes significant damage. This is the first wave of the triplicate loss caused by typhoon. The second wave of loss is resulted from the flood, due to the heavy rain brought by the typhoon. Then, in some mountainous areas the water will mix with loose ground and rocks to form the so-called soil-rock-flow. It is the soil-rock-flow that can cause the third wave of damage. So far, the formation and the size of typhoon, even its structural pattern and flight path, can be observed by using the meteorological satellites and radars. The purpose of this paper is to present the system design of a microsatellite named Formosa-X, for the experiments of soil-rock-flow monitor, mapping communication, space science, and space education.

99-397 0910am **The Lunar Prospector Mission: Final Results of Trajectory Design, Quasi-Frozen Orbits, Extended Mission Targeting, and the Lunar Topography and Potential Models** –D. Folta, M. Beckman, D. Lozier, K. Galal

The National Aeronautics and Space Administration (NASA) selected Lunar Prospector as one of the discovery missions to conduct solar system exploration science investigations. The mission was NASA's first lunar voyage to investigate key science objectives since Apollo and was launched in January 1998. In keeping with discovery program requirements to reduce total mission cost and utilize new technology, Lunar Prospector's mission design and control will focus on the use of innovative and proven trajectory analysis programs. As part of this effort, the Goddard Space Flight Center and the Ames Research Center became partners in the Lunar Prospector trajectory team to provide the trajectory analysis and orbit determination support. At the end of 1998, Lunar Prospector completed its one-year primary mission at 100-km altitude above the lunar surface. On December 19, 1998, Lunar Prospector entered its extended mission phase. The mission orbit was lowered from 100 km to a mean altitude of 40 km. Due to lunar potential effects, the altitude of Lunar Prospector varied from 25 to 55 km above the mean lunar geoid. After one month at 40 km, the lunar potential model was updated based upon the new tracking data at 40 km. On January 15, 1999, the altitude was lowered again to a mean altitude of 30 km. The spherical altitude varied between 15 km and 45 km above the mean lunar geoid while the topographical altitude varied between 10 km and 50 km. Various means were employed to get accurate lunar surface elevation including Clementine altimetry and LOS analysis. Based upon the best available terrain maps, Lunar Prospector reached actual altitudes of 8 km above lunar mountains in the southern polar region. This extended mission phase of six months will enable LP to obtain science data up to 3 orders of magnitude better than at the mission orbit. At the end of the operations mission, LP was targeted for impact at a chosen location that allowed optical observation of the lunar ejecta as LP ended its mission at 1.6 km/sec. This paper details the trajectory design and orbit determination planning and actual results of the Lunar Prospector nominal and extended mission including maneuver design, eccentricity vs. argument of perigee evolution, topographical altitude estimation, and lunar potential modeling. This paper provides understanding of the quasi-frozen orbit design of the LP mission, the optimization process of lunar orbit targets, the impacts that the selected lunar potential models play, and discusses the feasibility of meeting the mission goals. Observed evolution of the Keplerian orbit elements are compared to the theoretical predictions using the latest lunar potential model available which incorporates the Lunar Prospector Doppler data. Mapping orbit maintenance maneuver design along with results of the actual maneuvers to maintain the orbital requirements are also presented.

99-398 0930am **Genesis Trajectory Design** –J.L. Bell, M.W. Lo, R.S. Wilson

The Genesis mission will launch in 2001, sending a spacecraft into an L_1 halo orbit in the Sun-Earth system to collect solar wind samples. In 2003, the samples will be returned to the Utah Test and Training Range for a daylight, mid-air recovery. A parametric study of the Earth entry conditions has helped to characterize the solution space. Several perturbation and robustness studies were conducted to analyze the sensitivity of the trajectory. These studies indicate that the trajectory can be adjusted to accommodate multiple perturbations.

0950am - 1010am Coffee Break

99-399 1010am **Maneuver Design and Calibration for the Genesis Spacecraft** –K.E. Williams, P.E. Hong, D. Han, R.P. Zietz

As the fifth Discovery mission, Genesis will collect solar wind samples for a period of approximately two years while in orbit about the Earth-Sun L_1 point. This paper addresses the design of propulsive maneuvers for Genesis which achieve Science objectives while minimizing cost in light of a number of spacecraft design challenges and constraints. Topics to be discussed include the spacecraft design, maneuver decomposition algorithm, as well as operational timelines, procedures and plans for calibrations to improve end-of-mission performance for accurate delivery and sample recovery.

99-469 1030am **Entry Dispersion Analysis for the Genesis Sample Return Capsule** –P.N. Desai, F. McNeil Cheatwood

The fifth of NASA's Discovery class missions is a sample return mission known as Genesis. The spacecraft will be inserted into a halo orbit about the L1 (Sun-Earth) libration point where it will remain for two years collecting solar wind particles. Genesis is scheduled to be launched in January 2001 and will be the first mission to return samples from beyond the Earth-Moon system. Upon Earth return in August 2003, the entry capsule containing the solar wind samples, will be released from the spacecraft (decelerating with the aid of a parachute) for a mid-air recovery in Utah over the U. S. Air Force's Utah Test and Training Range. This paper will analyze the entry, descent, and landing sequence for the returning sample capsule. This analysis consists of performing a trajectory simulation of the entire entry to predict the descent attitude and air-snatch conditions. In addition, a Monte Carlo dispersion analysis will be performed to ascertain the impact of off-nominal conditions that may arise during the entry to determine the robustness of the Genesis entry scenario. Specifically, the capsule attitude near peak heating and parachute deployment is of interest, along with the overall air-snatch footprint ellipse.

99-400 1050am **Triana Mission Design** –M. Beckman, J.J. Guzman

Named for the sailor on Columbus' voyage who first saw the New World, Triana is a NASA mission to the vicinity of the Sun-Earth L_1 point. From L_1, Triana will have a continuous, near full disk, sunlit view of the Earth. One of the mission goals is to release the Earth images, in near real-time, over the Internet. The mission design for Triana includes design of the Lissajous orbit, design of the transfer trajectory from Earth to L_1, and design of the launch and injection into the transfer orbit. The design of the nominal Lissajous and transfer trajectories incorporates dynamical systems theory and numerical techniques from both Goddard Space Flight Center and Purdue University.

99-401 **Optimization of Insertion Cost for Transfer Trajectories to Libration Point Orbits -**
1110am R.S. Wilson, K.C. Howell, M.W. Lo

The objective of this work is the development of efficient techniques to optimize the cost associated with transfer trajectories to libration point orbits. To produce an initial approximation to the transfer trajectory, dynamical systems theory is used to determine invariant manifolds associated with the desired libration point orbit. To refine the trajectory, a two-level differential corrections process is applied to ensure a fully-continuous solution, while including any constraints placed on the trajectory. Based on this methodology, and using the manifold structure from dynamical systems theory, a technique is presented to optimize the cost associated with insertion onto a specified libration point orbit. Examples are presented demonstrating this technique, including applications to the Genesis mission design.

SESSION 14 *8:30 AM – 11:50 AM*

Session Room: Columbia Ballroom
Session Title: Trajectory Optimization
Session Chair: John M. Hanson
ED 13
NASA Marshall Space Flight Center
Huntsville, AL 35812
Tel: (205) 544-2239
Fax: (205) 544-5416
Email: john.hanson@msfc.nasa.gov

99-402 **Numerical Research in Non-Coplanar Orbital Maneuvers** –A.F.B. de Almeida Prado,
0830am M.M.N. dos Santos Paulo, M.L.O. e Souza

Since it became necessary the use of vehicles equipped with propulsion systems to perform space missions, it became also necessary the study of the optimal transfer problem of a spacecraft between two given orbits. In this study, we studied the tridimensional optimal bi-impulsive transfers, solving the formulation of Altman and Pistiner; we develop an algorithm to solve the tridimensional bi-impulsive maneuvers initially without thrust errors, that was extended to study the maneuver with thrust errors; and we present the simulation results available for biimpulsive transfers between two given orbits. Finally, we present the conclusions obtained concerning the effects of the thrust errors in the tridimensional biimpulsive maneuvers.

99-403 **Optimal Low-Thrust Earth-Moon Targeting Strategy for N-Body Problem** –H. Yan
0850am

This paper presents optimal low-thrust Earth-Moon targeting strategy for n-body problem with the indirect method and thrust-coast model. The good numerical example for Earth-Moon transfer was shown in the paper by combination of B-plane targeting with optimal low thrust trajectories.

99-404 **Thrust Programming in Aeroassisted Maneuvers** –M. Ross
0910am

Based on their application, aeroassisted maneuvers may be classified as aerocapture, aerobraking, aerogravity assist, synergetic maneuvers and so on. In many applications, the thermal load on the vehicle cannot be ignored, and quite often, it is a driver. In this paper, we address the atmospheric pass of the aeroassisted maneuver, and include the convective heating-rate on the vehicle as a constraint. We show that the interior singular thrust arc fails to satisfy the Goh-Robbins condition for optimality while the boundary singular thrust arc fails to satisfy a nonnegativity condition supplied by the Karush-Kuhn-Tucker conditions. These results are used to postulate that higher efficiencies may be achieved by multiple thrust switches.

99-405 **The Optimization of Continuous Constant Acceleration Transfer Trajectories in the**
0930am **Presence of the J2 Perturbation** –J.A. Kechichian

Two different formulations using equinoctial elements are presented for the analysis of the minimum-time orbit transfer between general elliptic orbits perturbed by the earth second zonal harmonic. A previous formulation that resolves the perturbation vector along the direct equinoctial orbital rotating frame is extended to a new formulation that uses the true longitude as the sixth element and the polar frame for the component resolution of the perturbation vector. The simplest form of the adjoint equations are thus obtained and the need to solve for the Kepler equation removed leading thereby to a more robust software with improved convergence characteristics.

0950am - 1010am Coffee Break

99-406 **Necessary and Sufficient Conditions for Optimal Impulsive Rendezvous and Linear**
1010am **Equations of Motions** –T.E. Carter

Necessary and Sufficient Conditions are known for the problem of the impulsive minimization of the total characteristic velocity of a spacecraft subject to linear equations of motions. These conditions are global in the sense that the primer vector must be known over the entire flight interval. The purpose of the present paper is to present a new localized sufficiency condition that is derived from the original one, but requires calculation of information about the primer vector only at a few specific points. This new sufficiency condition has both theoretical and practical advantages over the previous one.

99-407 **Generalized Planetary Reentry Minimal Heating Trajectories** - M. Human
1030am

The equations of motion for an atmospheric entry body are normalized into dimensionless form, resulting in a system of equations that contain parameters defining both atmospheric and vehicle characteristics. Thus, results can be generalized for a variety of mission specifications. This model is used with an integrated minimal heat input objective function. Heating rates are represented by empirical correlations and is including in the lagrangian term. A shooting method integration procedure is used to solve the system. The specific problem addressed is that of skip entry trajectories where small angle approximations linearize the flight path variable. Results are generated for a combination of atmospheric and vehicle specifications, and the resulting sensitivities are discussed.

99-408 **Optimal, Low-Thrust, LEO to GEO/MEO/HEO Trajectories** –A.L.Herman, D.B. Spencer
1050am

In this study, a trajectory optimization technique based upon higher-order collocation (HOC) was used to solve optimal, low-thrust, Earth orbit transfer problems. For the cases solved, a spacecraft transfer from LEO to a variety of mission orbits. In this work, several cases were solved assuming a range of thrust accelerations varying from 1 to 10^{-3} g. A comparison is made between the optimal transfers found in this work and the transfers found in previous work where techniques were used to approximate the optimal thrust pointing control profiles.

99-409 **Optimal DeltaV-Earth-Gravity-Assist Trajectories in the Restricted Three-Body Problem**
1110am –G. Colasurdo, L. Casalino

The theory of optimal control is applied in the restricted three-body problem. The trajectory is considered as being composed of ballistic arcs that connect at ``corners" where the spacecraft velocity experiences jumps; no explicit control is present but the necessary conditions for optimality are nevertheless provided by the optimal control theory. Thrust is applied parallel to the velocity adjoint vector, or primer vector, at the corners, where the primer must have unit magnitude; in time-free problems the time-derivative of the primer magnitude must be null in correspondence of the corners. The numerical examples deal with simple DV-EGA maneuvers, which are two-burn (departure and deep space) time-free transfers in the three-body problem. The results are compared to similar results obtained using the patched-conic approximation. Attention is given to cases that correspond to minimum-height flybys; these cases require the addition of a constraint on the minimum distance from the Earth. A powered flyby is required when the primer magnitude suggests an additional burn in close proximity to the Earth.

99-410 **Low-Thrust Control Law Development for Transfer from Low Earth Orbits to High**
1130pm **Energy Elliptical Parking Orbits** –L.P. Gefert, K.J. Hack

The use of solar electric propulsion (SEP) to improve a spacecraft mission performance is becoming common place. Recent human Mars architecture studies have shown that a high efficiency, reusable, non-nuclear Earth departure system is possible though the combined use solar electric and chemical propulsion. To make this concept viable, the SEP vehicle must transfer systems from low-earth orbit (LEO) to a high-energy elliptical parking orbit (HEEPO). A four phased analytic control law has been developed to perform a LEO-to-HEEPO transfer. Each steering phase performs a specific task required for transfer from LEO-to-HEEPO. LEO-to-HEEPO transfer performance is reported and compared.

SESSION 15 ***8:30 AM – 11:10 AM***

Session Room: Columbia Ballroom
Session Title: Tethers
Session Chair: Arun K. Misra
McGill University
Mechanical Engineering Department
817 Sherbrooke Street
W. Montreal, PQ
Canada H3A 2K6
Tel: (514) 398-6288
Fax: (514) 398-7365
Email: misra@mecheng.lan.mcgill.ca

99-411 0830am **Altitude Control of a Tethered Lifting Probe for Atmospheric Research** –B.L. Biswell, J. Puig-Suari

A control algorithm is presented for an atmospheric sampling mission using a tethered lifting probe from the Space Shuttle in circular orbit. Lift from the probe is used to raise the altitude of the probe and reduce the drag on the system when not over the region of interest. The controller treats the tether attachment point as being quasi-inertial and uses successive loop closure to control the attitude and altitude of the probe. Control gains are derived using classical methods with a simple rigid-tether model. Robustness is verified with a flexible tether model. Simulation results show system behavior including atmospheric disturbances and non-continuous thrust profiles on the orbiter.

99-412 0850am **Improved Tether Aerobraking Maneuvers Using a Lifting Probe** –B.L. Biswell, J. Puig-Suari

A lifting probe with moveable attachment point is applied to the traditional tether aerobraking maneuver. The control is designed to regulate tether tension, which is directly related to orbiter braking, without breaking the tether. Several tether cutting strategies are examined to achieve the minimum orbiter eccentricity without mission failure. The effects of trajectory perturbations and uncertainties, which can have a significant effect on the success of the maneuver, are examined.

99-413 **Advanced Tether Experiment Deployment Failure** – S.S. Gates, S.M. Koss, M.F. Zedd
0910am

The Advanced Tether Experiment (ATEx) was launched into orbit aboard the National Reconnaissance Office (NRO) sponsored Space Technology Experiment spacecraft (STEX) on October 3, 1998. ATEx was intended to demonstrate deployment and survivability of a novel tether design as well as controlled librational maneuvers. On January 16, 1999, after deployment of only 22 meters of tether, ATEx was jettisoned from STEX due to an out-of-limits condition sent by the experiment's tether angle sensor. This paper reviews the essential system design, presents the available flight data, suggests likely causes of failure, and lessons learned.

99-414 **A Study of the Deployment Dynamics of the ASTOR Tethered Satellite System** –
0930am A.P. Mazzoleni

This paper concerns analysis of the deployment dynamics of the Advanced Safety Tether Operation and Reliability (ASTOR) satellite; in particular, this paper studies the effects of parameter variations and initial conditions prior to deployment on the system dynamics and the ability of the satellite to fully deploy. The purpose of the ASTOR Satellite is twofold: (1) demonstrate the performance of the Emergency Tether Deployment (ETD) system which is designed to overcome the safety hazard caused by snags during tether deployment; (2) provide a platform for students to design and fly a variety of space-based experiments.

0950am - 1010am Coffee Break

99-415 **A Look at Tethered Satellite Identification Using Ridge-Type Estimation Methods** –
1010am D.A. Cicci, T.A. Lovell, C. Qualls

This paper will apply ridge-type estimation methods to the problem of identifying a tethered satellite using a short arc of observational data. Performance of the ridge-type filter will be evaluated for cases of differing tether lengths, tether orientations, types of observations being processed, and number of observations. Results will be compared to those obtained from classical filtering methods, which can provide inaccurate results using short-arcs of data, with special attention being given to the speed at which the identification of a tethered satellite can occur.

99-416 **Modeling Tethered Satellite Systems for Detection and State Estimation** - J.E. Cochran,
1030am Jr., S. Cho, T.A. Lovell, D.A. Cicci

Detection that a satellite in a two-body tethered satellite system is tethered using observations of that satellite alone over a relatively short period of time is more difficult when the system has a relatively short tether and/or significant libration. The estimation of the state of the system using the same information is also more difficult under these conditions. In this paper, two different formulations of the equations of motion of a two-body tethered satellite system with a "massless" tether are used to see which results in the best performance of detection and state estimation algorithms. The two formulations involve the use of different state variables. In the first the position and velocity of the observed satellite with respect to the center of the Earth and the relative positions and velocity of the other satellite are used. In the second formulation, we use the position and velocity of the observed satellite and its position and velocity relative to the center of mass of the system are adopted as variables. A three-stage detection/state estimation algorithm is then described and results for the two formulations and different tethered system parameters are presented.

99-417 **Tracking of the TiPS Tethered Satellite** - J. Barnds, M. Davis
1050am

This paper presents the post-mission findings of the Naval Research Laboratory's investigation into tracking and predicting the motion of a tethered satellite system: the Tether Physics and Survivability (TiPS) experiment. We provide a description of the analytic tools and methodologies developed for characterizing the dynamics of the system. A truth ephemeris has been produced using the combined observations of the satellite laser ranging (SLR) network, the US Space Command network and a dedicated radar operating on the Kwajalein Atoll (ALTAIR). The accuracy attainable from the component tracking networks is evaluated and various methodologies for improving their accuracy are described in detail, including a comparison of libration averaging and short-arc libration estimation. We include a discussion on the accuracy of the tether dynamics model used for tracking of the tether end-bodies.

Session 16 *1:30 PM – 4:10 PM*

Session Room: Columbia Ballroom
Session Title: Prediction/Uncertainty
Session Chair: Dr. Beny Neta
Naval Postgraduate School
Department of Mathematics
Code MA/Nd
Monterey, CA 93943
Tel: (831) 656-2235
Fax: (831) 656-2355
Email: bneta@nps.navy.mil

99-418 **Nonlinear Methods for the Propagation of Orbital Uncertainty** –S.R. Chesley, A. Milani
0130pm

One of the standard tools in astrodynamics is the linear propagation of orbital uncertainty, but there are cases when the linear assumption breaks down to the point where it must be abandoned because observational data is extremely sparse or is situated very far from the desired time of prediction. We describe two new methods which account for nonlinear effects in the prediction of confidence regions in the observational space. In particular, we will focus on the case of lost or nearly lost asteroids; however, the methods which we introduce can easily be applied to artificial bodies which have inadequate observational information, such as, for example, space debris or ``noncommunicating" satellites.

99-419 **Satellite Drag Error Growth Estimation** –F.R. Hoots, R.G. France
0150pm

Satellites of high value such as the space station are routinely monitored to determine possible close approaches with other orbiting objects. In general, these predictions will contain uncertainty. It is important to quantify this uncertainty so maneuvers will be performed only when necessary. A byproduct of the least-squares orbit determination solution is a covariance that can be used to estimate the prediction uncertainty in the future. Typically, such a covariance significantly underestimates the prediction uncertainty in an atmospheric drag environment. A new technique has been investigated which successfully captures the essential character and size of the unmodeled drag variation. This will provide a direct addition to the covariance matrix to account for drag.

99-420 **Improved Space Surveillance Network Observation Error Modeling and Techniques for**
0210pm **Force Model Error Mitigation** –W.N. Barker, S.J. Casali, C.A.H. Walker

Estimation algorithms must be able to accurately model the statistical properties of the observational data and control force model uncertainties. The possibility for improvement in these areas has been identified for the Cheyenne Mountain Operations Center (CMOC), which provides the orbit determination for the satellite catalog using Space Surveillance Network (SSN) observations. Currently, the sensor calibration performed by the CMOC on the SSN does not model geometric dependency or auto-correlation of observation error, though such systematic effects are now evident from external, precise data. Moreover, the present batch estimation used by the CMOC does not account for force model uncertainties.

99-421 **Covariance as an Estimator of Orbit Prediction Error Growth for Geosynchronous**
0230pm **Orbits** –K.T. Alfriend, S. Paik

In this paper, the performance of the covariance calculated by the least squares differential correction process in estimating the orbit prediction error growth for geosynchronous orbits is evaluated. The evaluation is performed for special perturbations and the Naval Space Command general perturbations model PPT3. A fit span of 30 days is used.

0250pm - 0310pm Coffee Break

99-422 **Covariance as an Estimator of Orbit Prediction Error Growth in the Presence of**
0310pm **Unknown Sensor Biases** –K.T. Alfriend, M.P.Wilkins

The procedures for determining biases for the sensors in the US Space Surveillance Network (SSN) show that there are biases that vary daily and probably track-to-track. These unknown biases have an effect on the quality or accuracy of the orbital elements determined by the least squares differential correction process (LSDC). The purpose of this paper is to develop a methodology for incorporating the effect of these unknown biases into the uncertainty estimate provided by the covariance. The methodology is evaluated on a simplified problem and then validated with simulations.

99-423 **Stardust Navigation Covariance Analysis** –P.R. Menon

Paper withdrawn.

99-424 **Monte-Carlo Analysis of Nonimpulsive Orbital Transfers Under Thrust Errors, 1** –
0330pm A.D.C. de Jesus, M.L.O. e Souza, A.F.B. de Almeida Prado

In this paper, we present the first part of a Monte-Carlo analysis of nonimpulsive orbital transfers under thrust errors. This was done as part of an extensive study conducted in three phases by Jesus under the supervision of the other authors of this paper. The first phase was the numerical implementation of a nonimpulsive trajectory optimization method and one example from Biggs and Prado; and of another example from Kuga et alli without thrust errors. The second phase was an extensive Monte-Carlo analysis on nonimpulsive orbital transfers under thrust errors. The third phase was a first algebraic explanation for some of the numerical relations found in the second phase. This paper emphasizes the first part of the second phase but mentions the other two phases. Its main results suggest and partially characterizes the progressive deformation of the trajectory distribution along the propulsive arc, turning 3-sigma ellipsoids into banana-shaped volumes curved to the center of attraction (we call them ‘bananoids’) due to the loss of optimality of the actual (with errors) trajectories with respect to the nominal (no errors) trajectory. A similar deformation, but due to initial condition gaussian errors, was shown by Junkins. As his plots also suggest, such deformations cannot be anticipated by covariance analysis on linearized models with zero mean errors which propagate ellipsoids into ellipsoids always centered in the nominal (no errors) trajectory. Its main results also characterize how close or how far are Monte-Carlo analysis and covariance analysis for those examples.

99-425 **Element Set Prediction Accuracy Assessment** – D. Snow, D. Kaya
0350pm

HQ Space Warfare Center (HQ SWC/AES) has been tasked to provide an automated system that can produce element set accuracy statistics for the entire Space Control Center(SCC) general perturbations catalog of satellites. An in-house tool known as Modeling Astrodynamics Element Set Trajectories and Orbits (MAESTRO) is being enhanced to produce the required element set accuracy statistics. MAESTRO determines element set accuracy statistics by predicting element sets against a known truth and then performing a Least Squares fit of the residuals to produce a 1-sigma curve in each of the *UVW* components for either the satellite itself or, if there is insufficient data, for the satellite group. This approach of inferring the future performance by an assessment of past performance avoids the difficulty of building error models for each of the two thousand maintenance groups, but it is not without flaw (as your stockbroker will tell you). This paper will address the HQ SWC/AES approach, the process being used to analyze the quality of the element set accuracy statistics, and future ideas for improving these statistics.

SESSION 17 ***1:30 PM – 4:50 PM***

Session Room: Columbia Ballroom
Session Title: Attitude Estimation
Session Chair: Don L. Mackison
University of Colorado
ASEN Campus, Box 429
Boulder, CO 80309-0429
Tel: (303) 497-6417
Fax: (303) 492-7881
Email: mackison@spot.colorado.edu

99-426 **Spacecraft Angular Velocity Estimation Using Sequential Observations of a Single Vector**
0130pm –Y. Oshman, F. Dellus

A new approach is presented for estimating the angular rate of a tumbling, momentum wheel-equipped spacecraft from sequential measurements of a single directional vector. The method consists of a two-stage algorithm. The first stage is a deterministic algorithm that provides a coarse estimation of the satellite angular velocity. Combining the kinematics equations, corresponding to the apparent motion of the vector measurements in the satellite body frame, with the satellite dynamic equations of motion, enables the development of an almost one-to-one relation between the vector measurements and the satellite angular rate. This relation is used to reconstruct the satellite angular velocity. The second stage is an extended Kalman filter which is initialized using the first stage crude estimates. This combination renders the filter accurate and robust. A simulation study, in which angular rates of up to 0.5 rad/s were implemented, is used to demonstrate the performance of the method.

99-427 **How to Estimate Attitude from Vector Observations** - F.L. Markley, D. Mortari
0150pm

The most robust estimators minimizing Wahba's loss function are Davenport's *q* method and the Singular Value Decomposition method. The *q* method, which computes the optimal quaternion as the eigenvector of a symmetric 4×4 matrix with the largest eigenvalue, is faster and produces more convenient output. The fastest algorithms, the QUaternion ESTimator (QUEST) and the EStimator of the Optimal Quaternion (ESOQ2), are less robust since they solve the characteristic polynomial equation for the maximum eigenvalue. This is only an issue for measurements with widely differing accuracies, so these estimators are well suited to sensors that track multiple stars with comparable accuracies.

99-428 **Application of Extended Kalman Filter for Small Satellite Attitude Estimation** –
0210pm Z. Fengqi, L. Hui, Z. Jun

In this paper, an attitude estimation algorithm suitable for three-axis stabilized small satellite is developed and simulation tested for obtaining precision attitude determination. Current small satellite attitude determination system of high accuracy typically relies on a stellar-inertial approach, in which the selected hardware consists of: (1) a set of rate-integrating gyros mounted along three body axes; (2) two skew star sensors; (3) if necessary, other redundant attitude sensors such as sun sensors, earth sensors; and (4) a data processor - onboard computer. The rate-integrating gyros are used as the short-term reference, providing real-time and continuous attitudes as well as attitude rates. Because of the existence of gyro drifts, measurement errors and integration errors, star sensors must be used as the long-term reference and supply corrections to attitude and gyro drift estimates. The gyro and star sensor measurements are combined, and a Kalman filter is employed to obtain the most accurate attitude estimates and update the gyro drift estimates. Such attitude estimation technique has been proved most applicable for small satellites. The attitude estimation algorithm developed in this paper is a six-state-variable extended Kalman filter (EKF) which employs aforementioned attitude estimation technique.

99-429 **A Fully-Redundant Set of Single-Axis Accelerometers** –D. Schaechter
0230pm

The linear acceleration at a point fixed on a rigid body may be deduced from a set of three single-axis accelerometers located at this point. The minimal number of accelerometers needed to derive this information in the event of a failure of any one of these accelerometers is four. One such symmetric arrangement uses four accelerometers whose sensitive axes are aligned with lines emanating from the center of a tetrahedron to its vertices. When the accelerometers cannot be located at the point whose linear accelerations are to be derived, coupling of the rotational and translational motions requires the use of additional accelerometers. The purpose of this paper is to provide a solution to this latter problem with the additional requirement of protecting against single point failures.

0250pm - 0310pm Coffee Break

99-430 **Mass-Properties Estimation for Spacecraft with Powerful Fuel Damping** –M. Peck
0310pm

This paper considers the problem of on-orbit mass-properties identification for spinning spacecraft with unknown energy-dissipating torques. These torques arise when, for example, structural flexure or fluid motion damps nutational energy. For such systems, many existing identification methods are inadequate; they start from Euler's equation with the assumption that all torques on the rigid spacecraft body are known with sufficiently small errors. In contrast, the present analysis is based on angular-momentum conservation, which holds even in the presence of unknown energy dissipation. The proposed method is shown to provide accurate answers for systems with nearly constant mass properties but high internal damping, where existing methods fail.

99-431 **DIGISTAR II Micro-Star Tracker: Autonomous On-Orbit Calibration and Attitude**
0330pm **Estimation** –G. Ju, T. Pollack, J.L. Junkins

A recursive on-orbit calibration approach for the recently proposed DIGISTAR II split field-of-view (FOV) star tracker, which use one mirror deflecting the sensor FOV to two orthogonal directions, is presented. This calibration study includes the corrections to effective focal lengths, focal plane offsets, lens distortion and electronic aging nonlinearities. Night sky ground-based experiments have also performed to generate image data for validating this approach and to provide a realistic set of data for optimizing this algorithm. In addition, the novel features of DIGISTAR II are described with the accuracy test. Final best attitude estimation approach based upon the recursive calibration is then summarized.

99-432 **Multiple Approaches for GYRO Calibration of INTELSAT Satellites** –K.V. Raman
0350pm

INTELSAT is the International Satellite Organization that provides voice, video and data transmission and distribution services to its 143 member nations through a fleet of 19 satellites. INTELSAT has performed transfer orbit operations on most of its satellites. While many of them were spin stabilized in transfer orbit, several three-axis stabilized missions have also been performed. While three-axis stabilized missions are more complex, they offer advantages such as: (1) common attitude control hardware for transfer-orbit and on-orbit phases, and (2) mass reduction and ease of spacecraft integration due to the removal of dynamic spin balancing. Missions like these require on-board gyros for rate and position reference during Orbit Transfer maneuvers. A significant source of error in these maneuvers is the attitude pointing error caused by the gyro drift. This drift has to be accurately estimated and compensated for in the attitude control loop. While some manufacturers offer on-board autonomous gyro calibration, INTELSAT spacecraft have relied on ground calibration of the gyro drift. This paper discusses multiple approaches for estimating the gyro drift from the ground. These approaches are based on: (1) an Extended Kalman Filter, (2) a Batch least squares method, and (3) a Linear curve fit of the integrated gyro angle. The gyro bias estimates are then compared with ground test values. Flight data from an INTELSAT satellite launch are used to evaluate different schemes for their accuracy and complexity. Conclusions are then drawn to determine the best approach for future missions.

99-433 **Precision Attitude Estimation with Star Trackers: Experience, Error Models and Their**
0410pm **Interpretation** –D.R. Haley

Many modern space missions have attitude knowledge requirements of the order of a few arcseconds or less. It is generally accepted that the most accurate and precise absolute inertial reference system for spacecraft attitude estimation is a modern star tracker, and a number of commercial organizations market more or less standard designs for such products which are advertised as having pointing accuracy of one to ten arcseconds. Descriptions of real data studies and simulations of realized accuracies are presented along with some of our experiences in interpreting calibration specifications.

99-434 **Landsat-4 Attitude Recovery and Other Experiences with Aging Landsat Spacecraft –**
0430pm P.K. Misra, J. Sedlak, J. Gregory, S. Jones, R. Smilek, D. Lorentz, R. Bowser, P. Mylod

The latest of the series, Landsat-4 and ~~5~~ were launched in 1982 and 1985 for Earth Observation Service. On June 10, 1997, Landsat **4** was configured to a safe survival mode after the redundant roll gyro replaced the failed primary. Star observations ensure integrity of on-board gyro-propagated inertial reference. Failed Earth Sensors, low beta angle sun-sensor operation and ultered star tracker characteristics, made it a difficult recovery operation. Innovations including customization of star catalog to match the aged trackers led to recovery.

SESSION 18 *1:30 PM – 4:50 PM*

Session Room: Columbia Ballroom
Session Title: Guidance and Navigation
Session Chair: Bobby Williams
M/S 301-125J
Jet Propulsion Laboratory
4800 Oak Grove Drive
Pasadena, CA 91109-8099
Tel: (818) 354-7422
Fax: (818) 393-6388
Email: bobby.williams@jpl.nasa.gov

99-435 0130pm **Image Navigation Based on Landmark and Earth Edge Measurements** – K.M. Ong, A.A. Kamel

A new algorithm to navigate imagery data of a radiometer onboard a 3-axis stabilized geosynchronous satellite is presented. The algorithm determines the combined radiometer and satellite time varying attitude in near-real-time by ground processing of the radiometer imagery data. Reduction of the data yields pixel coordinates of landmarks with known geographic coordinates and the coordinates of the Earth's limb. A sequential batch algorithm is used to process these measurements to yield precise attitude information for accurate image navigation. Computer simulation verifies the stability of the algorithm even in the presence of rapid attitude variations making it suitable for practical implementation.

99-436 0150pm **Theory and Flight Performance of the Magnetic Orbital Gyrocompass** – J.W. Fisher, W.R. Trochman, R.C. D'Ambrosio

The orbital gyrocompass concept is generalized to a system with the horizon sensor replaced by *any* direction sensor. The analysis is based on a vector formulation of the governing equations that provide a physical explanation for attitude and gyro drift observability. It is shown that all three components of gyro drift are observable *except* in the classical configuration (with horizon sensor and non-maneuvering spacecraft). The new concepts are illustrated by the *magnetometer gyrocompass*, implemented on the LM-700 series of low cost satellite buses. Flight telemetry data correlate well with the theoretical development and pre-flight predictions. Accuracy is better than 1 degree 1σ, limited primarily by errors in the magnetometer and in the magnetic field model.

99-437 **SP-Search Star Pattern Recognition for Multiple Fields of View Star Trackers** –
0210pm D. Mortari, J.L. Junkins

This paper contains an extension to multiple fields-of-view star sensors (DIGISTAR II, III), of the Spherical Polygon Search (SPS) algorithm for Star-ID problem. The method finds the admissible stars with a given (k) direction using a star pair basis (i and j). This is accomplished using the k-vector technique which avoids the searching phase, the heaviest load of the Star-ID procedures. When only one star is found admissible, then the [i, j, k] stars are identified. Then, using this information, the remaining stars are also identified. Meaningful plots show the method performances.

99-438 **Earth-Sun Attitude Sensor: Hardware Design and Ground Tests** –D. Mortari, S. Gigli
0230pm

The Earth-Sun and the Moon-Sun Attitude Sensor (ESAS, MSAS) concepts are based on the same idea: to estimate the Sun direction from the Earth or the Moon image observation, respectively. Both sensors, which can be used in a standalone configuration for attitude estimation, are at the feasibility study phase. However, while the MSAS needs of the Moon tracking, the ESAS avoids the tracking problem by mounting it along an Earth-pointing direction onboard spacecraft. This paper describes comparative and differential ground tests for ESAS-MSAS using two cameras, a notebook, and a GPS receiver.

0250pm - 0310pm Coffee Break

99-439 **Autonomous Navigation Using Celestial Objects** –D. Folta, C. Gramling, D. Leung,
0310pm S. Belur, A. Long

Goddard Space Flight Center is currently developing autonomous navigation systems for satellites in regimes in which use of the Global Positioning System or Tracking and Data Relay Satellite System is not feasible. This paper assesses the feasibility of using standard spacecraft attitude sensors and communication components to provide autonomous navigation for high-Earth-orbiting (HEO) and libration-point-orbiting (LPO) missions. Performance results are presented as a function of sensor measurement accuracy, measurement types, measurement frequency, initial state errors, and dynamic modeling errors. This analysis indicates that real-time accuracies ranging from 5 kilometers root mean square (RMS) for a 3 by 20 Earth Radii HEO satellite to 20 kilometers RMS for a LPO satellite are achievable.

99-440 **Calibration of Radiometric Data for General Relativity and Solar Plasma during the**
0330pm **Near-Earth Asteroid Rendezvous Spacecraft Solar Conjunction** - C.J. Weeks, J.K. Miller

A radio signal passing close to the Sun is affected by the charged particle environment of the solar plasma and the increased path length associated with the curvature of space predicted by general relativity. In January 1997, the Near-Earth Asteroid Rendezvous (NEAR) spacecraft experienced a deep solar conjunction. For a month before and after conjunction, the spacecraft was pointed at the sun and attitude operations were suspended. Since the NEAR spacecraft is equipped with two-way X-band Doppler and range and the spacecraft attitude was favorable, a high-precision spacecraft ephemeris was obtained. It is possible to solve explicitly for the total electron content of the solar plasma along the signal path by using the signal delay in the range data to calibrate the signal advance in the Doppler data. The result is a high-precision estimate of the general relativity delay, and of the total electron content along the signal path which can be used to develop a model of the solar plasma emanating from the Sun. The general relativity parameter gamma can be determined to less than 0.2 percent, which is competitive with the most accurate current verifications.

99-441 **Navigation Strategy for the Mars 2001 Lander Mission** –R.A. Mase, D.A. Spencer,
0350pm J.C. Smith, Dr. R.D. Braun

The MSP 2001 project will send an orbiter, a lander, and a rover to Mars in the 2001 opportunity. The lander will demonstrate precision landing at Mars by utilizing improved approach navigation and hypersonic aeromaneuvering. The guided entry will result in a landed footprint that is an order of magnitude smaller than the Mars Pathfinder and Mars Polar Lander ballistic entry footprints. This paper will focus on improvements in interplanetary navigation that will decrease entry errors and will reduce the size of the landed footprint.

99-442 **Navigation of Mars Mission** –NOZOMI –T. Ichikawa, T. Kato
0410pm

Nozomi mission is to send a spacecraft to our neighboring planet Mars. After orbiting Earth for more than 5 months, Nozomi departs for the Mars. Now cruising through interplanetary space after earth swing-by and injected, it will be finally injected into Mars orbit. The initial trajectory of Nozomi was the trans-lunar ellipse on which the spacecraft was injected through a short low-Earth parking. Eighteen orbit maneuvers were so far done from launch and two times lunar swing-by through the powered earth swing-by to finally kick the spacecraft out to the interplanetary field. We describe the method of the navigation which includes attitude dynamics for spin-stabilization type considering the maneuver estimations in this paper. And also, the evaluation and result of several points described by using covariance technique ring the maneuver estimations in this paper. And also, the evaluation and result of several points described by using covariance technique.

99-443 **Navigating Mars Global Surveyor through the Martian Atmosphere: Aerobraking 2 -**
0430pm P.B. Esposito, V. Alwar, P.D. Burkhart, S.W. Demcak, E.J. Graat, M.D. Johnston, B.M. Portock

The Mars Global Surveyor spacecraft was successfully inserted into an elliptical orbit around Mars on 9/12/97 with an orbital period of 45.0 hours. After two phases of aerobraking, the orbital period was reduced to 11.6 hours on 3/27/98 and to 1.97 hours on 2/4/99. Aerobraking, through an uncertain Martian atmosphere, was responsible for circularization of the MGS orbit. This paper describes: (1) the estimation of an atmospheric density model for every drag pass or periapsis-passage by analyzing doppler tracking data, (2) the generation of short-term, that is over one to several orbits, accurate atmospheric density predictions, (3) maintaining the spacecraftš orbit within upper and lower bounds of atmospheric density or dynamic pressure during each periapsis-passage, and (4) the prediction of accurate periapsis-passage times (Tp) over one to fifteen orbits.

Session 19 *Thursday, August 19, 1999*
8:30 AM – 11:10 AM

Session Room: Columbia Ballroom
Session Title: Orbit Dynamics
Session Chair: Paul W. Schumacher, Jr.
Department of the Navy
Naval Space Command
Mail Code VN63T
5280 Fourth Street
Dahlgren, VA 22448
Tel: (540) 653-5476
Fax: (540) 653-6148
Email: schumach@nsc.navy.mil

99-444 **Troubles with Lambert** –D. Sonnabend
0830am

In solving Lambert's problem, one must be prepared to cope with multiple iterative solutions, extraneous algebraic solutions, and during the iteration, wandering outside the permitted ranges of the variables. Once, when the calculations were manual, mere brainpower got you through the difficulties. Today, if we want to solve many such problems automatically, we must instruct a ‘high-speed idiot’ to apply the insight of a Gauss. This paper reduces Gauss' sector/triangle method to the solution of an equation in a single variable, and provides means to cope.

99-445 **Velocity Distribution of Satellite Breakups** –J.G. Miller
0850am

It is shown that the change in velocity, $\Delta\mathbf{v}$, of each piece of a satellite breakup causes a rotation of the plane of the satellite through an angle α about the radius vector, $\mathbf{r}$, at the time of the breakup. The angle α for each piece can be measured from the classical orbital parameters of inclination and longitude of the ascending node of the satellite and the breakup piece. The 3-dimensional period-eccentricity-α scatter plot of the breakup pieces shows the distribution of the shape of the perturbed orbits as well as the orientation of the perturbed orbital planes relative to the orbital plane of the satellite prior to its breakup. It is shown that this 3-dimensional scatter plot is a nonlinear transformation of $\Delta\mathbf{v}$ of the breakup pieces, which depends on the magnitude of the radius vector, r, and the velocity vector, $\mathbf{v}$, of the satellite at the time of its breakup. This transformation can be analytically inverted to solve for $\Delta\mathbf{v}$. This inverse transformation is applied to an actual satellite breakup to obtain the distribution of $\Delta\mathbf{v}$ of the pieces.

99-446 **The Gravitational Potential of a Body in Terms of Ellipsoidal Harmonics** - L.A. Cangahuala

Paper withdrawn.

99-447 **Third Body Short-Period Terms in Analytic Orbit Prediction** - C.C. Seybold
0910am

This paper details a project to analytically recover third body short-period terms from the results of Long-Term Orbit Predictor (LOP) runs using a doubly-averaged potential. Rather than the more traditional averaging methods, the code uses averaging-by-inspection, where the short-period terms are removed by choosing indices in the potential that set the mean anomaly coefficient to zero. The importance of being able to recover these third body terms within LOP was realized when the Mars Global Surveyor aerobraking phase had to be re-analyzed due to the deployment problems with a solar array.

99-448 **Relation between Osculating and Mean Orbital Elements in the Case of the Second Order**
0930am **Theory** - E. Wnuk

The paper presents a solution of a problem of relation between mean and osculating elements based on the Mersman (1970) algorithm for the Hori-Lie perturbation method. The inverse transformation from osculating into mean elements has the same general form as the direct transformation from mean into osculating elements but with the opposite sign of a generator. The both transformations are performed with the same accuracy. Taking into account appropriate number of geopotential coefficients and an appropriate order of perturbations one can realize the both transformations with a given accuracy. The algorithm presented in the paper bases on a general form of the theory of the second order geopotential perturbations (Wnuk, 1995) which take into account arbitrary degree and order zonal and tesseral coefficients.

0950am - 1010pm Coffee Break

99-449 **On the Improved Orbit Stability Criterion Around An Irregularly Shaped Body** –
1010am J. Kawaguchi, S. Yoshikawa

The orbit stability around a small celestial body is discussed. The stability criterion was proposed and demonstrated at this conference last year. What this paper describes is the extension of it including the discussion on the perturbation effect. What is characteristic of the stability around an irregularly shaped body is, as well known, the resonance with respect to the spin (or rotation) frequency. As the past researches strongly pointed out, the strong instability occurs when the orbital period coincides with that of the body rotation. This resonance depends on the inclination angle and can be circumvented when it is 180 degrees. What the authors developed at this conference last year concluded that the strong instability is also found at the other orbit-to-rotation period ratio points than the trivial ratio of one (unit). The paper last year

presented how such instability strength is assessed and provided with the new criterion. It well accounted for the numerical simulation results. However, a rigorous inspection still indicates that the unstable period-ratio points do not perfectly agree with those predicted by the analytical criterion given in the previous paper. The paper this year makes an attempt to narrower the difference between the simulation and the analysis. The major contribution of the secular perturbation was found to shift the unstable points and the results qualitatively agree with the numerical examples. And the paper also gives a strategy of inserting the spacecraft around such an object, taking the advantage of the properties obtained through the improved criterion.

99-450 1030am **Dynamical Issues Associated with Relative Configurations of Multiple Spacecraft Near the Sun-Earth/Moon L_1 Point** - K.C. Howell, B.T. Barden

The recent interest in applying Dynamical Systems Theory to spacecraft trajectory design in the three-body problem has resulted in new insights. In addition to increased understanding of the rich dynamics in this region of space, the opportunity for new mission concepts has emerged as well. One proposed option is that of flying multiple spacecraft in some specified relative configuration near the collinear libration points. In this investigation, some fundamental issues associated with this concept are explored further. Beginning with a review of the natural dynamics observed on tori that envelope halo orbits in the circular restricted problem, similar motions are pursued in the more complex dynamical model that includes ephemerides for the positions of the Sun, planets and moon. Additionally, criteria are established to evaluate the fitness of the formation as it evolves.

99-451 1050am **The Genesis Trajectory and Heteroclinic Cycles** –M.W. Lo, W.S. Koon, S.Ross, J. Marsden

The Genesis Mission will be NASA's first robotic sample return mission. The purpose of this mission is to collect solar wind samples for two years in an L_1 halo orbit and return them to the Utah Test and Training Range (UTTR) for mid-air retrieval by helicopters. This requires the Genesis spacecraft to make an excursion into the region around L_2. This transfer between L_1 and L_2 requires no deterministic maneuvers and is provided by the existence of so-called heteroclinic cycles. The Genesis trajectory was designed with the knowledge of the conjectured existence of these heteroclinic cycles. We now have provided the first semi-analytic construction of such cycles. The heteroclinic cycle provides several interesting potential applications for future missions. First, it provides a rapid low-energy dynamical channel between L_1 and L_2 such as used by the Genesis mission. Second, it provides a dynamical mechanism for the temporary capture of objects around a planet without propulsion. Third, a thorough understanding of this dynamics is essential for an optimal design of any constellations to study the magnetosphere region. Lastly, an understanding of the resonance structure of this dynamical regime may provide new strategies for low energy planetary captures at Mars or Europa.

SESSION 20 ***8:30 AM – 11:50 AM***

Session Room: Columbia Ballroom
Session Title: Attitude Dynamics
Session Chair: John E. Cochran, Jr.

Auburn University
Department of Aerospace Engineering
Auburn University, AL 36849-5338
Tel: (334) 844-4874
Fax: (334) 844-6803
Email: jcochran@eng.auburn.edu

99-459 **Relative Equilibria of Orbiting Gyrostats** – C.D. Hall, J.A. Beck
0830am

The steady motions of an earth-oriented satellite in a circular orbit are represented mathematically by equilibrium solutions to the equations of motion expressed in an orbiting reference frame. For a rigid satellite, the classical analysis identifies twenty-four equilibria corresponding to all possible positions where the principal inertia axes are aligned with the local vertical, tangential, and orbit normal. Introduction of angular momentum by addition of symmetric rotors allows for a two-parameter family of equilibria. The current work applies noncanonical Hamiltonian methods to examine this result in light of limitations inherent in the classical approach.

99-456 **Translational Motion of Gravity Gradient Stabilized Satellites Due to Attitude Oscillation**
0850am – S. Ueno

This paper discusses on translational motion of gravity gradient stabilized satellites. The satellites cause rotational oscillation by gravity gradient torque. This oscillation, however, is different from the oscillation of pendulums on the ground because there is no supporting point and the motion is described in a rotating frame. The numerical simulations show complicated translational motion caused by initial attitude error. Approximated equations derived from conservation laws make clear the reason of the motion. The translational motion depends on the pitch rotational oscillation, which causes due to the initial disturbance on not only pitch angle but also roll angle.

99-454 **Object-Oriented Modeling of the Dynamics of Flexible Multibody Systems** –B. Min,
0910am A.K. Misra, X. Cyril, V.J. Modi, C.W. deSilva

This paper provides a theoretical base for modeling flexible multibody systems using an object-oriented approach. Several objects are developed, for example: frames, links, joints, inertia, etc. Modeling is achieved by the interconnection of the object modules in a simple and clear manner. In order to conveniently accommodate body flexibility as well as maintain the modular representation for each individual body in a system, the formalism is based on the Euler-Lagrange method in conjunction with the natural orthogonal complement of the kinematic constraint. Modification of the architecture of the system can be accommodated quite easily in this modeling process. Several simulation examples are presented in the paper.

99-457 **Effects of Internal Mass Flow on the Attitude Dynamics of Variable Mass Systems** -
0930am F.O. Eke, E.M. Eke

Analytical methods used in the study of variable mass systems include the simple particle dynamics approach, the so-called control volume method, the force replacement method, and, more recently, the constraint relaxation approach. This paper compares results obtained by the various analysis methods, and indicates the circumstances under which each method is appropriate. Since the main difference between the various approaches lies in the level of detail with which they capture internal mass flow, the focus is on the effects of internal mass flow on the dynamics of variable mass systems.

0950am - 1010am Coffee Break

99-460 **A Simplified Variation of Parameters Formulation of Euler's Equations of Motion of an**
1010am **Arbitrarily Torqued Asymmetric Rigid Body** –J. Mitchell, D. Richardson

A normalized form of Euler's equations is rewritten using a variation of parameters approach with amplitudes and angular displacement as parameters. This new form is compact and yields a much more accurate numerically integrated solution over longer simulation intervals than a conventional integration of Euler's equations. The complete variation of parameters formulation involves the classical Jacobian elliptic functions as well as standard elliptic integrals. Because this formulation is developed in the fixed reference frame of the body's principal axes, these variation of parameters equations can be simply grouped with the dynamical equations for rotation, i.e., the variation of the Euler-Rodrigues parameters, to provide singularity free information about the attitude of the body in a local inertial reference frame. This formulation is then applied to a simple, gravity-gradient stabilized satellite to investigate error propagation behavior.

99-458 **Nonlinear Dynamics of a Viscously Damped Spacecraft with a Flexible Appendage and**
1030am **Time-Dependent Forcing** –A.J. Miller, G.L. Gray, A.P. Mazzoleni

We study the attitude dynamics of a spacecraft that is perturbed by a flexible appendage and oscillating sub-masses. We are interested in the nonlinear dynamics that occur for sets of the physical parameter values of the spacecraft when energy dissipation is present. Energy dissipation drives the spacecraft in a minor to major axis transition and is implemented by a viscously damped rotor. Via a Melnikov analysis, we derive an analytical test for chaos in terms of satellite parameters. Finally, we numerically simulate the full system of equations to ascertain the accuracy and applicability of Melnikov's method and to understand the range of possible motions.

99-452 **Velocity Pointing Errors Associated with Spinning Thrusting Spacecraft** –D. Javorsek II,
1050am J.M. Longuski

Due to the imperfection of spacecraft assembly, there always exist misalignment and offset torques during thrust maneuvers. In the case of an axially thrusting spin-stabilized spacecraft, these torques disturb the angular momentum vector in inertial space causing a velocity pointing error. Much insight can be gained by analytically solving the problem of time-varying torques and time-varying moments of inertia. We use approximate analytic solutions to suggest how the velocity pointing error can be reduced for some practical assumptions based on current technology. For example, in the case of solid rocket motors, a significant improvement in velocity pointing can be realized by judicious distribution of the propellant.

99-453 **Periodic Solutions for a Spinning Asymmetric Rigid Body with Constant Principal-Axis**
1110am **Torque** - R.A. Gick, M.H. Williams, J.M. Longuski

We analyze the motion of a spinning asymmetric rigid body subject to constant torque along one of the principal axes. Solutions for the angular velocity vector and the corresponding kinematic parameters are given in terms of Fourier series expansions. Three semi-analytic solution methods are presented and compared. Numerical simulations confirm that the solutions for the attitude motion can be as accurate as desired. When the applied torque is small, only a few terms are needed in the Fourier series.

99-455 **Rotational Motions of a Satellite in a Circular Orbit** - S-J. Ying
1130pm

The rotational motions of a space station in a circular orbit are studied. There are three possible rotations: pitching, rolling and yawing. The equations governing the motions are derived following E. Neal Moore's approach. Details are given in this study. Because the equations are nonlinear and complicated, no analytical solution is available. However, they can be solved numerically by Runge-Kutta method. In order to see the effects of parameters involved, different values for moments of inertia and different initial conditions are studied and presented in this paper.

SESSION 21 ***8:30 AM – 11:30 AM***

Session Room: Columbia Ballroom
Session Title: Asteroids and Comets
Session Chair: Victoria Coverstone-Carroll
University of Illinois at Urbana-Champaign
MC-236
306 Talbot Lab
104 S. Wright Street
Urbana, IL 61801
Tel: (217) 333-0678
Fax: (217) 244-0720
Email: vcc@uiuc.edu

99-461 0830am **Asteroid Close Encounters with Earth: Risk Assessment** –A. Milani, G.B. Valsecchi, S.R. Chesley

When an asteroid with an orbit close to the Earth's is observed only over a short arc, there is no way to accurately predict the future close approaches. The first approach could change the orbital period in such a way that the asteroid is injected into a resonance, allowing an even closer approach a few years later. By using both a qualitative theory of these resonant returns, and a systematic numerical investigation of alternate orbital solution, it is possible to identify future encounters and to assess their impact risk. This allows one to monitor the impact risk by all known Near Earth Asteroids. The occurrence of close approaches in cascades of resonant returns makes the potential problem of deflection in a non-approaching orbit highly non-trivial.

99-462 0850am **Predicting Close Approaches and Estimating Impact Probabilities for Near-Earth Objects** –P.W. Chodas, D.K. Yeomans

Future close approaches of Near-Earth Objects are computed via numerical integration of their contemporary orbits, which are determined from optical and radar astrometric measurements. Impact probabilities are estimated from uncertainties in impact-plane parameters at each close approach. Last year's news report suggesting that asteroid 1997 XF11 might collide with Earth in 2028 was quickly disproven using a linear analysis. Since then, nonlinear techniques have been developed which enable prediction of close approaches farther into the future, as well as estimation of their impact probabilities. The one-kilometer asteroid 1999 AN10, discovered this year, has an impact probability of several parts in a million in 2044.

99-463 **Determination of Eros Physical Parameters for Near-Earth Asteroid Rendezvous Orbit**
0910am **Phase Navigation** –J.K. Miller, P.G. Antreasian, J.D. Giorgini, W.M. Owen, B.G. Williams, D.K. Yeomans

Navigation of the orbit phase of the Near-Earth Asteroid Rendezvous (NEAR) mission will require determination of certain physical parameters describing the size, shape, gravity field, attitude and inertial properties of Eros. Prior to launch, little was known about Eros except for its orbit which could be determined with high precision from ground-based telescope observations. Radar bounce and light curve data provided a rough estimate of Eros shape and a fairly good estimate of the pole, prime meridian and spin rate. However, the determination of the NEAR spacecraft orbit requires a high-precision model of Eros and the ground-based data provides only marginal a priori information. In this paper, estimates of Eros physical parameters obtained from the December 23, 1998, flyby will be presented. This new knowledge will be applied to simplification of Eros orbital operations that will be conducted in February 2001. The resulting revision to the orbit determination strategy will also be discussed.

99-464 **Near-Earth Asteroid Rendezvous (Near) Revised EROS orbit Phase Trajectory Design** -
0930am C.E. Helfrich, J.K. Miller, P.G. Antreasian, E. Carranza, B.G.Williams, D.W. Dunham, R.W. Farquhar, J.V. McAdams

As a result of the unplanned termination of the deep-space rendezvous maneuver on December 20, 1998, the NEAR spacecraft passed within 3830 km of Eros on December 23, 1998. This flyby provided a brief glimpse of Eros, and allowed for a more accurate model of the rotational parameters and gravity field uncertainty. The application of the NEAR orbit phase trajectory design to the current best estimate of the Eros physical parameters is described in this paper. The resulting orbit is the prototype for the actual trajectory design to be performed upon arrival at Eros in February 2000. The trajectory is described and illustrated, and some of the problems encountered in the design and their resolution are discussed.

0950am - 1010am Coffee Break

99-465 **Preliminary Planning for NEAR's Low-Altitude Operations at 433 Eros** - P.G. Antreasian,
1010am C.E. Helfrich, J.K. Miller, W.M. Owen, B.G. Williams, D.K. Yeomans, J.D. Giorgini, D.J. Scheeres, D.W. Dunham, R.W. Farquhar, J.V. McAdams, A.G. Santo, G.L. Heyler

On February 14, 2000, NASA's Near-Earth Asteroid Rendezvous (NEAR) spacecraft will be placed into orbit around asteroid 433 Eros. The spacecraft will orbit Eros with increasingly lower altitudes as the one-year orbit phase progresses. This paper will provide preliminary plans for mission design and navigation during the last two months of the orbit phase, where several close passes to the surface will be incorporated to enhance the science return. The culmination of these close passes will result in the eventual landing of the spacecraft on the surface of Eros. These close flybys and landing designs will incorporate the preliminary navigation information obtained during NEAR's recent flyby of Eros on December 23, 1998.

99-466 **Rendezvous Mission to a First-Apparition Comet** –A. Friedlander, J. Niehoff
1030am

This paper describes the scientific objectives and measurements, trajectory performance, and spacecraft requirements for implementing a rendezvous mission to a parabolic-type comet making its first pass through the inner solar system. Advanced low-thrust propulsion of lightweight design is a needed capability, with SEP and solar sail flight systems examined as alternative options. Results presented emphasize the astrodynamics aspects of this mission, namely, detection of new comet targets, launch/arrival timing, trajectory characteristics, mass trades, and frequency of mission opportunities. Mass statements and cost estimates are also shown as indication of technical/programmatic feasibility.

99-467 **How Much Thrust, Energy, or Propellant Does it take to Guide a Natural Celestial Body?**
1050am –N. Markopoulos

Classical rocket guidance is generalized to the case in which the "rocket" is a large, natural celestial body (CB). The main qualitative difference with the classical case lies in the gravitational interaction between the CB and the ejected propellant. The achieved thrust is zero for ejection speeds, u, less than the escape velocity, v, from the CB. The theory presented is valid for arbitrary u. It examines both gravitational interaction effects and mass loss effects due to conversion into energy using the unifying framework of the Schwarzschild solution of general relativity. Gravitational interaction effects dominate when u is close to v, while effects of mass loss due to conversion into energy dominate when u is close to the speed of light. The practically most important result of the paper is the proof that to impart a fixed impulsive velocity change to the CB, by ejecting part of it as propellant, one needs a, nonzero, minimum amount of energy if the gravitational field of the CB is sufficiently large. This constitutes the most important qualitative difference with classical rocket guidance, in which case the corresponding minimum energy requirement is a trivial zero.

99-468 **Deviating an Earth-Pointing Asteroid under Lunar Effect** - J-S. Chern, D-L. Sheu
1110am

It has been found that when the initial relative speed is 10km/s and initial distance is 150 times of Earth's radius, a total impulse of 3.02 10^8 m/s is required to produce a miss distance of 2 times of Earth's radius for a 10m asteroid. It needs a typical large launching rocket to provide such a total impulse. The purpose of this paper is to study the total impulse required when the lunar effect is not negligible and must be considered. The possibility of deviating the Earth-pointing asteroid to the Moon will also be investigated. The lunar effect can be neglected if the Moon is 20°of arc length away from the asteroid. In this paper, the case for the Moon within 20°of arc length will be analyzed. Also, it is intuitive and easier to deviate the asteroid to the Moon when they are near enough

RECORD OF MEETING EXPENSES

AAS/AIAA
Astrodynamics Specialist Conference

Westin Alyeska Prince Hotel

Girdwood, Alaska

August 16-19, 1997

Name _____

Organization _____

Registration Fee: Member @ $190 _____

Non-Member @$250 _____

Student @ $25 _____

Non-members @ $290 _____
(and includes 1 year new membership in AAS,
does not apply to membership renewal)

Conference Proceedings @ $160 _____

Glacier Express Dinner Ticket @ $20 _____

TOTAL: _____

Technical Paper Purchases:

No. Papers	Cost	Date
_____	_____	_____
_____	_____	_____
_____	_____	_____
_____	_____	_____
_____	_____	_____